*Lockheed Martin*
*Missiles & Space- Huntsville*

LMSC P038190

# Advanced Transportation System Studies Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development Contract

## NAS8-39208
## DR 4

## Final Report

**Prepared by**
**Lockheed Martin Missiles & Space**
**for the**
**Launch Systems Concepts Office**
**of the**
**George C. Marshall Space Flight Center**

**July 1995**

# Advanced Transportation System Studies Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development Contract

## NAS8-39208
## DR 4

## Final Report
## Volume II
## Technical Results

**Approved:**

**James B. McCurry**
**TA-2 Study Manager**

***Lockheed Martin***
***Missiles & Space- Huntsville***

# Preface

The Advanced Transportation System Studies (ATSS) Technical Area 2 (TA-2) Heavy Lift Launch Vehicle Development contract, NAS8-39208, was led by the Missile Systems Division of Lockheed Martin Missiles & Space, and supported by principal TA-2 teammembers Lockheed Martin Space Operations (LMSO), Aerojet, ECON, Inc., and Pratt & Whitney. Addition technical task support was provided by Lockheed Martin Skunk Works (LMSW).

The ATSS TA-2 contract was managed by James B. McCurry, Lockheed Martin Missiles & Space (LMMS), and performed for Mr. Gary W. Johnson, Contracting Officer's Technical Representative (COTR), of the Launch Systems Concepts Office (Organization Code PT-51), National Aeronautics and Space Administration George C. Marshall Space Flight Center (MSFC).

The purpose of the TA-2 contract was to provide advanced launch vehicle concept definition and analysis to assist NASA in the identification of future launch vehicle requirements. Contracted analysis activities included vehicle sizing and performance analysis, subsystem concept definition, propulsion subsystem definition (foreign and domestic), ground operations and facilities analysis, and life cycle cost estimation. The basic period of performance of the TA-2 contract was from May 1992 through May 1993. No-cost extensions were exercised on the contract from June 1993 through July 1995.

This document is the final report for the TA-2 contract. The final report consists of three volumes:

| | |
|---|---|
| Volume I | Executive Summary |
| Volume II | Technical Results |
| Volume III | Program Cost Estimates |

Volume I provides a summary description of the technical activities that were performed over the entire contract duration, covering three distinct launch vehicle definition activities: heavy-lift (300,000 pounds injected mass to low Earth orbit) launch vehicles for the First Lunar Outpost (FLO), medium-lift (50,000-80,000 pounds injected mass to low Earth orbit) launch vehicles, and single-stage-to-orbit (SSTO) launch vehicles (25,000 pounds injected mass to a Space Station orbit).

Per direction from the TA-2 COTR, Volume II provides documentation of selected technical results from various TA-2 analysis activities, including a detailed narrative description of the SSTO concept assessment results, a user's guide for the associated SSTO sizing tools, an SSTO turnaround assessment report, an executive summary of the ground operations assessments performed during the first year of the contract, a configuration-independent vehicle health management system requirements report, a copy of all major TA-2 contract presentations, a copy of the FLO launch vehicle final report (NASA document with contributions from TA-2), and references to Pratt & Whitney's TA-2 sponsored final reports regarding the identification of Russian (NPO Energomash) main propulsion technologies.

Volume III provides a work breakdown structure dictionary, user's guide for the parametric life cycle cost estimation tool, and final report developed by ECON, Inc., under subcontract to Lockheed Martin on TA-2 for the analysis of heavy lift launch vehicle concepts.

Any inquiries regarding the TA-2 contract or its results and products may be directed at Mr. Gary W. Johnson, NASA Marshall Space Flight Center, (205) 544-0636.

## Acknowledgments

The TA-2 study manager wishes to acknowledge the outstanding working relationships that developed during the TA-2 study contract, involving a true team effort between each of the TA-2 participants. The TA-2 COTR, Mr. Gary W. Johnson, was instrumental in fostering a team-play environment between the NASA and contractor participants that resulted in an extraordinary amount of engineering analysis results being produced. The TA-2 participants were immersed in a very dynamic environment in which the scope of the launch vehicle analysis activities constantly changed, reflecting the extraordinarily dynamic events that were unfolding at NASA Headquarters during the period of March 1992 through June 1994. The following personnel, listed by participating organization, are gratefully acknowledged for their outstanding contributions to the TA-2 contract. In addition, special recognition is due Messrs. Keith Holden and Kevin Sagis for their unique and innovative contributions during the entire course of the TA-2 contract in the development of vehicle sizing tools and the assessment of vehicle performance, respectively.

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| Bob Kummerer | Base heating analysis |
| Jim McCurry | TA-2 study manager, vehicle definition and integration |
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| Louis Rustenburg | Systems engineering |
| Kevin Sagis | Vehicle simulation and performance assessment |
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| Arlene Reese | CAD drawings |
| Pat Scott | Ground operations assessments |

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| Larry Tanner | Russian propulsion assessment |

Jim McCurry, TA-2 Study Manager
Huntsville, Alabama
September 1995

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# 1.0 Introduction

The original charter of the Advanced Transportation System Studies (ATSS) Technical Area 2 (TA-2) Heavy Lift Launch Vehicle (HLLV) Development contract was to define and assess HLLV concepts that supported cargo and crewed missions to the Moon and Mars, as part of the Bush Administration's Space Exploration Initiative (SEI). Congressional support for SEI was slow to materialize, and culminated with the ellimination of funding specifically for SEI in the Fiscal Year 1993 budget. A recovery plan was divised by the Office of Space Flight Development at NASA Headquarters during the spring and summer of 1993 to define the complete picture of NASA's space transportation requiremens. Three study teams were formed, comprised principally of representatives from each of the NASA centers and the Department of Defense, to identify launch system requirements for three respective space transportation system architectures: to upgrade and evolve the Space Shuttle fleet to continue Shuttle operations through the year 2020 using existing technologies (Option 1), replace the Shuttle fleet in the 2005 timeframe with a series of expendable launch vehicles (ELVs) and crew/cargo return vehicle elements utilizing current technologies (Option 2), and to replace the Space Shuttle fleet in the 2005 timeframe with a fully reusable launch system utilizing advanced technologies (Option 3).

The Advanced Transportation System Studies (ATSS) Technical Area 2 (TA-2) Heavy Lift Launch Vehicle (HLLV) Development contract team was initially tasked to support the Option 2 team with the definition of ELV concepts, as was discussed in the executive summary of this document (Volume I). Examples of the vehicle configurations that were defined for Option 2 are provided in two of the ATSS contract summary presentations for the year 1993, as containted in Section 12 of this volume.

The Option 3 team, led Gene Austin of the Marshall Space Flight Center (MSFC), assessed various kinds of two-stage and single-stage, rocket-only and mixed-propulsion-cycle concepts. The Option 3 team ultimately down-selected to rocket-only, fully reusable, SSTO vehicles, and more specifically, focused on the definition and detailed assessment of a winged (or wing-body) vertical-takeoff/horizontal-landing SSTO concept that the Langley Research Center (LaRC) had divised. The TA-2 team was directed in June of 1993 to assess first-order design sensitivities of Single Stage to Orbit (SSTO) launch vehicle concepts that had not previously been addressed by the Option 3 team. The first seven sections of Volume II of this report present a detailed discussion of the significant results from TA-2's Option 3 support.

## Why SSTO?

A brief explanation of why SSTO concepts should be considered for new space transportation systems is in order, prior to further discussion of TA-2's SSTO design efforts. Classical rocket sizing equations based on the rocket equation have historically indicated that the combination of multiple launch vehicle stage elements, usually two to three stages, provides the "best" solution to accomplishing a given mission delta velocity ($\Delta V$). The definition of what constitues the "best" solution becomes a direct function of the figures of merit that are used in the assessment. Figure 1.0-1 illustrates the typical figures of merit that have historically been used in past launch system definition studies.

**Past Figures of Merit**

- Vehicle Size (Physical Dimensions)
- Gross Liftoff Weight
- Structural (Dry) Mass
- Propellant Mass Fraction (Mprop/Mtotal)
- Structure Mass Fraction (Mstr/Mtotal)
- Payload Mass Fraction (Mpl/Mtotal)
- Safety and Reliability
- Mission Model Requirements
- DDT&E Cost
- Life Cycle Cost
- Cost Per Flight

Figure 1.0-1 Past Figures of Merit

Figure 1.0-2 illustrates the primary figures of merit used in today's launch system studies.

**Today's Figures of Merit**

- DDT&E
- Recurring Costs
- Safety
- Reliability
- Operability
- Performance
- Programmatic Risk
- Way of Doing Business

Figure 1.0-2 Today's Figures of Merit

Historic subsidies by governments to develop new launch vehicles has masked the influence of economic forcing functions on launch system design, and has diluted the ability to incentivize operational efficiencies. The mission cost chargeable to payload customer is typically a function of several factors, such as: recurring fixed cost, as influenced by such things as the system infrastructure and ways-of-doing-business; cost due to vehicle size, as influenced by such things as materials selection and manufacturing methods; cost due to technologies and design, as influenced by such things as design complexity, integration, degree of resuse or refurbishment, and test and check-out; and design, development, test, and evaluation (DDT&E) cost amortization. In order for an SSTO concept to be valid, the concept must ballance the benefit of fewest number of stage elements with performance efficiency, operational efficience, and design complexity needed to accomplish the applicable mission requirements.

## TA-2 Approach

Four first-order SSTO design aspects were addressed by the TA-2 team: outer moldline considerations, major structural element layout, main propulsion propellant combinations, and main propulsion selection. Figure 1.0-3 summarizes the major steps of the approach that were used by the TA-2 team to define and assess SSTO concepts. In order to have an "apples-to-apples" comparison between TA-2's SSTO concepts and those of the Option 3 team, the groundrules, assumptions, mission requirements, and types of technologies that were used by the Option 3 team were used by Lockheed to define the wing-body SSTO concepts; the details of which are discussed in Section 5 of Volume 2.

- Concurrent Engineering brainstorming of first order design issues (pros/cons) between SSTO configuration types
  - -- Qualitative identification of major design weaknesses
- Baseline a common set of mission requirements, ground rules, and constraints and figures of merit
  - -- Bounds the design solution set
- Identify sets of vehicle configurations to be sized that will assess the relative benefit of different design solutions subject to the figures of merit
  - -- Propellant combination
  - -- Propulsion system
  - -- Structural materials
  - -- Major subsystem layout
  - -- Outer moldline shape
  - -- TPS types
  - -- Operations scenarios
- Size the vehicles, simulate ascent/entry trajectories
  - -- Performance
  - -- Aerodynamic heating
  - -- First order loads
  - -- Flight mechanics
- Resize as needed

Figure 1.0-3 SSTO Design Process

Lockheed developed SSTO vehicle sizing tools that were calibrated against know sizing methods used by LaRC in the definition of NASA's wing-body, integral tank, tripropellant SSTO concept. A common sizing methodology was used by Lockheed during the assessment of each configuration type. Unfortunately, TA-2 funding was depleted prior to the completion of engine/propellant option trade studies, engine performance sensitivity assessments, and the assessment of enhancing technology sensitivities. Four major categories of fully reusable SSTO concepts were identified that were fundamentally defined by the vehicle's method of performing takeoffs and landings: horizontal takeoff and vertical landing (HTVL), vertical takeoff and landing (VTOL), vertical takeoff and horizontal landing (VTHL), and horizontal takeoff and horizontal landing (HTHL). From the work performed by the Option 3 team, and initial brainstorming of vehicle concept pros and cons performed by the TA-2 team, Lockheed decided to focus on the definition of VTOL and VTHL concepts, as illustrated in Figure 1.0-4. A design trade tree was defined for the VTOL and VTHL assessments that encompassed the majority of first-order design options that were possible, as shown in Figures 1.0-5 and 6. The primary focus of the SSTO configuration assessments was to compare side-entry VTOL concepts against lifting body VTHL concepts. A wing-body VTHL SSTO configuration was also sized and compared to the Option 3 team's initial wing-body concept as a calibration point for Lockheed's SSTO sizing tools. Time and budget limited the TA-2 team's ability to assess further design options. Figures 1.0-7 through 9 illustrate the respective three types of SSTO concepts.

The TA2 team used a similar integrated approach in defining and assessing candidate SSTO concepts as was utilized during the expendable launch vehicle assessments, in which subsystem-independent and subsystem-dependent vehicle design goals were balanced against the following first-order design drivers:

- Basic sizing and performance capability
- Definition of the vehicle's outer moldline
- Shroud/payload concept
- Stage propellant tank design
- Vehicle construction/manufacturig methods

- Primary structure materials selection
- Intertank/interstage design
- Stage thrust structure design
- Propellant feed subsystem design
- Main stage propulsion type

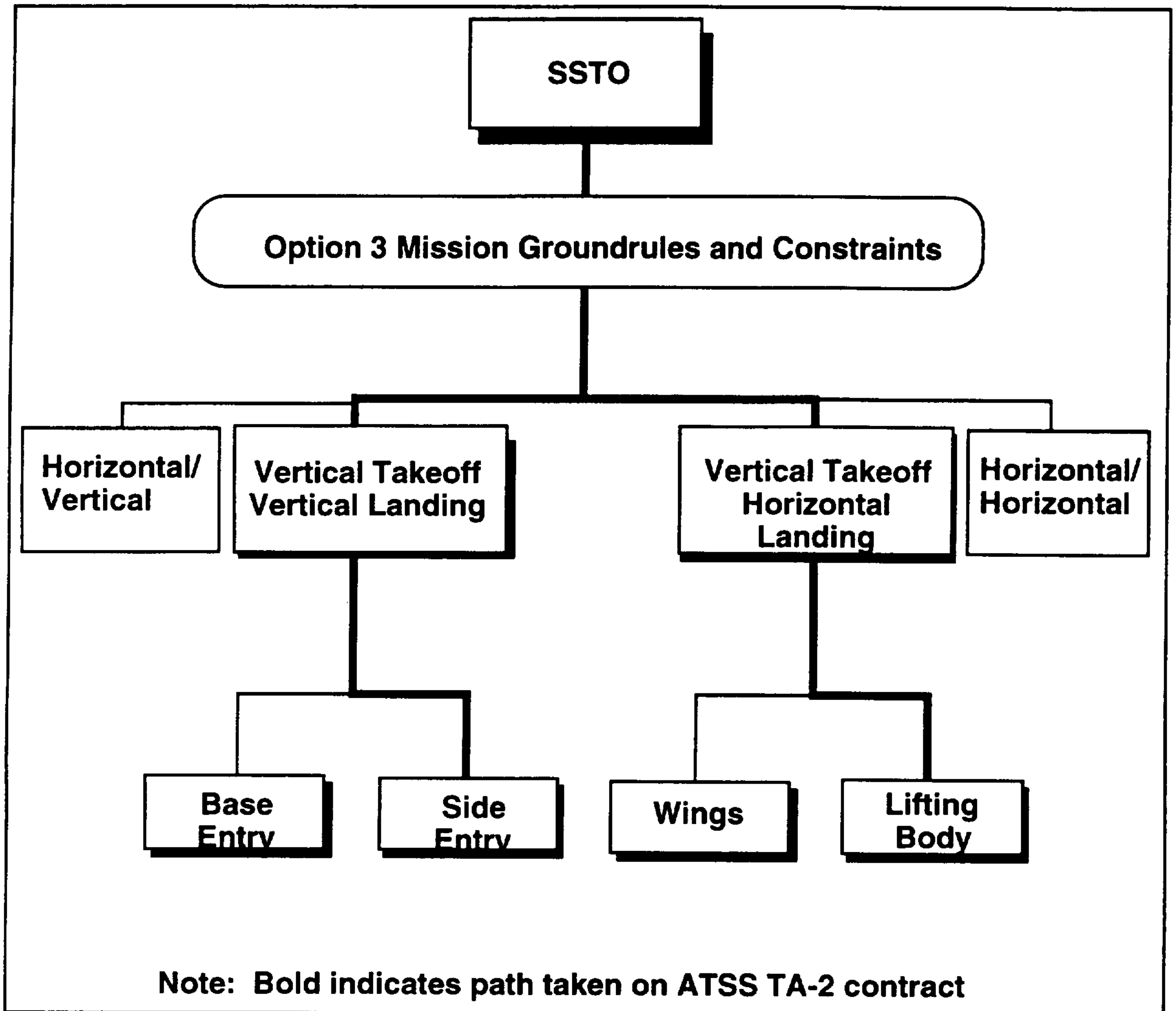



**Figure 1.0-4 Single Stage to Orbit Vehicle Design Path**

Figures 1.0-10 through 22 summarize the significant subsystem-independent and subsystem dependent design goals that were utilized during the TA-2 SSTO configuration assessments. Many of the design goals are applicable to any class of advanced transportation system.

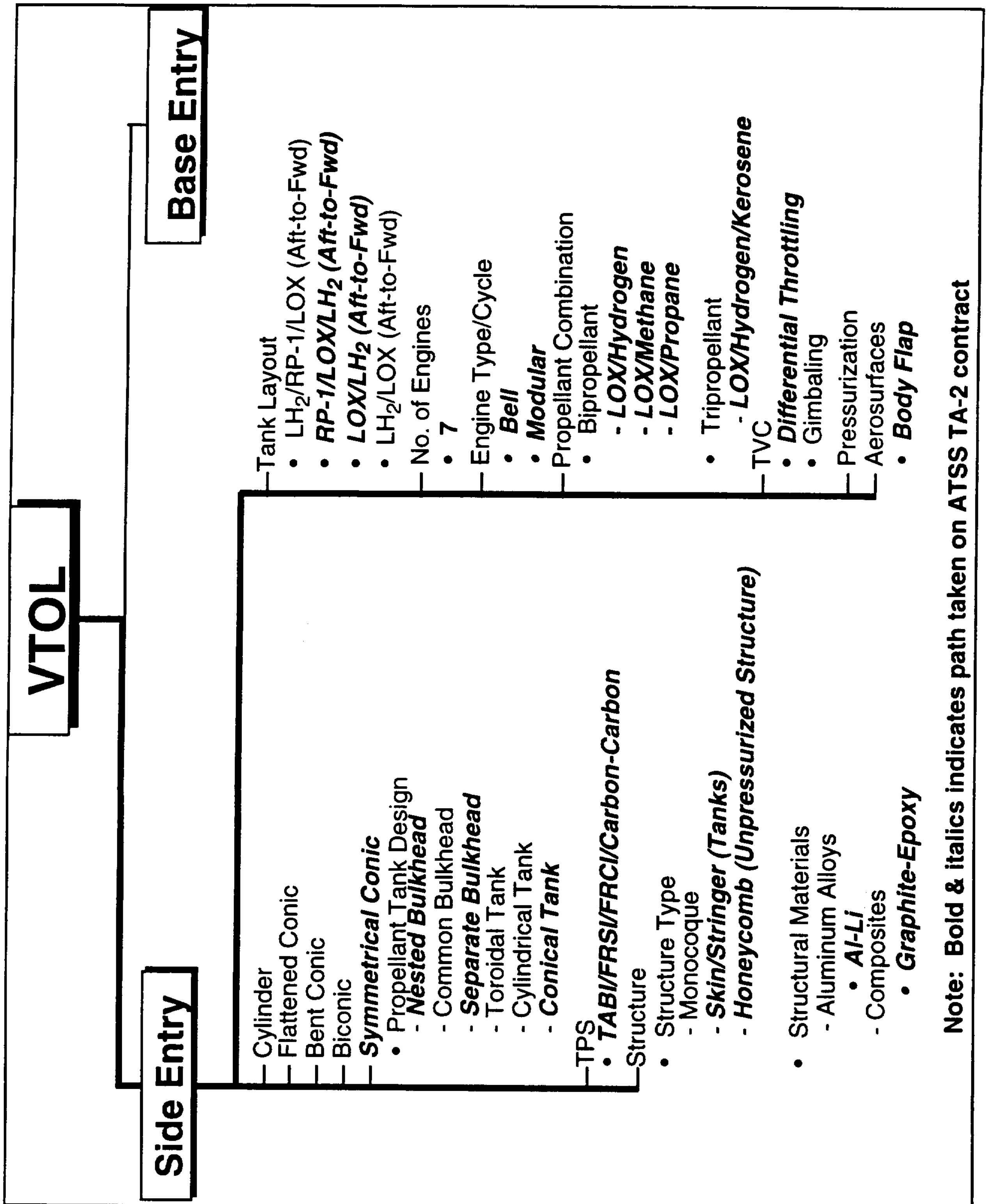



Figure 1.0-5 VTOL Vehicle Design Path

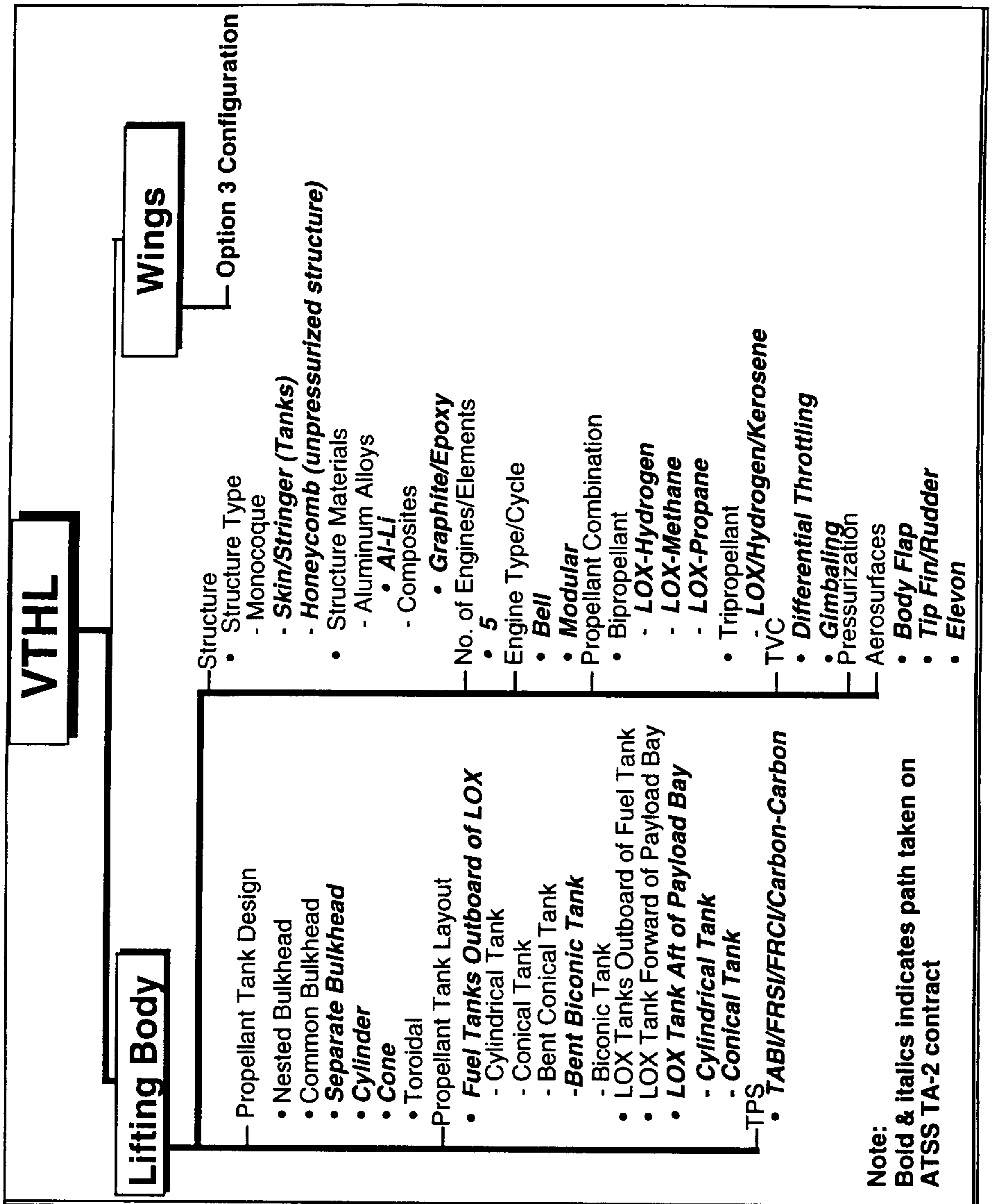



**Figure 1.0-6 VTHL Vehicle Design Path**

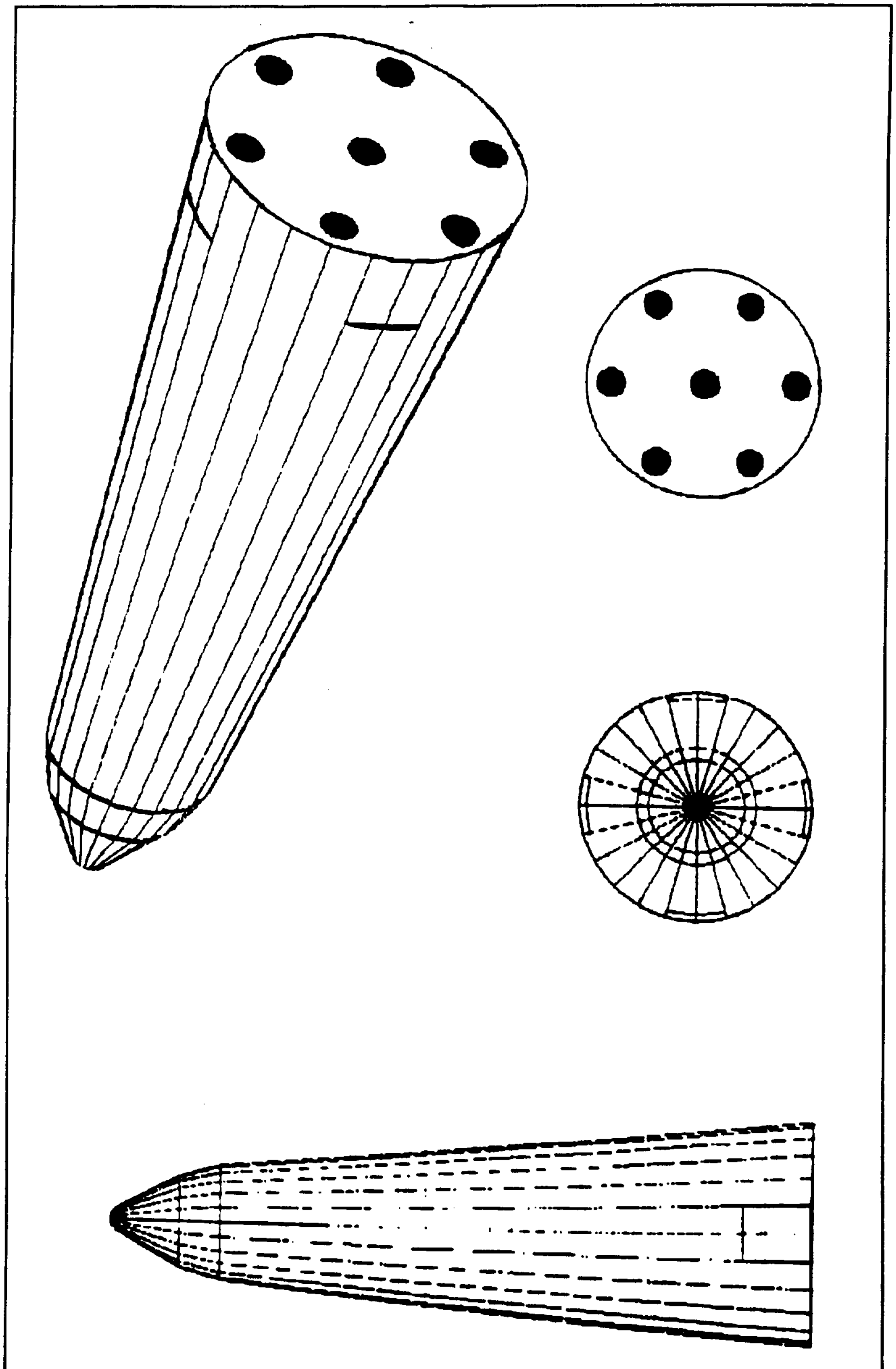


Figure 1.0-7 VTOL Concept

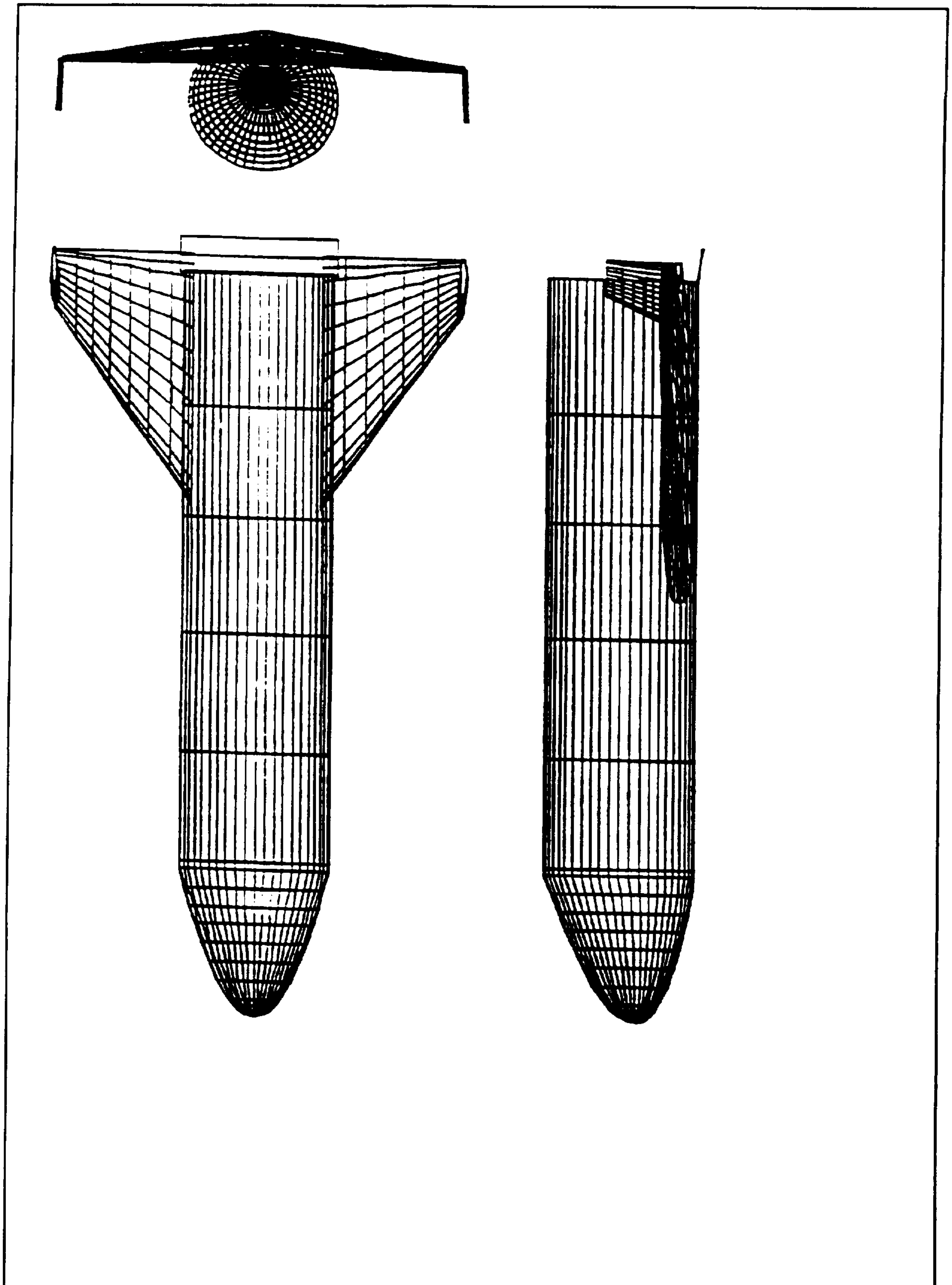


Figure 1.0-8 Wing-Body VTHL Concept

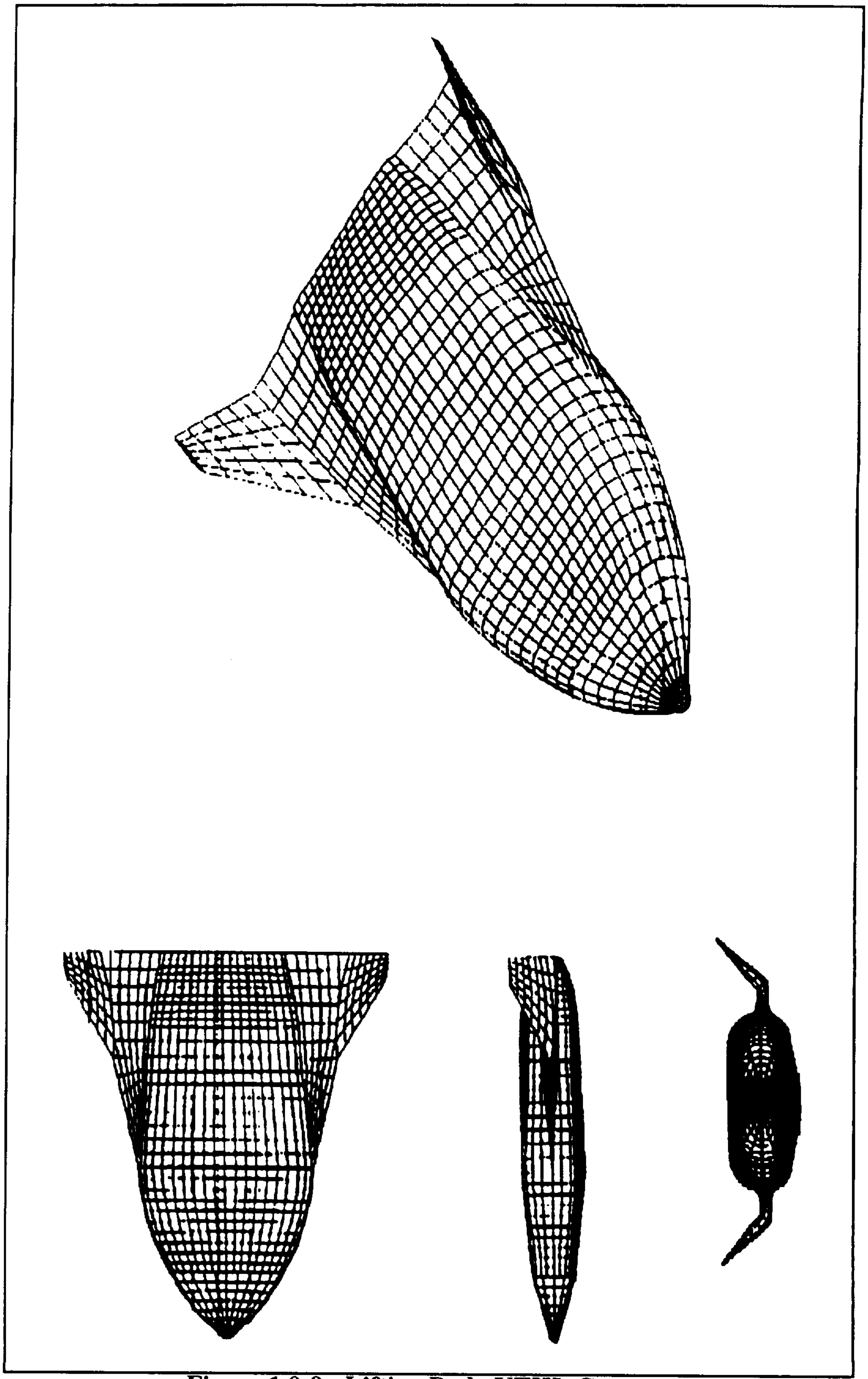


Figure 1.0-9 Lifting Body VTHL Concept

*Subsystem Independent Design Goals*

- Minimize number of subsystem-to-subsystem functional interfaces
- Minimize to maximum extent possible all Criticality 1 failure modes (loss of crew or vehicle)
  - Strive for conversion of Crit 1 failure modes to Crit 1R/2 or Crit 1R/3 (dual redundant or triple redundant
  - Based on safety and cost of failure
- Minimize to extent possible all Crit 2 failure modes (loss of mission)
  - Strive for conversion of Crit 2 failure modes to 2R/2 (dual redundant)
- Minimize to extent possible Critical Items (essential to mission or life)
  - Redundant items not capable of being checked out prelaunch
  - Loss of a redundant item is not readily detectable in flight
  - All redundant items can be lost by a single cause or event
- Maximize extent of line-replaceable units and ease of accessibility
- Maximize autonomous subsystem test, check-out, and health management
- Strive for VHM test/check-out down to LRU
- Allow for routine access and servicing
  - Minimize ground support equipment (GSE)
  - Eliminate "tail number specific" GSE
  - Service in shirt-sleeve environment
- Avoid use of hazardous fluids and gases to enhance operability

Figure 1.0-10 Subsystem-Independent Design Goals

Active Thermal Control

- Retain design options for 5-7 day mission duration
  - Current Space Station baseline used by Access to Space studies and typical for satellite retrieval/servicing missions
- Design options must handle five mission phases
  - Prelaunch
  - Ascent
  - On-orbit
  - Entry
  - Post-landing (which may include ferry flight)
- Should not have an abort mode specific ATC
  - Maximize mission use & minimize payload capability hit

Figure 1.0-11 Active Thermal Control Design Goals

Avionics

- Use open architecture
  - Independent of flight software language and CPU/DPU type
  - Distributed multiplexers/demultiplexers
  - Provide transparent component state-of-the-art upgrades
- Provide autonomous guidance, navigation, and control
  - Maximize use of mission independent flight software
  - Autonomous targeting for orbital insertion, on-orbit op.s, deorbit, and terminal area energy management
- Eliminate requirement for ground uplink capability for real-time reconfiguration
  - Studies show cost of autonomous capability less than verification, training, and flight controller op.s costs
- Eliminate requirement for flight-to-flight ground-based validation of onboard flight software
  - Validate on ground only when major software "Operational Increment" functional updates occur

Figure 1.0-12 Avionics Design Goals

*Crew Escape*

- Level I decision needed on basic crew escape requirement
  - "Vehicle itself" is the lifeboat
  - Varying degrees of crew escape are provided (seats, escape capsule, etc.)
  - Relative ability of vehicle's VHM or crew's capability to detect and act upon a life-threatening failure determines the failure modes protected
- Will be cost-prohibitive to eliminate all "black zones"
- Crew escape modules have historically been turned down for launch vehicles due to cost, weight penalty, and associated dynamics & flight control issues during module ejection

Figure 1.0-13 Crew Escape Design Goals

*Electrical Power*

- Electrical power generation requirements directly tied with input requirements of other vehicle subsystems
  - Degree and location of power conditioning a trade between complexity of EPS versus other subsystems
- Power generation will impose a major load on the active thermal control subsystem
- Classical trade between high power density, high complexity, more complicated maintenance & refurbishment of high density fuel cells versus APUs/generators, and batteries
  - Fuel cells have additional requirement of special grade reactants
- Operability trade pits all-electric vehicle against design having hydraulics and pneumatics
  - Industry/Govt. development studies of EMAs have cleared actuator technology hurdles; power systems now pacing items

Figure 1.0-14 Electrical Power Design Goals

*Environmental Control & Life Support*

- Initial decision to be made on type of crew cabin environment
  - Shirt-sleeve
  - Partial/full pressure suit
- Safety considerations rule out pure oxygen crew cabin environment
- Possible requirement to support EVA capabilities requires trade of EVA supportability (minimum/no pre-breathe) versus crew comfort and fire/leak contingencies
- Use of air-cooled equipment favors use of one-atmosphere in equipment bays
- Degree of ECLSS loop closure based on mission duration
  - Closed loop decreases consumables requirement but increases design complexity, power requirements, and lowers reliability
- Level I decision required regarding degree of crew interaction with in-flight ECLSS servicing
  - Crew involvement detracts from mission timeline & requires training

Figure 1.0-15 Evironmental Control & Life Support Design Goals

*Main Engine*

- Strive for maximum density-impulse to keep vehicle dry weight to a minimum
  - Helps to minimize number of required engines for vehicle thrust-to-weight goal
- Strive for lift-off thrust-to-weight ratio of 1.3-1.4, while balancing ascent thrust acceleration limiting (4-5 Gs) with gravity losses
  - Helps to minimize number of required engines
- Provide for active control of overboard mixture ratio to keep flight performance reserve low
- Strive for minimum NPSP capability to help minimize pressurization system and POGO suppression sizing
- Provide minimum of step-throttle capability for operational flexibility
- Allow for fuel depletion cutoff to eliminate fuel bias
- Allow for shutdown from any throttle setting for op.s flexibility

Figure 1.0-16 Main Engine Design Goals

*Main Engine Propellant Pressurization & Feed*

- Minimize number of piece parts to maximize reliability and operability
- Minimize number of flow control valves to maximize reliability
  - Utilize fixed orifice flow control where possible
- Minimize joints, flex lines, and avoid interconnects and cross-feed
  - Minimizes isolation valve count
  - Minimize leak potential and cost of leak checks
- Minimize complexity of pressurization subsystem
  - Avoid use of combustion gas driven heat exchangers (Crit 1 failure source)
- Maximize on-component VHM for prelaunch test/verification to minimize processing time
- Trade MPS modularity and single-element-checkout (with higher parts count) against integrated (minimum parts count) design requiring Main Propulsion Test Article certification
- Utilize spherical flanges to minimize load concentrations, damaged seals, and allow relaxed fit tolerances (as perfected by Russians)

Figure 1.0-17 Main Engine Propellant Pressurization and Feed Design Goals

*Mechanical*

- Requirement for unmanned vehicle operations will require autonomous activation of mechanical subsystems, thereby increasing complexity and decreasing associated reliability
- Trade study between ground uplink (as prime or backup) activation versus solely onboard autonomous for mission critical components
  - Trade of onboard redundancy level and alternate path redundancy
- Built-in-test via component resident VHM needed to significantly reduce preflight test and checkout
- Utilize electromechanical actuation in place of hydraulic or pneumatic actuation
- Strive for minimum number of mechanical components to increase vehicle reliability and operability

Figure 1.0-18 Mechanical Subsystem Design Goals

*Orbital Maneuvering*

- Size for ~1000 fps $\Delta V$ capability (insertion, on-orbit, deorbit)
- Avoid interconnects with RCS to enhance reliability
  - Minimizes isolation valve count
- Consider use of +X RCS for OMS function
  - Lowers vehicle complexity and operations costs versus performance
- Avoid dependency on helium blow-down pressurization to avoid helium leak contingencies
- Minimize need for active engine/propellant thermal conditioning to help minimize piece parts
- Allow nozzle gimbaling to increase burn attitude flexibility
  - RCS burn-to-attitude serves as back-up to gimbaling

Figure 1.0-19 Orbital Maneuvering Subsystem Design Goals

*Passive Thermal Control*

- Allow weather penetration for outer moldline PTCS
  - Enhances operability while maintaining vehicle safety/integrity
- Allow capability to "patch" repairs to outer moldline PTCS
  - Enhances operability
- Design outer moldline PTCS for minimum recurring touch labor
- Avoid requirement for minimum cold-soak times to enhance contingency flexibility

Figure 1.0-20 Passive Thermal Control Design Goals

*Reaction Control*

- Avoid interconnects with OMS to enhance reliability
  - Minimizes isolation valve count
- Consider use of +X RCS for OMS function
  - Lowers vehicle complexity and operations costs versus performance
- Avoid dependency on helium blow-down pressurization to avoid helium leak contingencies
- Minimize need for active engine/propellant thermal conditioning to help minimize piece parts
- Provide vernier RCS capability for proximity operations
  - Helps to minimize plume impingement issues while keeping approach velocities low
- Leverage use of "low Z" off-axis RCS/VRCS to help minimize plume impingement issues during prox. op.s
- RCS sizing and associated $\Delta V$ for ascent governed by method of roll control and desired rates (which is an ascent performance tradeoff)
- Size $\Delta V$ capability for sum of on-orbit and entry requirements to ~100 fps

Figure 1.0-21 Reaction Control Subsystem Design Goals

*Structure*

- Load path design is coupled with aerodynamics, MPS, and propulsion design & layout
  - Strive for short and simple load paths
- Static and dynamic load paths for free-standing vehicle will drive structural design of propellant tanks, intertank(s), interstage(s), etc.
  - Propellant tank arrangement a trade between load path and vehicle stability & control requirements/capabilities
- Manufacturing designs chosen to minimize mechanical fasteners and manufacturing touch labor, while facilitating non-destructive test and certification
- Classical factors of safety 1.4 for "dynamic" structures and 1.2 for nondynamic
  - Design margins a trade between performance (inert mass penalty) and operability
- Design to avoid requirement for active load relief during ascent and entry
- Design to avoid pre-loaded structural elements, to simplify ground processing

Figure 1.0-22 Structure Design Goals

# 2.0 SSTO Design Groundrules

The top level design rules that were used in the TA-2 analysis of candidate SSTO concepts are shown in Figure 2.0-1. These groundrules were explicitly taken from the Access to Space Option 3 guidelines in order to provide the most consistent comparison between the SSTO concepts developed by the Option 3 team and those of the TA-2 team.

| | |
|---|---|
| Cargo Bay-- | *Diameter = 15 ft.; Length = 30 ft.* |
| Payload Capability-- | *25,000 lbm to 220 nm, 51.6 deg. orbit (uncrewed option)* |
| Crew Capability-- | *2 flight crew and 4 passengers for Space Station crew rotation (crewed option)* |
| Crossrange Capability-- | Not a design constraint |
| Flight Loads: | |
| Ascent-- | *3 Gs max. axial acceleration* |
| Entry-- | 2.5 Gs max. normal acceleration (winged only) |
| Abort-- | Mission completion with engine-out not a design constraint |
| Mission Duration-- | *7 days (launch through landing)* |
| On-Orbit DV Capability-- | *1,100 fps* |
| Dry Mass Contingency-- | *15 percent (applied to all subsystems)* |
| Launch Window-- | *5 minute minimum for Space Station rendezvous* |

Italics indicate an Access to Space Option 3 Team guideline that was used

Figure 2.0-1 SSTO Design Groundrules

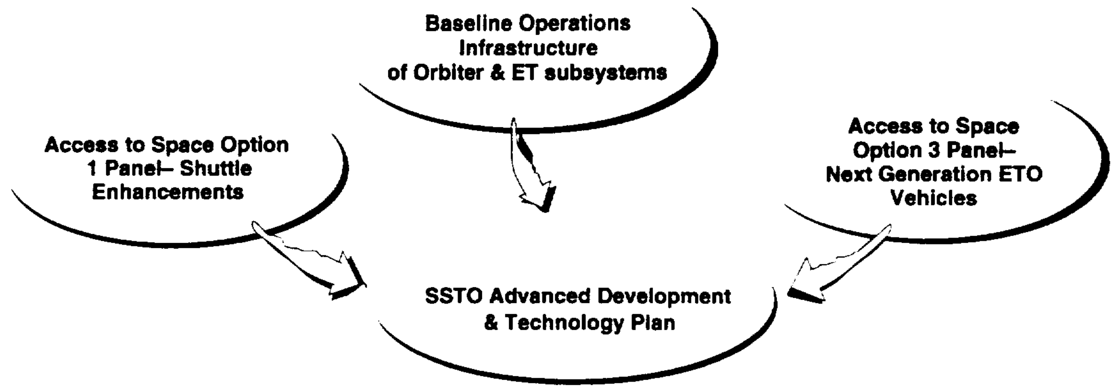
# 3.0 Operations Issues and Lessons Learned

During the definition and assessment of new launch systems, a major aspect of the concurrent engineering design process is the provision for operability. A thorough understanding of those factors that influence recurring launch services costs is also required from the very start of the vehicle design process, in order to produce a programmatically viable launch system concept. Operations costs become a dominant portion of the recurring costs of operating an SSTO fleet under the premises that the SSTO fleet is small and fully reusable. The Lockheed Space Operations Company (LSOC), which is responsible for performing all of the ground operations activities for NASA's only partially-reusable launch vehicle, the Space Shuttle, was tasked under TA-2 to leverage Shuttle operations lessons-learned in helping to guide the SSTO concept definition activities. The following sections document LSOC's SSTO effort. Additionally, Section 9 contains the findings of an SSTO turnaround assessment that was performed by reliability, maintainability, and supportability (RM&S) personnel at Lockheed's Skunk Works for TA-2, utilizing an aircraft-based RM&S approach.

## 3.1 Operations Issues — Lessons Learned

An important corequisite to SSTO technology development is the application of operations and program management lessons-learned from the Space Shuttle Program, as suggested in Figure 3.1-1. NASA has flown over fifty-five Shuttle missions and thus has accumulated a large amount of experience in operating a reusable fleet of launch vehicles and spacecraft. It is important that the comparable subsystems between SSTO and Shuttle be identified, and that the operations "costs" for these Shuttle subsystems be baselined as accurately as possible. This baselining process will allow the current problem areas to be identified and prioritized for new technology or methodology investment, and will provide insight into alternate design solutions.

An example of operations baselining is the 1993 Orbiter APU/Hydraulics Baseline Assessment, performed by LSOC for NASA KSC as part of the Electric Actuation Technology Bridging Program. Shuttle APU, hydraulics, and flight control subsystem launch processing was baselined. The baseline included overviewing flight hardware and processing tasks, identifying GSE, performing schedule analysis, identifying planned and unplanned maintenance tasks, and estimating manpower and costs. This baseline process should be expanded to cover the mission operations functions at JSC for comparable Shuttle subsystems. It should also be broadened to cover non-hardware related operations functions such as program management.




- **NASA has attempted only one partially reusable ETO vehicle fleet**
- **Apply operations and programmatic lessons-learned from Shuttle to SSTO**
- **SSTO will utilize subsystems comparable to Orbiters and ETs -- baseline operations infrastructure for each subsystem**
- **Example -- 1993 Orbiter APU/Hydraulics Baseline Assessment**
  **(Hardware & processing overview, GSE and shop aids, planned/unplanned maintenance analysis, schedule analysis, manpower estimates)**

**Figure 3.1-1 Operations Issues– Lessons Learned**

To summarize, NASA must clearly understand and prioritize the operations and management problems encountered with the current reusable Shuttle fleet before investing in a new reusable launch system.

## 3.2 Operations Issues — Requirements Flowdown

Early in the SSTO design, a substantial amount of effort should be placed on defining operations requirements. The capability to rapidly turnaround and operate a reusable launch vehicle will be driven at least as much by program requirements as technology. The ground and mission operations philosophy is driven from the top-down. It is imperative that streamlined Level I and Level II requirements be dictated in the program, since these multiply dramatically at the ultimate "operator level" where operations and maintenance instructions (OMIs) requirements are levied, as illustrated in Figure 3.2-1.

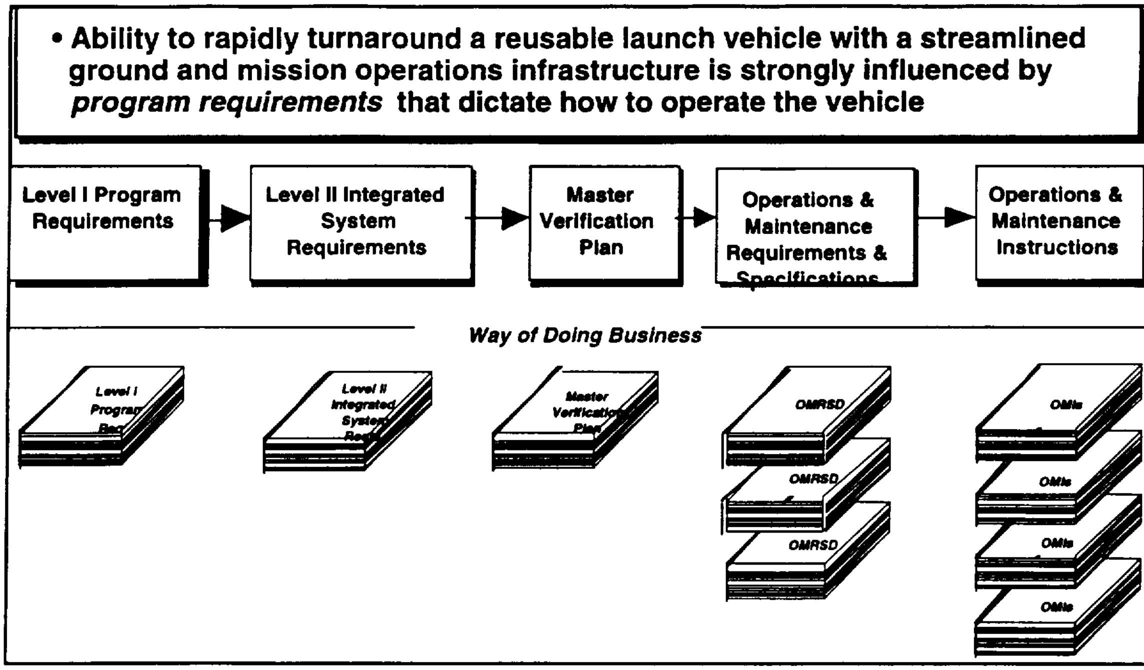



**Figure 3.2-1 Operations Issues– Requirements Flowdown**

Top-level oversight and enforcement of the end-to-end requirements flowdown process must be maintained to insure that an unwieldy number of end-user requirements are not levied on operators who are attempting to turnaround and operate an SSTO faster and cheaper than today's Shuttle fleet.

## 3.3 SSTO Operability Pros/Cons of Tripropellant Versus Bipropellant

A top-level trade which needed to be resolved early in the SSTO vehicle design was propellant selection. This was important since propellants are the number one driver of vehicle volume, tankage and propellant feedline layout and engine selection. The ATSS TA-2 team members qualitatively assessed the operational merits and weaknesses of utilizing two versus three propellants for main propulsion on the SSTO vehicle. The bipropellant was assumed to be LOX/hydrogen, and the tripropellant was assumed to be LOX, hydrogen and a hydrocarbon, such as RP-1.

Pros and cons of tripropellant versus bipropellant selection were made assuming that recurring operating cost was the major vehicle design driver. Propellant type was the variable in this qualitative parametric analysis. Thus, it was assumed that vehicle configurations would all meet the same reference payload performance requirements, and that all vehicles in the same class (i.e., vertical lander) would require the same number of main engines.

The tripropellant was compared to the bipropellant vehicle concept with the following question in mind: "Does tripropellant help reduce recurring operations costs relative to bipropellant?" If tripropellant reduced costs, the reasoning was denoted in the "pro" column, and if tripropellant increased costs relative to bipropellant, the reasoning was denoted in the "con" column. Three charts of these pros and cons (or neutral issues) were drafted by the ATSS TA-2 team in a concurrent engineering meeting, shown in Figure 3.3-1.

| Pros | Cons | Neutral |
|---|---|---|
| • Less stand-off structure (for non-integral propellant tank designs) allows less structural maintenance<br>• Less TPS allows less body TPS refurb. & repair<br>• Use of noncryogenic third propellant facilitates prop. loading timeline vs. cryogenic third propellant<br>• Smaller vehicle will require less primary structure and associated TPS materials | • ~50% increase in main prop. feed & press. parts count, increasing processing test & checkout by 50%<br>• Increased parts count increases likelihood of unscheduled maintenance<br>• Increased unscheduled maintenance increases logistics burden (spares)<br>• "New" nature of triprop. propulsion increases likelihood of infant mortality failures in propulsion components<br>• Increased complexity increases processing learning curve<br>• Decreased vehicle size and increased parts count increases maintenance accessibility difficulty (if not considered in the design)<br>• Increased propulsion complexity will require more ground checkout and launch software<br>• Increased ground checkout and launch software will increase sustaining software maintenance<br>• Increased hydrogen tank sizing for dual-fuel Mode 1 is traded against not having capability to fully verify engine health on-pad if single-fuel in Mode 1<br>• No capability to verify 90% engine health onpad in both modes prior to liftoff<br>• Higher flight performance reserve for 3 propellants<br>• Use of cryogenic third prop. complicates prop. loading timeline<br>• Fuel mode optimization complicates nominal/abort flight design | •Smaller vehicle but not a driver for SSTO class<br>– Same number of engines to process as biprop. for new "rubber engine"<br>– Processing not affected by vehicle size (up to a point) |

Figure 3.3-1 Bipropellant vs. Tripropellant Pros/Cons

| Pros | Cons | Neutral |
|---|---|---|
| | • Increased propulsion complexity will require more flight ops software<br>• Third propellant is an additional commodity to buy, transport, store and load at launch pad<br>• Environmental hazard mitigation for hydrocarbons will require spill pond, water sample wells, and possibly a waste water treatment facility<br>• Increased propulsion complexity will require more extensive engine qualification and certification program<br>• Additional hazardous gas detection hardware onboard<br>• Additional propellant tankage with associated tank insulation | |

**Figure 3.3-1 Bipropellant vs. Tripropellant Pros/Cons (Concluded)**

The major benefit of utilizing tripropellant on an SSTO vehicle is that it reduces the propellant tank volume and attendant tank weight, thermally protected surface area and structural attachment weight (for non-integral tank designs) compared to bipropellant. However, size is not a primary driver of operations processing costs for vehicles roughly 30 percent different in volume. Subsystem parts count and complexity instead are first-order operations drivers.

The major benefit of utilizing bipropellant is that it reduces propulsion subsystem complexity/parts count. Bipropellant main propulsion and attendant systems will have only half the parts count of tripropellant. This will result in approximately half the test and checkout procedures compared to tripropellant. Bipropellant systems have been in operation on various launch systems for decades, whereas liquid tripropulsion is a new endeavor. Therefore, tripropellant will likely experience more infant mortality failures early in the operations phase of the SSTO program life cycle.

Bipropellant fuel selection simplifies not only the vehicle propulsion system hardware compared to tripropellant, but also the ground and flight operations software. Bipropellant will likely require less ground checkout and launch software and attendant software maintenance during the operations phase of the program life cycle. Bipropellant will also simplify ascent flight design/planning due to fewer engine-out permutations existing for two versus three propellants. Bipropellant will also be safer, since engine health can be verified at approximately 90% throttle before liftoff. Tripropellant cannot feasibly verify 90% engine health in both fuel modes before liftoff.

Bipropellant fuel selection simplifies not only the vehicle propulsion system hardware compared to tripropellant, but also the ground hardware. A third propellant would require more equipment for propellant transport, storage and transfer. Further, the hydrocarbon third propellant would require environmental hazard mitigation. In summary, tripropellant requires the same support as bipropellant with added support required for the hydrocarbon third propellant.

It is the recommendation of the ATSS TA-2 team that bipropellant propulsion be chosen over tripropellant for use on SSTO, given the previous assumptions. Tripropellant is an unnecessarily more complex solution to the earth-to-orbit transportation problem than bipropellant, and will likely result in a more expensive, harder-to-maintain vehicle than bipropellant. A tripropellant vehicle will likely cost more to operate than the bipropellant due to added complexity, and will likely cost more to develop due to the new tripropellant engine type.

# 4.0 Vertical-Takeoff/Landing Versus Vertical-Takeoff/Horizontal-Landing

The ATSS TA-2 team (LMSC, LSOC, Aerojet, ECON) met in a concurrent engineering session to define the qualitative pros and cons of SSTO generic vertical takeoff and landing (VTOL) versus vertical takeoff/ horizontal landing (VTHL) configurations. The pros and cons were comprised of the following areas: engine design and development, flight control risk, structural efficiency, landing system design, landing opportunities, vehicle processing and operations, and miscellaneous. Four worksheets of pros and cons were developed for the VTOL concept, and four worksheets were developed for the VTHL concept.

## 4.1 VTOL/VTHL Pros/Cons Results Summary

A summary of the pros and cons of VTOL versus VTHL SSTO concepts are shown in Figure 4.1-1. The major benefits of the VTOL vertical lander compared to VTHL are that it provides for more control during terminal descent due to the engine power-on state, it allows for a simpler load path due to the symmetrical shape, and the vehicle allows for a single launch processing orientation (vertical). The major weaknesses of this landing concept are that the MPS system is more complex (engine power-on during ascent and descent), landing and post-mission propellant deservicing is riskier, and the vehicle requires vertical (versus horizontal) access throughout the entire prelaunch, launch and landing processing flow.

The major benefits of the VTHL horizontal lander compared to VTOL are that no engine conditioning and restart is required for descent, the flight mechanics are well-understood for vertical powered ascent and unpowered descent, and airstrips of 10000 - 15000 ft length exist worldwide for a horizontal lander. The weaknesses of the horizontal lander concept are that it is less structurally and volumetrically efficient due to the asymmetrical shape, and it requires both horizontal and vertical access during the prelaunch, launch and landing processing flow.

| | VTOL | VTHL |
|---|---|---|
| Pros | • More physical options for landing opportunities<br>• Simpler load path and primary structure design<br>• Larger static stability margin possible | • No requirement for main engine conditioning & restart post-MECO<br>• Less hazardous post-flight deservicing<br>• Conventional flight mechanics & dynamics during all phases |
| Cons | • More complicated main propulsion & feed subsystems<br>• Higher risk entry/TAEM flight mechanics & dynamics<br>• Higher risk post-flight deservicing<br>• Vertical ground processing required | • Less volumetrically efficient outer moldline<br>• Higher structural dry mass |

**Figure 4.1-1 VTOL/VTHL Pros/Cons Results Summary**

## 4.2 Vertical Take-off/Vertical Landing Pros and Cons

The engine design and development pros and cons for the VTOL configuration are shown in Figure 4.2-1. They include potential for base-entry/descent during engine firing and ease of engine arrangement on an axisymmetrical boattail. Engine weaknesses include the requirement to perform on-orbit propellant conditioning and restart for descent, deep throttling and attitude control requirements during terminal descent, and increased transonic base/engine drag.

| Pros | Cons |
|---|---|
| • Use of altitude compensating nozzle allows reentry on engine<br>- Heat loads on engine nozzle higher when engine firing than during reentry<br>• Plug nozzles can utilize engine for entry heat shield<br>• Less yaw moment from engine-out<br>• Easy to incorporate modular engine concept<br>- Thrust cells<br>• Easy to incorporate TVC by throttling engine sectors<br>• Modular engine reduces development | • Requires deep (10:1) throttling engines<br>• Engine design and development more complex<br>- Deep throttling required<br>- 2 position nozzle required<br>- Restart required<br>- 2 cycles/mission = 1/2 life<br>• Requires restart conditioning<br>• Base area large. Limits propulsion options<br>• Requires engine restart for safe landing<br>• Must have a roll control system (if use differential throttling)<br>• No satisfactory existing plug nozzle engine<br>- New engine development<br>• Engine development test facilities more complex<br>• Require on-board purge (engine) for restart (especially for an abort return) |

Figure 4.2-1 VTOL Engine Design and Development Pros and Cons

The flight control pros and cons for the VTOL configuration are shown in Figure 4.2-2. The benefit for VTOL includes reduced yaw moment from engine-out. All cons were related to concerns over powered descent, such as propellant slosh and increased gimbal rate requirements during terminal descent, powered pitcharound maneuvering, and plume blow-back. A larger number of failure modes were anticipated for a vertical lander during terminal descent/landing.

| Pros | Cons |
|---|---|
| • Reduced yaw moment from engine out | • Vehicle flight dynamic during landing<br>• Slosh damping required for powered pitcharound in addition to ascent/trade of which "MV" term sizes the baffles<br>• Flight dynamics of powered pitcharound for landing is complex and risky, including plume blow-back issues<br>• Unfamiliar control requirements<br>• High gimbaling rate requirement<br>• Larger gimbal angle requirement<br>• Requires large body flaps for aerodynamic control<br>• Failure modes associated with landing higher than horizontal landing<br>• Array of intact abort options is more complicated to design autonomously than their benefit<br>• Vertical landing vehicles have inherent higher accident rates than HL<br>• Center of gravity placement versus Cp difficult to achieve |

**Figure 4.2-2 VTOL Flight Control Risk Pros and Cons**

The landing opportunity pros and cons for the VTOL configuration are shown in Figure 4.2-3. The benefits for the VTOL concept include the need for only a small prepared landing area due to the VTOL's accurate vertical landing capability (powered flight), the ability to land in many potential sites due to the small area, and the ability to dissipate unnecessary fuel in hover mode prior to landing. The major landing opportunity weakness was the assumed lower L/D compared to VTHL and lower resulting crossrange.

| Pros | Cons |
|---|---|
| • Propellant dissipation maneuver for abort (atmospheric) is hover mode<br>• Return "anywhere"<br>- More options<br>• Minimum take-off landing facility<br>• Have more abort site opportunities<br>• Small landing area<br>• Does not require a runway for landing<br>• Wider choice of possible landing places<br>• More potential launch sites | • Landing dispersions<br>• Low vehicle L/D translates to low cross range capability |

Figure 4.2-3 VTOL Landing Opportunities Pros and Cons

The landing system pros and cons for the VTOL configuration are shown in Figure 4.2-4. The VTOL configurations have the weakness of requiring added structure for handling horizontal (sideways drift) as well as vertical loads, and the need to support the vehicle's higher center of gravity.

| Pros | Cons |
|---|---|
| | • Landing gear requires extra beef-up for drift protection as well as vertical loads<br>• Requires robust landing gear |

Figure 4.2-4 VTOL Landing System Design Pros and Cons

The payload integration pros and cons for the VTOL configuration are shown in Figure 4.2-5. The benefits include the ability to integrate and launch payloads in a single orientation, and the ability to have a larger payload center of gravity envelope, due to powered descent.

| Pros | Cons |
|---|---|
| • Larger C.G. envelope<br>• Payload can be on top of vehicle<br>- Easily accommodate variable length payload<br>- Vehicle less sensitive to payload c.g. | |

Figure 4.2-5 VTOL Payload Integration Pros and Cons

The miscellaneous pros and cons for the VTOL configuration are shown in Figure 4.2-6. Despite the concerns of landing acoustics (particularly on an flat surface with no water deluge for attenuation), the VTOL concept could be easily evolved/modified for use as a vertical lander for lunar or planetary exploration.

| Pros | Cons |
|---|---|
| • Provides for free vent of Hydrogen/Oxygen<br>• All vertical payload operations and integration<br>• All vertical vehicle integration and operations<br>• Can probably evolve to a "Lunar" lander | • Landing acoustics |

Figure 4.2-6 VTOL Miscellaneous Pros and Cons

The structural efficiency pros and cons for the VTOL configuration are shown in Figure 4.2-7. The benefits of VTOL include high volumetric efficiency and simple load path due to the axisymmetrical layout of a generic configuration and the vertical launch and landing loads. The body shape is simpler which allows easier tooling and manufacturing (i.e., tanks with circular cross sections). Lighter landing gear are possible due to the lower landing speeds and use of pads versus wheels with brakes. The structural efficiency cons of VTOL focus on the requirement for storing additional main engine and reaction control system (RCS) propellant for deorbit and landing use. Added propellant causes weight increases in tankage, support structure, and thermal insulation. Landing propellant and associated flight performance reserves (FPR) reduce the payload capability at a 1:1 ratio.

| Pros | Cons |
|---|---|
| • High mass fraction structure<br>• Should have best vehicle mass fraction<br>  - Smaller, lighter, cheaper vehicle<br>• Body shape inherently stiff<br>• Circular cross section tanks possible<br>• Squat shape reduces vehicle height and loads<br>• Shape should lend itself to "unitized"construction of major structural elements<br>• Body shape has good volumetric efficiency<br>• Simple structure<br>• Simple load path<br>• Lighter landing gear<br>• Shape allows in-line propellant tank configuration<br>• Less propellant tanks due to geometry<br>• Less high temperature TPS area<br>• Minimizes thermal protection surface<br>• Allows for non-lifting body design which increases accessibility by not being as volumetrically limited<br>• Simple aerodynamics (easy to predict)<br>• Simple body shape for tooling and manufacturing | • Propellant for landing is payload<br>• Must add either an extra subsystem or size extra propellant tanks, structure for $\Delta V$<br>• On-orbit storage of LO2/LH2 for 3-14 days (boiloff and propellant management)<br>• Larger mission velocity requirement<br>  - More drag on ascent<br>  - Landing maneuver<br>  - Questions on how much hover capability required<br>• Probably heavier than VTHL<br>• Higher on-orbit and deorbit mass<br>• Size propellant tanks to carry landing propellant which is payload hit (includes tanks, insulation, structures)<br>• Fuel bias extra hit if can't handle fuel depletion cut<br>• Base area larger, requiring more engineering to minimize base drag<br>• RCS propellant required for landing maneuvers (weight penalty)<br>• Perception problem, does not land on a runway<br>• Safe abort (recovery of vehicle) during landing with propulsion system failure is large penalty<br>• FPR sizing rqmts. for landing; hit to payload<br>• Limited pilot visibility during final descent<br>• Requires large propellant mass, larger tanks |

**Figure 4.2-7 VTOL Structural Efficiency Pros and Cons**

The vehicle processing and operations pros and cons for the VTOL configuration are shown in Figure 4.2-8. The VTOL concept is simplified by the single vehicle orientation during launch and landing operations and during payload integration. The weakness of VTOL is that this processing is done in the vertical. Vertical access is more restrictive and requires taller, therefore more expensive, access platforms and buildings. Vertical ground transportation would be a problem, particularly cost-effective transport from remote landing sites to the launch site(s). Post landing deservicing operations are more complex and hazardous due to the presence of cryogenic propellant residuals. Blast debris danger will also exist in the VTOL landing area.

| Pros | Cons |
|---|---|
| • Single orientation (vertical) for payload operations and integration | • Landing area blast debris<br>• Vertical cargo integration<br>• Vertical processing<br>• Vertical checkout required<br>• Requires vertical vehicle processing and payload integration<br>• Range safety issue of landing with propellant<br>• Post landing servicing of vehicle with propellant residuals<br>• IOP or MLP<br>• Ground transportation of vehicle<br>• Payload volume difficult to integrate within vehicle mold line |

Figure 4.2-8 VTOL Vehicle Processing and Operations Pros and Cons

## 4.3 Vertical Take-off/Horizontal Landing Pros and Cons

The engine pros and cons for the VTHL configuration are shown in Figure 4.3-1. They are derived from a single start requirement-- the engines only need to fire during ascent. Engines require no restart for landing, and the propulsion system is simplified. No added landing propellants need to be stored. The tanks can be vacuum-inerted on-orbit following main engine cutoff (MECO). Only moderate engine throttling is required since the engines are not used during landing when the vehicle is lightest. Also, the base/engine area drag is reduced due to the decreased base area. The engine con for VTHL is that the smaller boattail allows less engine exit area. Parametrically, the effect of this would be to require a higher engine chamber pressure to achieve the same thrust level.

| Pros | Cons |
| --- | --- |
| • Can use either bell or plug nozzle<br>• Could use existing engines<br>• Minimum base area expands propulsion configuration options<br>• Body shape incurs less base drag for larger range of propulsion options<br>• Moderate (3:1) throttling requirement<br>• Engines can be stowed for return<br>• Capability to purge $LO_2/LH_2$ system on-orbit to vacuum (no post flight propellant hazards)<br>• More choices on TVC<br>- Differential throttling<br>- Gimbaling engines<br>• No restart requirement<br>• Single engine burn<br>• No requirement for main engine restart post-MECO<br>- With associated MPS simplification and payload savings<br>• Engine has "fewer" operating requirements- throttling, restart control, etc.<br>• Engines not required for landing | • Smaller boattail requires a higher engine chamber pressure for a given area ratio engine |

**Figure 4.3-1 VTHL Engine Design and Development Pros and Cons**

The major flight control pros and cons for the VTHL configuration are shown in Figure 4.3-2. They are the ascent, entry and landing guidance and control modes are well understood from Shuttle. Also, the nominal guidance is less complex, since descent and landing is unpowered. The flight control cons of VTHL are that the vehicle is more sensitive to center of gravity during descent, and more yaw moment is potentially created from an outboard engine-out.

| Pros | Cons |
|---|---|
| • Simpler flight software (fewer guidance modes)<br>• Perception, lands on a runway<br>• More robust landing method<br>• Able to handle higher crosswinds during landing (terminal descent); body should have weathercock stability<br>• Entry and terminal area energy management maneuvers are less dynamic and more predictable (no PPA, no slosh issues during entry)<br>• Well understood landing process (Shuttle) | • More yaw moment from engine-out<br>• Vehicle sensitive to cg<br>• Use of bell engines makes vehicle cg more critical |

**Figure 4.3-2 VTHL Flight Control Risk Pros and Cons**

The structural pros and cons for the VTHL configuration are shown in Figure 4.3-3. They are no added propellants need to be carried, stored and conditioned on-orbit for descent and landing, and an inert weight penalty for wings can be avoided by using a lifting body shape. The structural efficiency weaknesses of VTHL are that the body shape is less volumetrically efficient, the lifting body shape causes more propellant tanks to be used, and to be shaped more complex. This increases the tooling and manufacturing costs of the tankage. Finally, the structure must absorb vertical ascent loads and horizontal landing loads.

| Pros | Cons |
|---|---|
| • Easier to fly a lifting trajectory<br>• Moderate gimbal angle requirement<br>• Moderate gimbal rate requirement<br>• Reasonable volumetric efficiency possible<br>• Probably lightest option<br>• Lower inert weight than VTOL<br>• No return propellant requirements<br>• $G_{loss}$ can be less if fly lifting ascent<br>• Lower total mission velocity required<br>• Possible to have simple load path<br>• Structurally stiff<br>• No inert weight penalty for wings<br>• Will not require ablative or actively cooled heat load<br>• Lifting reentry reduces peak heat flux temperature | • Body shape less volumetrically efficient<br>• Moderate gimbal rate requirement<br>• Lifting body not an efficient propellant tank<br>• Less efficient volume<br>• More propellant tanks due to geometry<br>• Possibly complex body shapes, increasing complexity of tooling, fabrication, production<br>• Requires high angle-of-attack re-entry flight angle<br>• Greater demand for TPS materials<br>• More high temperature TPS area<br>• Larger cross range is at expense of worse vehicle mass fraction<br>• Load path vertical for ascent and horizontal for re-entry and landing<br>• More restrictive payload bay, larger payload bay increases vehicle size and weight<br>• Wings or body lift required for landing<br>• Tankage not necessarily of circular cross section<br>• Hard to achieve high mass fraction structure |

**Figure 4.3-3 VTHL Structural Efficiency Pros and Cons**

The landing opportunities pros and cons for the VTHL configuration are shown in Figure 4.3-4. They include a higher L/D which allows a larger cross range. This allows more opportunity to land at existing runways. Horizontal landers can take advantage of this existing landing site infrastructure instead of having custom landing sites prepared. The major landing opportunity weakness is that the landing speed will be greater for VTHL, thus requiring a runway in the first place.

| Pros | Cons |
|---|---|
| • Existing landing infrastructure<br>• Can have large cross range<br>• Improved landing opportunities | • Terminal landing speed higher (H-dot, Vx) requiring heavier landing gear(?)<br>• Fewer return site options<br>• Limited places to land or abort to requires a runway<br>• Requires large landing facility<br>• Requires prepared landing surfaces |

**Figure 4.3-4 VTHL Landing Opportunities Pros and Cons**

The landing gear design pros and cons for the VTHL configuration are shown in Figure 4.3-5. The weakness is that the larger landing speeds and horizontal landing profile require stronger gear with wheels and brakes, thereby requiring more structure in the landing gear. However, less landing and deceleration subsystem "loss of vehicle" failure modes are envisioned during terminal landing for VTHL than VTOL.

| Pros | Cons |
|---|---|
| • Consequences of landing/ deceleration subsystem during terminal area energy management maneuvers/ landing are more survivable (crew/payload) than VTOL | • High landing speeds require extensive landing gear tire development |

**Figure 4.3-5 VTHL Landing System Design Pros and Cons**

The vehicle processing and operations pros and cons for the VTHL configuration are shown in Figure 4.3-6. The benefits include horizontal vehicle checkout and ground transport, and horizontal or vertical payload integration flexibility. Horizontal processing facilities provide for faster, easier access, and are less costly to build than vertical facilities. Existing shuttle orbiter processing facilities (OPFs) could possibly be used for VTHL horizontal processing. Another benefit is that VTHL requires no post landing hazardous cryogenic propellant deservicing. The central VTHL processing and operations weakness is that the vehicle must be rotated to vertical for launch following horizontal prelaunch processing.

| Pros | Cons |
|---|---|
| • Enables horizontal processing/ check-out pre-and post-mission with better accessibility<br>• Large experience base<br>• Traditional experience<br>• Easy transport to processing facility<br>• Can be towed from place to place on its landing gear by aircraft tow cart<br>• Horizontal P/L integration, engine;<br>• Accessory access<br>• Cockpit<br>• Can use Shuttle facilities<br>• Can be horizontally processed<br>• Payload volume<br>• Easy to integrate<br>• Rollover ground transport<br>• All horizontal payload integration (if baselined)<br>• Option of vertical or horizontal checkout | • Must have residual propellant disposal prior to landing<br>• Requires horizontal to vertical repositioning<br>• GSE<br>• Rotation to vertical<br>• Mixed horizontal and vertical vehicle operations |

Figure 4.3-6 VTHL Vehicle Processing and Operations Pros and Cons

# 5.0 SSTO Design Results

This section presents the results of a preliminary design trade study that assessed the three major types of SSTO vehicle configurations using a common set of groundrules and program requirements. Both bipropellant and tripropellant main propulsion concepts were also assessed. The goal was to assess, within a vehicle concept and engine type, how the choice of propellant combination affected the vehicle dry mass. A secondary goal was to determine if the propellant choice effects were also dependent upon the vehicle concepts used.

The three vehicle concepts that were evaluated were a vertical take-off/vertical-landing (VTOL) side-entry cone, a vertical-takeoff/horizontal-landing (VTHL) winged body, and a VTHL lifting body, as shown in Figures 5.0-1, 5.0-2, and 5.0-3 respectively.

The propellant combinations selected were oxygen/hydrogen, oxygen/hydrogen/RP-1, and oxygen/hydrogen/propane. The combination of oxygen/hydrogen was used because of the high achievable specific impulse, and availability of advanced engine concepts using this propellant combination. The combination of oxygen/hydrogen/RP-1 was chosen because it had become a popular tripropellant combination within the main propulsion community. The combination of oxygen/hydrogen/propane was chosen because its density and specific impulses fall between the other two propellant combinations. Aerojet provided engine data on three different engines with each engine using the propellant combinations.

During the course of the trade study, additional engines were added, as discussed in Section 5.3. The definition of these engines came from different propulsion vendors, thereby implying different design assumptions being used. Each main engine option used only one propellant combination. The engines were added to the trade study engine matrix because they were of interest to the NASA RLV program and they provided additional data on possible engine concepts.

Since the SSTO vehicle dry mass is expected to correlate well with the expected vehicle development and production costs, it was used as the primary figure of merit in this trade study.

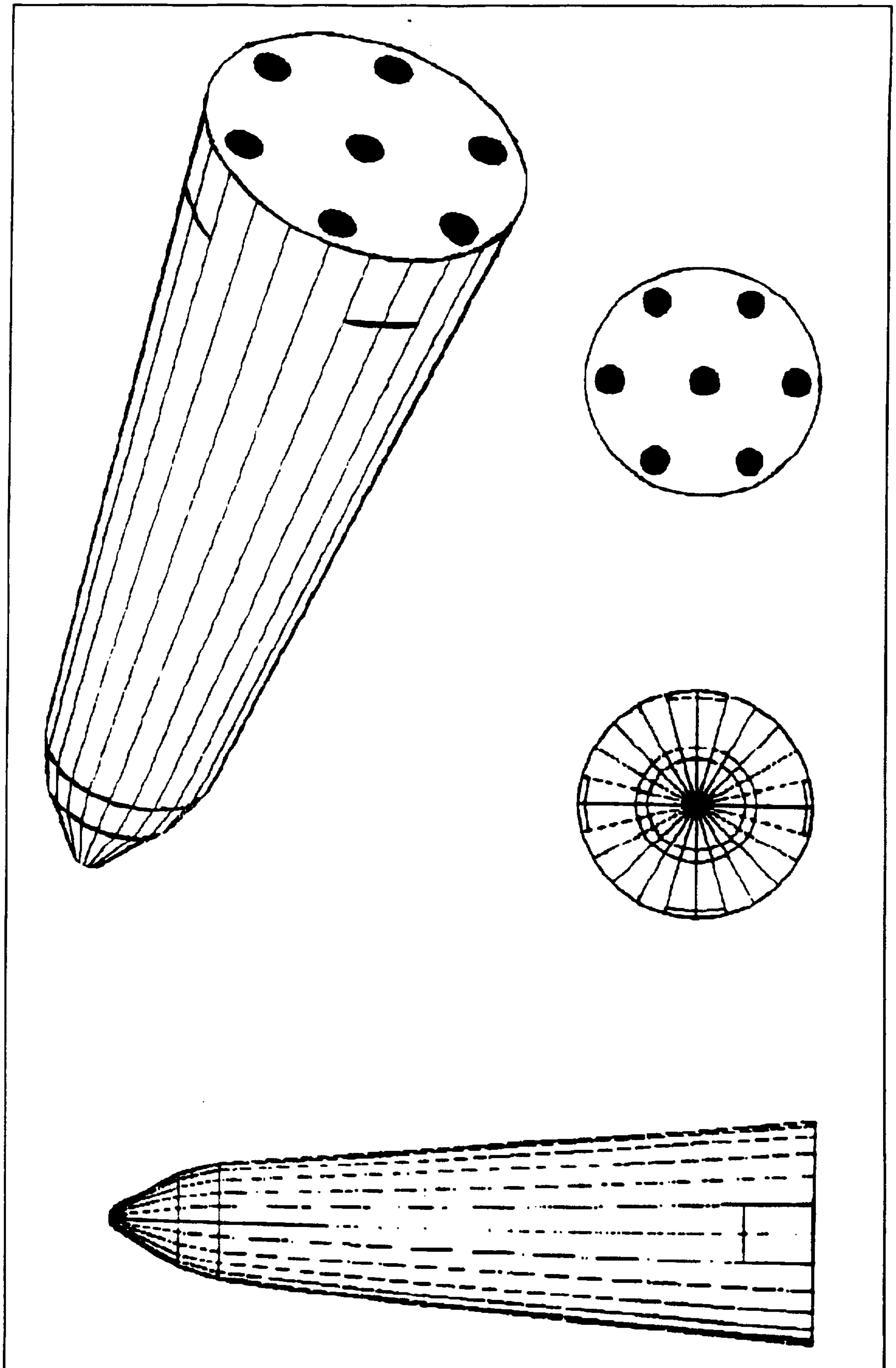


Figure 5.0-1 Side Entry Conical VTOL Launch Vehicle Configuration

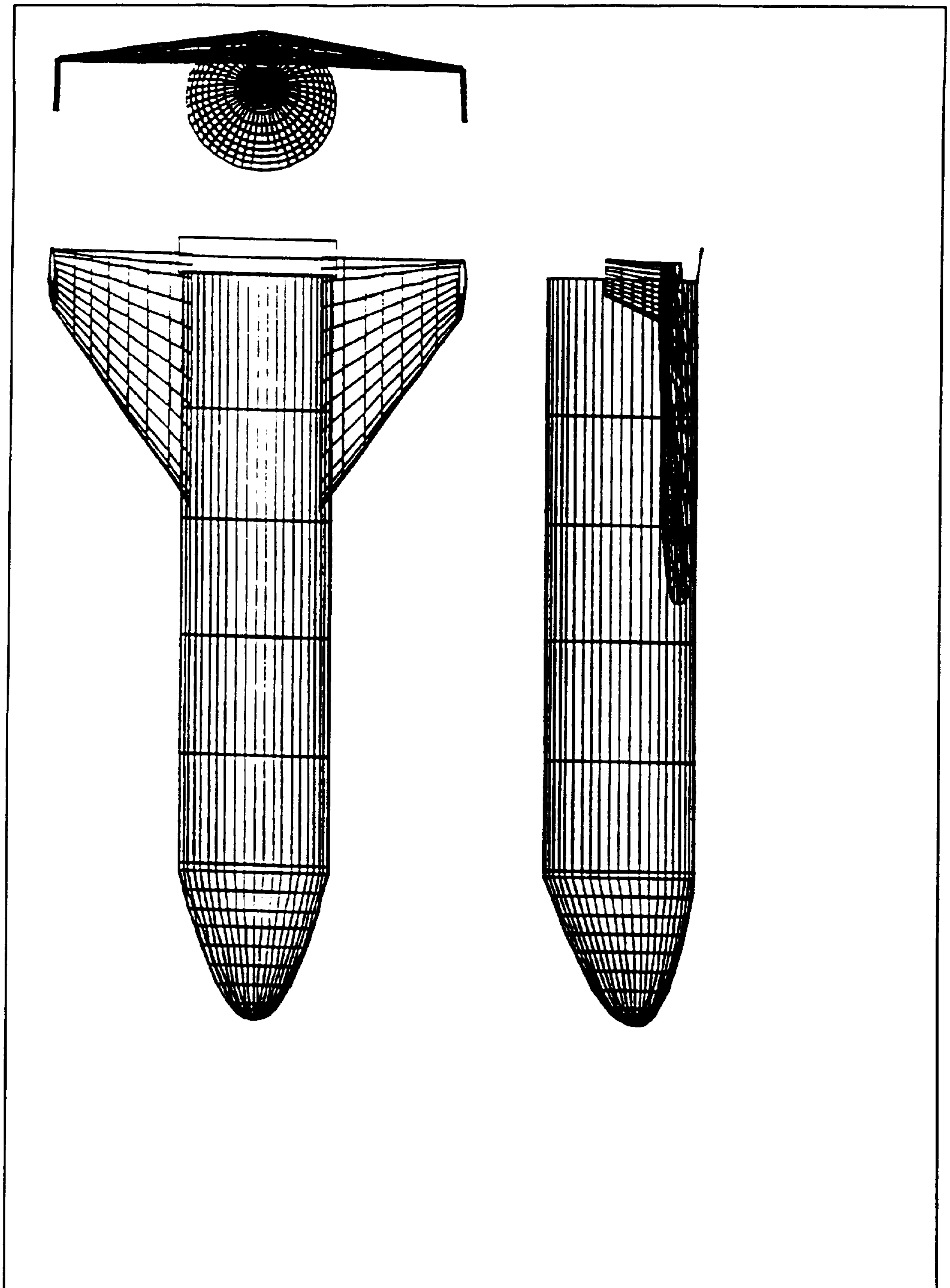


Figure 5.0-2 Winged Body VTHL Launch Vehicle Configuration

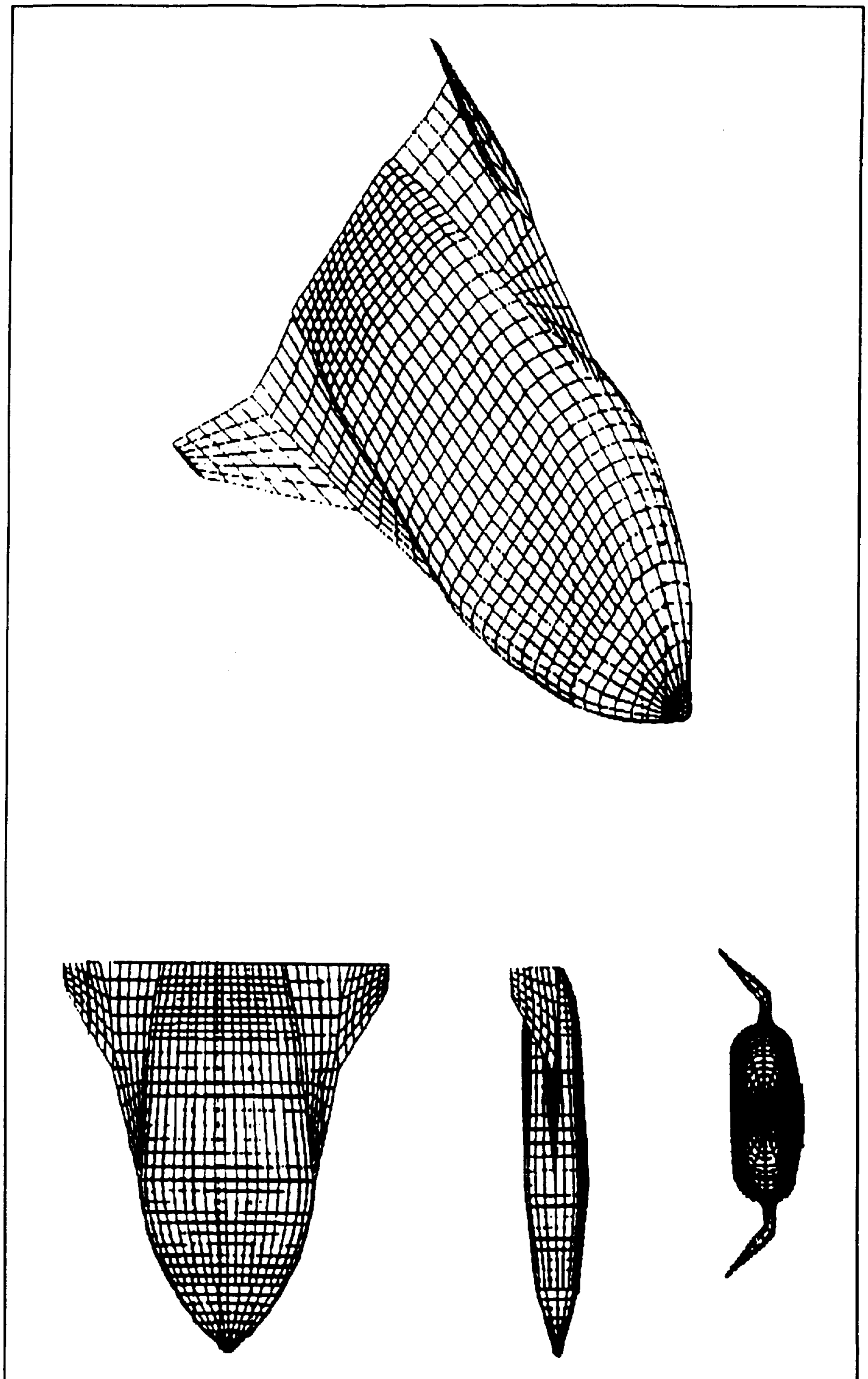


Figure 5.0-3 Lifting Body VTHL Launch Vehicle Configuration

## 5.1 Major Design Considerations

The major launch vehicle configuration concept design decisions can be split into the outer mold line decisions, the major structural element layout decisions, the propellant combination used, and the type of main engines used.

The vehicle concept is fundamentally defined by the outer mold line decisions. The outer moldline provides answers to such questions as: is the vehicle reusable?; how is reentry handled?; how are launch and landing handled?; what approach is used to handle the crossrange requirements?; where is the payload stored?; and what does the vehicle body look like? The decisions made at this point in the design process define the vehicle aerodynamic and reentry environments. The vehicle concepts used in this study are discussed in Sections 5.5, 5.6, and 5.7.

The next set of decisions involve the location of the major vehicle elements. These decisions define the vehicle load paths, the vehicle cg location during all flight regimes, and the vehicle volumetric efficiency. The cg location in relation to the vehicle cp location affects the vehicle's controllability. The vehicle's volumetric efficiency is a measure of how much of the vehicle's internal volume is taken up by propellant tankage. The vehicle's volumetric efficiency has a major impact on its mass fraction and therefore on it's size, mass, and volume.

The use of a bipropellant combination vs. a tripropellant combination is the first decision made on the selection of a propellant combination used on the vehicle concept. The next decision made is the selection of the propellants used. These decisions affect the operability of the vehicle concepts. The propellant densities define the tankage volume required to hold the propellant used for ascent and therefore the mass of these propellant tanks. The propellant tank masses in turn affects the vehicle mass fraction and therefore the size, mass, and volume of the resulting vehicle concept. The propellant combinations used in this analysis are discussed in Section 5.3.

The next decision is the choice of engines used on the vehicle concept. The major engine parameters are the engine thrust-to-weight ratio (which defines the engine mass) and the engine specific impulse (which defines the amount of propellant required). Both of these parameters affect the vehicle mass fraction and therefore the size, mass, and volume of the resulting vehicle concept. The engine length and mass distribution affect the vehicle cg location. The engine diameters and gimbal requirements define the minimum engine spacing distances. The engine choice also defines a point vs. distributed engine load thrust structure design. The engines used in this analysis are discussed in Section 5.3.

### 5.2 Vehicle Sizing Process

Vehicle sizing is an iterative process, as shown in Figure 5.2-1. The first step is entering the mission definition information into the sizing tool. This information includes the payload, the target orbit inclination, perigee and apogee, and the ascent trajectory acceleration constraints. The sizing tool goes through an iterative process to calculate the vehicle size and mass properties for the required mission velocity. Lockheed used a proprietary version (modified for advanced space transportation system assessments) of the NASA-standard Simulation and Optimization of Rocket Trajectories (SORT) program, which is a three-degrees-of-freedom trajectory optimization and simulation tool, to calculate the actual vehicle payload when flying an optimal nominal ascent trajectory.

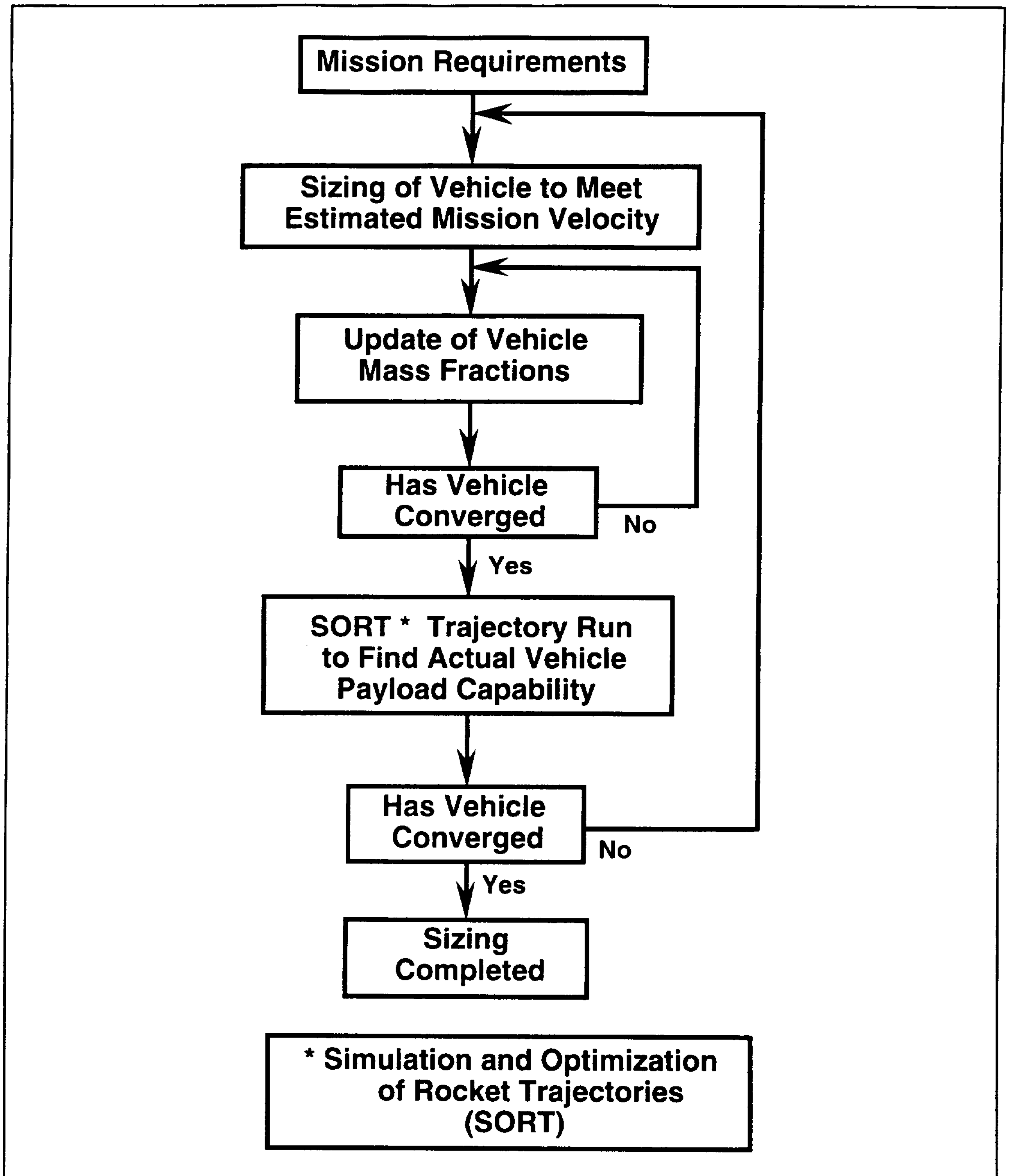



**Figure 5.2-1 Vehicle Sizing Process**

The preliminary value of the mission payload requirement used to size the SSTO concept is compared to the payload calculated by the SORT program. If these two payload values are within acceptable limits, the vehicle sizing process has converged and there is a solution. If the two payload values are not within acceptable limits, a new mission velocity requirement is calculated and the sizing tool goes through another iteration. This process is continued until a converged solution is reached.

For more detail regarding the sizing tool and its use, see the sizing tools User's Guide contained in Section 8 of this volume.

## 5.3 Technology Assumptions and Sizing Groundrules

The matrix of the launch vehicle configurations, engines, and propellant combinations used in this study are shown in Table 5.3-1.

**Table 5.3-1 SSTO Configuration Assessment Main Propulsion Matrix**

| Engine | Source | Propellant | Configuration |
|---|---|---|---|
| Evolved SSME | Option 3 | 1 | 1,2,3 |
| RD-701 | Option 3 | 3 | 1,2,3 |
| RD-704 | Pratt | 3 | 1,3 |
| Full Flow Staged Combustion | Rocketdyne | 1 | 1,3 |
| Dual Mixture Ratio (7/1 & 10/1) | Rocketdyne | 1 | 1,3 |
| Expander Cycle | Rocketdyne | 1 | 1,3 |
| Dual Expansion | Aerojet | 1,2,3 | 1,3 |
| Dual Throat | Aerojet | 1,2,3 | 1,3 |
| Plug | Aerojet | 1,2,3 | 1,3 |

| Propellant Key | Configuration Key |
|---|---|
| 1 O2/H2 | 1 Side Entry Cone VTOL |
| 2 O2/H2/Propane | 2 Wing/Body VTHL |
| 3 O2/H2/RP-1 | 3 Lifting Body VTHL |

The performance characteristics of these engines are shown in Table 5.3-2.

**Table 5.3-2 SSTO Engine Performance Characteristics**

| | Expander Cycle | Dual Expansion | | |
|---|---|---|---|---|
| Mode 1: | | | | |
| Oxidizer | Oxygen | Oxygen | Oxygen | Oxygen |
| Fuel 1 | NA | NA | Propane | RP-1 |
| Fuel 2 | Hydrogen | Hydrogen | Hydrogen | Hydrogen |
| MR Oxidizer (%) | 85.71 | 87.50 | 79.50 | 77.19 |
| MR Fuel 1 (%) | NA | NA | 17.32 | 19.63 |
| MR Fuel 2 (%) | 14.29 | 12.50 | 3.18 | 3.18 |
| Isl1 (sec) | 367.50 | 366.00 | 329.00 | 329.00 |
| Iv1 (sec) | 444.80 | 448.00 | 374.00 | 373.00 |
| Fsl/We (lbf/lbm) | 84.73 | 80.08 | 79.79 | 81.08 |
| Mode 2: | | | | |
| Oxidizer | Oxygen | Oxygen | Oxygen | Oxygen |
| Fuel 1 | NA | NA | NA | NA |
| Fuel 2 | Hydrogen | Hydrogen | Hydrogen | Hydrogen |
| MR Oxidizer (%) | 85.71 | 87.50 | 87.45 | 87.45 |
| MR Fuel 1 (%) | NA | NA | NA | NA |
| MR Fuel 2 (%) | 14.29 | 12.50 | 12.55 | 12.55 |
| Isl2 (sec) | NA | NA | NA | NA |
| Iv2 (sec) | 444.80 | 468.00 | 462.00 | 462.00 |

**Table 5.3-2 SSTO Engine Performance Characteristics (Continued)**

| | Dual Throat | | |
|---|---|---|---|
| Mode 1: | | | |
| Oxidizer | Oxygen | Oxygen | Oxygen |
| Fuel 1 | NA | Propane | RP-1 |
| Fuel 2 | Hydrogen | Hydrogen | Hydrogen |
| MR Oxidizer (%) | 87.42 | 79.48 | 77.25 |
| MR Fuel 1 (%) | NA | 17.37 | 19.61 |
| MR Fuel 2 (%) | 12.58 | 3.45 | 3.14 |
| Isl1 (sec) | 366.00 | 326.00 | 325.00 |
| Iv1 (sec) | 442.00 | 375.00 | 372.00 |
| Fsl/We (lbf/lbm) | 73.72 | 69.49 | 71.67 |
| Mode 2: | | | |
| Oxidizer | Oxygen | Oxygen | Oxygen |
| Fuel 1 | NA | NA | NA |
| Fuel 2 | Hydrogen | Hydrogen | Hydrogen |
| MR Oxidizer (%) | 87.69 | 87.66 | 87.61 |
| MR Fuel 1 (%) | NA | NA | NA |
| MR Fuel 2 (%) | 12.31 | 12.34 | 12.39 |
| Isl2 (sec) | NA | NA | NA |
| Iv2 (sec) | 461.00 | 461.00 | 471.00 |

Table 5.3-2 SSTO Engine Performance Characteristics (Concluded)

| | Plug Nozzle | | |
|---|---|---|---|
| **Mode 1:** | | | |
| Oxidizer | Oxygen | Oxygen | Oxygen |
| Fuel 1 | NA | Propane | RP-1 |
| Fuel 2 | Hydrogen | Hydrogen | Hydrogen |
| MR Oxidizer (%) | 87.50 | 79.56 | 77.29 |
| MR Fuel 1 (%) | NA | 17.13 | 19.43 |
| MR Fuel 2 (%) | 12.50 | 3.31 | 3.28 |
| Isl1 (sec) | 354.00 | 344.00 | 340.50 |
| Iv1 (sec) | 460.00 | 401.00 | 397.00 |
| Fsl/We (lbf/lbm) | 80.35 | 106.67 | 109.09 |
| **Mode 2:** | | | |
| Oxidizer | Oxygen | Oxygen | Oxygen |
| Fuel 1 | NA | NA | NA |
| Fuel 2 | Hydrogen | Hydrogen | Hydrogen |
| MR Oxidizer (%) | 87.50 | 87.47 | 87.47 |
| MR Fuel 1 (%) | NA | NA | NA |
| MR Fuel 2 (%) | 12.50 | 12.53 | 12.53 |
| Isl2 (sec) | NA | NA | NA |
| Iv2 (sec) | 460.00 | 460.00 | 460.00 |

All three vehicle configurations used in this study (a VTHL winged body, a VTHL lifting body and a VTOL side entry cone) were SSTO. Both the lifting body and side entry conical vehicle configurations were included in this study because they are being considered by industry as possible alternatives to a winged body vehicle configuration. Due to the lack of publicly available information, the lifting body configuration and the side entry conical vehicle configuration used in this study were independently defined. Both of the configurations used the full set of engines and propellant combinations shown in Table 5.3-1.

The winged body vehicle configuration used in this study was derived from the winged body vehicle configuration in the NASA Option Three Access to Space Study. The vehicle body length to diameter ratio, the wing shape, the wing loading, the winglet definition, propellant tank locations, and the payload bay definition were all taken from the Option Three Access to Space Study final report. The winged body vehicle configuration was included in this trade study to calibrate the vehicle sizes and masses generated by the sizing tools used in this trade study against the sizes and masses of this vehicle configuration generated in the Option Three Access to Space Study. Due to lack of time, only the two engines from the Option Three Access to Space Study in Table 5.3-1 were used on this vehicle configuration.

Data on the first two engines in Table 5.3-1 were taken from the Option Three Access to Space Study. The version of the evolved SSME used here has a lower chamber pressure, a larger throat and a smaller nozzle area ratio than a standard SSME. This engine used the oxygen/hydrogen propellant combination. The RD-701 engine used here is the version of the RD-701 engine used in the Option Three Access to Space Study. This engine used the oxygen/hydrogen/RP-1 propellant combination. Unlike the other bell nozzle engines, the RD-701 engine was assumed to have a self contained hydraulic engine gimbal system built into it. The mass of this gimbal system was included in the engine mass, therefore the vehicle mass model did not include an allowance for the engine gimbal system mass or an allowance for the mass of the engine gimbal system energy supply system. These two engine designs do not have a common set of groundrules.

The next engine in Table 5.3-1 is the RD-704 engine. This engine data was supplied by Pratt & Whitney. This engine used the oxygen/hydrogen/RP-1 propellant combination. It was added to the study engine and propellant trade matrix because NASA is interested in the Russian tripropellant engine for SSTO application.

The major difference between the RD-701 and RD-704 engines is the lower hydrogen flow rate of the RD-701 engine during Mode 1. The RD-701 engine therefore has a lower specific impulse and higher propellant density during Mode 1. Also, the vehicle configuration with the RD-704 engines has a separate engine gimbal system.

Data on the next three engines in Table 5.3-1 were from a Rocketdyne study on advanced engines for SSTO applications. These three engines used the oxygen/hydrogen propellant combination. The first engine used a Full Flow Staged Combustion Cycle (FFSCC). The second engine used a dual mixture ratio cycle where the engine mixture ratio was 10/1 at liftoff. The engine shifted to a mixture ratio of 7/1 during ascent to orbit. The third engine used an expander cycle. Due to the smaller amount of power available to drive the turbopumps in an expander cycle engine, this engine used a significantly lower chamber pressure and nozzle area ratio than the other two Rocketdyne engine designs. The FFSCC engine is heavier and has higher specific impulses than the dual mixture ratio engine and the expander cycle engine. Since the dual mixture ratio engine runs at a higher mixture ratio, it's propellant combination is denser than the propellant combination used by the other two engine cycles. A common set of groundrules was used in these three engine designs. The three engines were added to the engine and propellant trade study matrix to enable a comparison between vehicle configurations with a conventional staged combustion cycle bell nozzle engine, a dual mixture ratio bell nozzle engine and an expander cycle bell nozzle engine.

Data on the last three engines in Table 5.3-1 were generated by Aerojet for this trade study. To enable comparisons between vehicle configurations using different propellant combinations, the three engine types each have a version that could use the propellant combinations of oxygen/hydrogen, oxygen/hydrogen/RP-1, and oxygen/ hydrogen/propane. The dual expansion engine is a bell nozzle engine with a ring of thrust cells wrapped around its throat. During Mode 1, both the center engine and the thrust cells are operating. During Mode 2, the thrust cells are shut down and the center engine only is operating. The second engine type is a dual throat. This type of engine uses a small modular, rectangular thrust cell split into two parts by a partition in the combustion chamber (Figure 5.3-1). During Mode 1, both parts of the combustion chamber are operating. During Mode 2, one part of the combustion chamber is shut down. The last engine type is a plug nozzle engine. The engines using the propellant combination of oxygen/hydrogen have the highest Mode 1 vacuum specific impulse. The engines using the propellant combination of oxygen/hydrogen/ propane have a slightly higher Mode 1 vacuum specific impulse than the engines using the propellant combination of oxygen/hydrogen/RP-1. The dual throat engines have a lower engine thrust-to-weight ratio than the dual expansion engines. Across the three propellant combinations, the engine thrust-to-weight ratio does not change significantly for the dual throat and dual expansion engines. For the propellant combination of oxygen/hydrogen, the plug nozzle engine has a slightly better thrust-to-weight ratio than the dual expansion engine. Unlike the other two engine types, the engine thrust-to-weight ratio improves significantly as the propellant combination is changed from oxygen/hydrogen to oxygen/ hydrogen/propane and then to oxygen/hydrogen/RP-1.

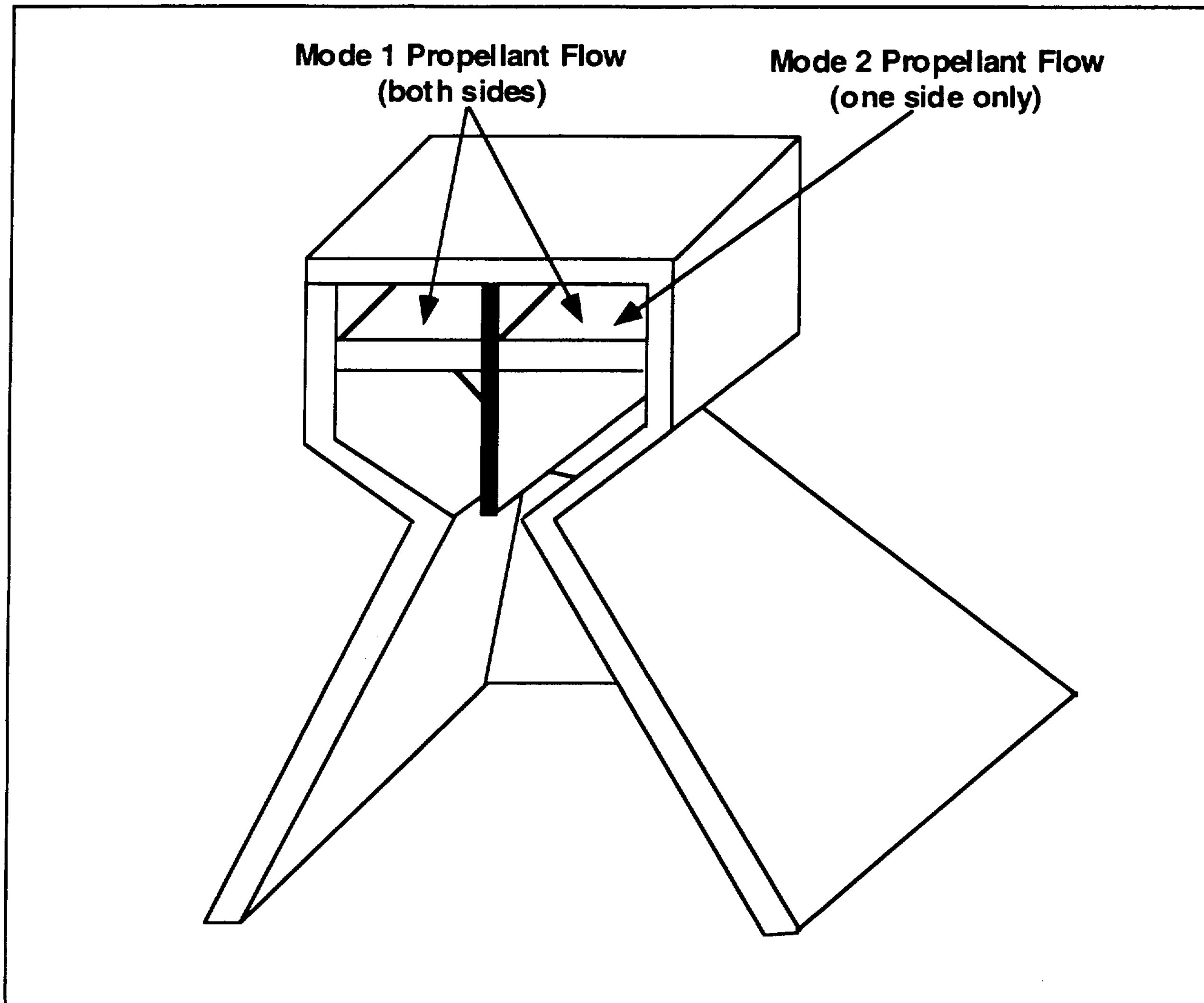



**Figure 5.3-1 Typical Dual Throat Thrust Cell Concept**

With the exception of the Rocketdyne expander cycle engine, all engines in this trade study use a staged combustion power cycle.

With the exception of the Aerojet dual throat and plug engines, all engines in this main propulsion matrix use a bell nozzle.

The bell engine cases use engine gimbaling to provide vehicle Thrust Vector Control (TVC). The engines are gimbaled by electromechanical actuators. Vehicle prime power supplies the energy required to gimbal the engines. The dual throat and plug engine cases use differential throttling for TVC. (The RD-701 engines are assumed to use self-contained hydraulic engine gimbaling systems, therefore no vehicle resources are used.)

Table 5.3-3 shows the technology assumptions used in this trade study that were common to all vehicle configurations. Most of these technology assumptions were taken from the Option Three Access to Space Study. The exceptions were the use of a graphite epoxy liquid hydrogen tank and the use of Space Shuttle fuel cells for prime power. The graphite epoxy liquid hydrogen tank was used because this is the direction that the NASA RLV program is heading. The Space Shuttle fuel cells were used for prime power because data was not available on the advanced fuel cells and batteries used in the Option Three Access to Space Study. The side entry conical VTOL vehicle configuration has additional technology assumptions associated with being able to perform the landing maneuver. These additional technology assumptions will be discussed in the conical side entry VTOL vehicle configuration section.

Table 5.3-4 shows the vehicle sizing groundrules used in this trade study that were common to all vehicle configurations. The groundrules that were specific to each vehicle configuration are discussed in the vehicle configuration sections. Most of the technology assumptions were taken from the Option Three Access to Space Study.

The decision to vent the main propellant tanks and feed lines was partially an operability issue decision and partially a performance decision. The operability issue was the potential safety hazard of working around a vehicle with propellant trapped in the main propellant tanks and feed lines. The performance issue was venting residuals reduces the mass of the vehicle during on-orbit operations and therefore it reduces the amount of propellant required for on-orbit operations.

All vehicle configurations in this trade study are designed to land with the design payload in the payload bay after the ascent propellant has been burned.

**Table 5.3-3 Technology Assumptions**

- Graphite epoxy is used for the LH2 tank
- * • Graphite epoxy is used for the unpressurized structures
- * • Aluminum-lithium is used for the LO2 and kerosene tanks
- Aluminum-lithium is used for the propane tank
- * • Skin stringer construction is used for the propellant tank construction
- * • Honeycomb with ring frames is used for the unpressurized structures
- * • Thermal Protection System
  - Advanced Carbon Carbon (ACC) is used for the high temperature areas
  - Tailorable Advanced Blanket Insulation (TABI) is used on windward side of the vehicle
  - Advanced Flexible Reusable Surface Insulation (AFRSI) is used on the leeward side of the vehicle
  - The blankets are attached to the structure by a silicone rubber adhesive (RTV)
- * • Engine bay heat shield is a graphite epoxy honeycomb structure with a TABI blanket bonded to it
- * • Propellant tank cryogenic insulation is an external Rhoacell foam
- * • Advanced composite landing gear is used
- * • The main propellant system (MPS) uses composite and metallic feedlines with foam insulation
- * • The RD-701 engines use a self contained hydraulic system to gimbal the engines
- * • The thrust structure uses graphite epoxy truss
- * • Reaction Control System (RCS) uses pressure fed LO2/LH2 engines
- * • Orbital Maneuvering System uses pump fed LO2/LH2 engine

**Same as Access to Space Option 3 SSTO(R) Assumption*

**Table 5.3-3 Technology Assumptions (Concluded)**

* * • Prime power is supplied by Space Shuttle Orbiter O2/H2 fuel cells and batteries

* * • Power conversion and distribution system supplies 270 volt DC electrical power to vehicle systems
  - Power conversion is done locally

* * • Electromechanical actuators (EMAs) with light-weight rare earth magnets are used to move aero surfaces

* * • Avionics
  - Adaptive guidance navigation & control (GN&C)
  - Health monitoring systems
  - Smart sensors

* * • Environmental control and life support systems
  - No crew on the vehicles modeled
  - Avionics waste heat is heat sinked into the vehicle structure

* Configuration specific technology assumptions will be discussed in the configuration sections

**Same as Access to Space Option 3 SSTO(R) Assumption*

**Table 5.3-4 Sizing Groundrules**

- * • 25,000 lbm payload
- * • Payload bay size is 15 feet in diameter and 30 feet long
- * • No crew
- * • 3 Gs maximum acceleration during ascent
- * • Mission duration is 7 days
- • 1.4 factor of safety used for items subjected to a dynamic environment
  - Applied to ultimate strength of materials
  - Used in sizing of wings and unpressurized structures
- • Allowable stresses reduced by 20% to account for fatigue
- * • Target orbit is a 220 n.mi. circular orbit with 51.6° inclination (Space Station)
- * • MECO condition is 50 by 100 n.mi. orbit with 51.6° inclination
- • Propellant tank ullage factor is 5%
- * • RD-701 engine used
  - Described in the Access to Space, Advanced Technology Team Final Report
  - Updated propellant mass flow rate data supplied by Doug Stanley/NASA-Langley
  - RD-701 engine gimbal system weight included in engine weights
- * • Liftoff thrust-to-weight is 1.2 Gs
- * • Electromechanical actuators are used
  - RD-701 engine gimbal system is self contained
- * • Oxygen/hydrogen OMS and RCS systems are used
  - OMS velocity budget is 1,100 ft/sec
  - RCS velocity budget is 110 ft/sec for on-orbit operations and 40 ft/sec for entry
  - OMS and RCS engine performance is from the Access to Space, Advanced Technology Team Final Report

**Same as Access to Space Option 3 SSTO(R) Assumption*

**Table 5.3-4 Sizing Groundrules (Concluded)**

* • Flight performance reserves
  - Ascent flight performance reserve is 1% of ascent velocity and is bookkept as 1% degradation in engine specific impulse
  - OMS and RCS flight performance reserve, 40 ft/sec and 45 ft/sec respectively, is bookkept as additional on-orbit propellant

- Propellant densities
  - LO2 density is 71.20 $lbm/ft^3$
  - LH2 density is 4.43 $lbm/ft^3$
  - Kerosene density is 50.50 $lbm/ft^3$
  - Propane density is 36.26 $lbm/ft^3$

- Main propellant tanks and propellant feed systems are vented upon reaching orbit (operability issue)
  - Main propellant tanks are pressurized to just over one atmosphere for entry
  - Main propellant flight performance reserves and residuals are vented

- Thrust structure mass for the modular engine vehicle configurations and the plug nozzle engine vehicle configurations are 75% of the thrust structure mass of the bell engine vehicle configurations

- Average on-orbit power demand is 5 kw

- Average on-orbit heat rejection demand is 10 kw

- Configuration specific groundrules will be discussed in the configuration sections

**Same as Access to Space Option 3 SSTO(R) Assumption*

## 5.4 Sizing Tool Description

Launch vehicle configurations modeled by the sizing tools are a conical side entry VTOL SSTO, a winged body VTHL SSTO, and a lifting body VTHL SSTO.

The launch vehicle sizing tools use an iterative approach to calculate the size and mass of a launch vehicle configuration, as was shown previously in Figure 5.2-1. The sizing tools estimate the velocity requirement to reach the mission orbit, the propellant load required to reach the velocity requirement, and the vehicle structural mass necessary to contain the amount of propellant required. Each iteration gets these three parameters closer to a converged solution. After the vehicle sizing calculations have converged, the estimate of the velocity requirement to reach the mission orbit should be checked against a trajectory analysis on the resulting vehicle configuration. This trajectory analysis should then be used to calculate a temperature map of the launch vehicle configuration during reentry. This temperature map should be used to check the initial assumptions on the thermal protection system (TPS) required to protect the launch vehicle configuration during reentry.

The mission requirements are entered into the sizing tool input data file. These requirements include the payload mass, payload bay size, acceleration limits, destination orbit inclination, apogee, and perigee, on-orbit mission velocity requirements, number of crew, time spent on-orbit, and the average on-orbit power and heat rejection requirements. The Q-bar and Q-alpha limits are not used by the sizing tools. However, they do come into play when the mission velocity requirement is refined by a trajectory analysis.

The side entry cone VTOL sizing tool, the winged body VTHL sizing tool, and the lifting body VTHL sizing tool were developed from a generic SSTO sizing tool. Separate sizing tools were developed because these launch vehicle configurations were too different to be covered by a single general purpose sizing tool. These three sizing tools have a performance spreadsheet and a weights spreadsheet, as illustrated in Figure 5.4-1. Information flows both ways between these spreadsheets until the sizing tool has converged on a solution.

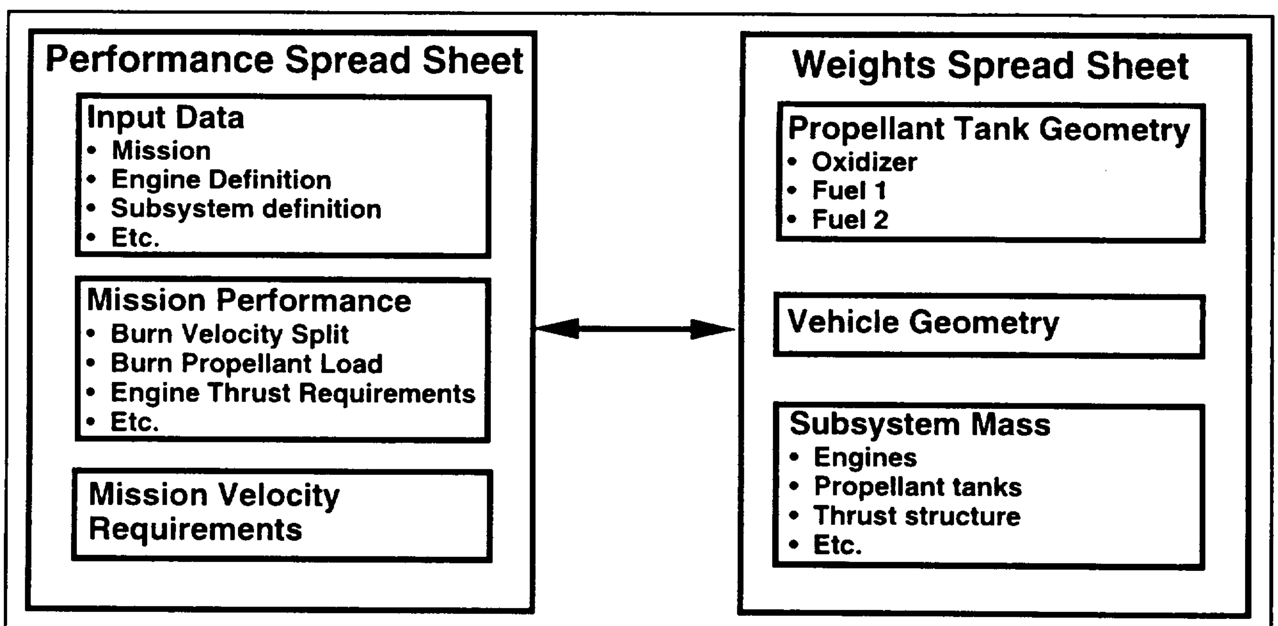



**Figure 5.4-1 Common Sizing Tool Features**

The performance spreadsheet has an input data section, a mission performance section, and a mission velocity requirements section. The input data section contains all of the data used by the sizing tool to define the launch vehicle configuration model. The mission performance section uses the vehicle masses supplied by the weights spread sheet, the calculated mission velocity requirement, and the rocket equation to calculate the amount of propellant needed by the vehicle to reach main engine cutoff (MECO) conditions. The mission delta velocity requirements section calculates the delta velocity required as a function of the Mode 1 burn and the Mode 2 burn initial thrust-to-weight ratios.

The weights spreadsheet has a propellant tank geometry section, a vehicle geometry section, and a subsystem mass section. The propellant tank geometry is calculated from the propellant load requirement supplied by the performance spreadsheet. The vehicle geometry is calculated from the tank geometry. The vehicle masses are calculated from the vehicle geometry, the propellant masses, and the engine thrust supplied from the performance spreadsheet.

The vehicle masses are then supplied back to the performance spreadsheet for the next pass through the iterative loop. This iterative loop continues until the model has converged onto a solution.

The approach used in the sizing tools is to split the mission velocity required to reach orbit into endoatmospheric (first) and exoatmospheric (second) velocity segments. The sizing program calculates the total velocity requirement. The user supplies a burn two velocity estimate. The sizing program then calculates the burn one velocity requirement, the burn one and two propellant requirements and the resulting vehicle masses. The second burn delta velocity capability is varied by the user to find the first and second burn propellant loads that result in the total minimum structural mass. This approach was used to allow the use of rocket engines that have different performance characteristics in modes one and two. If the vehicle configuration used the mixed mode concept, Mode 1 is burn one and Mode 2 is burn two.

The vehicle sizing tools will converge on a single point design. Each of the three launch vehicle configurations have their own configuration specific geometrical sizing parameters. In addition, the burn two velocity split, the initial vehicle liftoff thrust-to-weight ratio, and the burn two initial thrust-to-weight ratio are general launch vehicle sizing parameters. The configuration can be optimized by varying the sizing parameters.

The sizing tools use a thrust structure mass model where the thrust structure mass is a function of the number of engines and the engine maximum thrust. This thrust structure mass model was designed for use with the point loads from bell nozzle rocket engines. If the rocket engines have a distributed load around the edge of the vehicle and there is a short load path to react these engine thrust loads into the vehicle mass, one engine and a reduced set of thrust structure coefficients are used.

Table 5.4-1 shows the sources of the vehicle subsystem mass equations used in the development of the sizing tools.

Further information can be found in the sizing tools User's Guide in Section 8.

**Table 5.4-1 Sizing Tool Description**

| |
|---|
| Parentage of the Sizing Model |
| • Most of the equations and some of the technology coefficients were from NASA TM 78661, "Techniques for the Determination of Mass Properties of Earth-To-Orbit Transportation Systems," by I. O. MacConochie and P. J. Klich, June 1976 |
| • Additional technology coefficients were from "Space Transportation Architecture Study Special Report - Final Phase, Book 3," General Dynamics Space System Division, November 1987, Contract NAS8-36615 |
| • Residual propellant equation and the data that was used to develop the thrust structure equations were from "Space Shuttle Synthesis Program (SSSP), Volume II, Weight/Volume Handbook Final Report," General Dynamics Convair Aerospace Division, December 1970, Contract NAS9-11193 |
| • An equation to calculate the non optimum weight factors on the design of propellant tanks was from "A Semi-Empirical Method for Propellant Tank Weight Estimation," L. A. Willoughby, 27th Annual Conference of the Society of Aeronautical Weight Engineers, May 1968 |
| • Space Shuttle Orbiter component mass information was from "Orbiter Detail Weight Statement (OV-103)," SD75-SH-0116-216, Rockwell International, August 2, 1993 and "Press Information, Space Shuttle Transportation System," Rockwell International, January 1984 |
| • The SSTO(R) component mass information was from "Access to Space Study, Advanced Technology Team (Option 3) Final Report," July 1993 |
| • Equations to calculate the unpressurized structure unit mass for the side entry conical configuration were from "Aerospace Vehicle Design, Volume II, Spacecraft Design", by K. D. Wood |

## 5.5 Side Entry Conical VTOL Concept

This launch vehicle configuration is a conical VTOL design with integral propellant tanks. The vehicle reenters on its side. Prior to landing, it does a rotation maneuver. This rotation maneuver changes the vehicle orientation from horizontal to vertical. The launch vehicle then lands vertically on it's base.

The vehicle configuration uses some combination of the main engines, the reaction control system, and the body flaps to rotate the vehicle. The two basic approaches to the rotation maneuver are to rotate the vehicle without changing it's velocity vector or to bring the vehicle's velocity vector to a halt during the rotation maneuver. The first alternative would minimize the propellant used during the rotation maneuver. However, it would require the launch vehicle to fly through it's main rocket engine plume. The second alternative would not require the launch vehicle to fly through its main rocket engine plume. However, the launch vehicle's body flaps would loose effectiveness during the rotation maneuver and the propellant requirements for the rotation and landing maneuvers would be higher.

The rotation/landing maneuver requires the ability to ignite and throttle the engines that will be used for the rotation/landing maneuver in a timely and dependable manner.

Since there was not sufficient time to optimize the rotation and landing maneuvers, the body flap size and landing hover time used in sizing these launch vehicle configuration cases are the best current estimates and are subject to further design iterations.

This launch vehicle concept requires the assumption that there are satisfactory answers to the problems in rotating the vehicle and igniting the engines for rotation/landing maneuver.

Since this launch vehicle configuration is a cone with integral propellant tanks, it has a load path going from the main engines through the skin of the vehicle to the propellant tanks and payload. Therefore, the cases using the dual throat engines and the plug nozzle engines have their engines on the periphery of the launch vehicle base. These cases therefore use smaller thrust structure coefficients (see Section 5.4).

Table 5.5-1 shows the sizing groundrules that are specific to this launch vehicle configuration. The entry RCS velocity budget was increased from 40 ft/sec to 80 ft/sec because a conical axisymmetric vehicle configuration must be held at a sideslip angle to give it a cross range capability and the RCS thrusters would be used to do this prior to the body flaps becoming effective. The allowance of 16 seconds of hover time (which translates to a 1,000 ft/sec rotation/ landing maneuver velocity requirement) and a total body flap planform area of 25 % of the vehicle base area were used prior to doing an optimization of the rotation and landing maneuvers.

Figures 5.5-1, 5.5-2, and 5.5-3 show the results of a vehicle configuration sensitivity study on the rotation and landing maneuver requirements, the vehicle base diameter and the vehicle cone half angle respectively. This vehicle configuration is very sensitive to a rotation landing maneuver velocity requirement larger than 1,500 to 2,000 ft/sec. There is an optimum value for the vehicle base diameter and cone half angle. This study used a landing hover time of 16 seconds (which translates into a rotation/landing maneuver velocity requirement of approximately 1,000 ft/sec) and a vehicle cone half angle of 5.5 degrees. An optimum vehicle diameter was found for each engine and propellant combination case.

Cases were run using the side entry conical VTOL vehicle configuration and the engine and propellant combination shown in Table 5.3-1. The resulting vehicle dry masses are plotted in Figure 5.5-4. Descriptions of the resulting vehicle configuration mass properties and sizes for these cases are shown in Figures 5.5-5 through 5.5-19.

The first two cases in Figure 5.5-4 use the Evolved SSME and the RD-701 engines. As shown in the option three Access to Space final report, a vehicle configuration using the RD-701 engines is significantly lighter than a vehicle configuration using the Evolved SSME.

The next case uses the RD-704 engine. The dry mass of a vehicle configuration using the RD-704 engine is about half way between the dry mass of vehicle configurations using the Evolved SSME and the RD-701 engines. The reasons for the difference in the dry masses of vehicle configurations using the RD-701 engines and the RD-704 engines are the RD-704 engine is heavier and the extra hydrogen flow during the Mode 1 burn make the RD-704 vehicle configuration propellant tanks larger and therefore heavier.

The next three cases use the Rocketdyne SSTO study engines. A vehicle configuration using FFSCC engines is lighter than vehicle configurations using dual mixture ratio engines or expander cycle engines. The higher specific impulse of the FFSCC engines offset this engine cycle's higher weights.

The next nine cases used the three Aerojet engines with the three propellant combinations.

Vehicle configurations using the dual throat engine and the plug nozzle engines are lighter than vehicle configurations using the dual expansion cycle engines. The reason for this is thought to be the lighter thrust structures and the use of differential throttling for TVC used on the vehicle configurations using these engines. There was not enough time to check this hypothesis out.

Vehicle configurations using the plug nozzle cycle engines were lighter than vehicle configurations using dual throat cycle engines.

A striking result here is the change in the relative ranking of the vehicle dry mass for the different propellant combinations and engine types. The conclusion is the best propellant combination is a function of which type of engine that is used.

**Table 5.5-1 VTOL Side Entry Cone Concept Results**

**Configuration Specific Sizing Groundrules**

- Payload bay mass is 5,786 lbm (Option 3 vehicle payload bay mass and mass of the faring over the payload bay and crew cabin)
- Payload bay is transverse to the vehicle axis
- Entry RCS budget has increased to 80 ft/sec to allow holding the vehicle at a side slip angle to increase the vehicle crossrange capability by use of the RCS jets prior to the body flaps becoming effective
- Vehicle has an allowance of 16 seconds of hover time after the vehicle terminal velocity has been nulled (rotation/landing maneuver velocity requirement approximately 1000 ft/sec)
- There are four body flaps with a total planform area of 25% of the vehicle base area
- A minimum vehicle area unit weight of one pound psf is used for the unpressurized structures
- The nose cone is a biconic with hemispherical nose tip; dimensions are defined by the user
- Number of engines
  - Vehicle configurations using bell engines have seven engines
  - Vehicle configurations using modular engines and plug nozzle engines have one engine

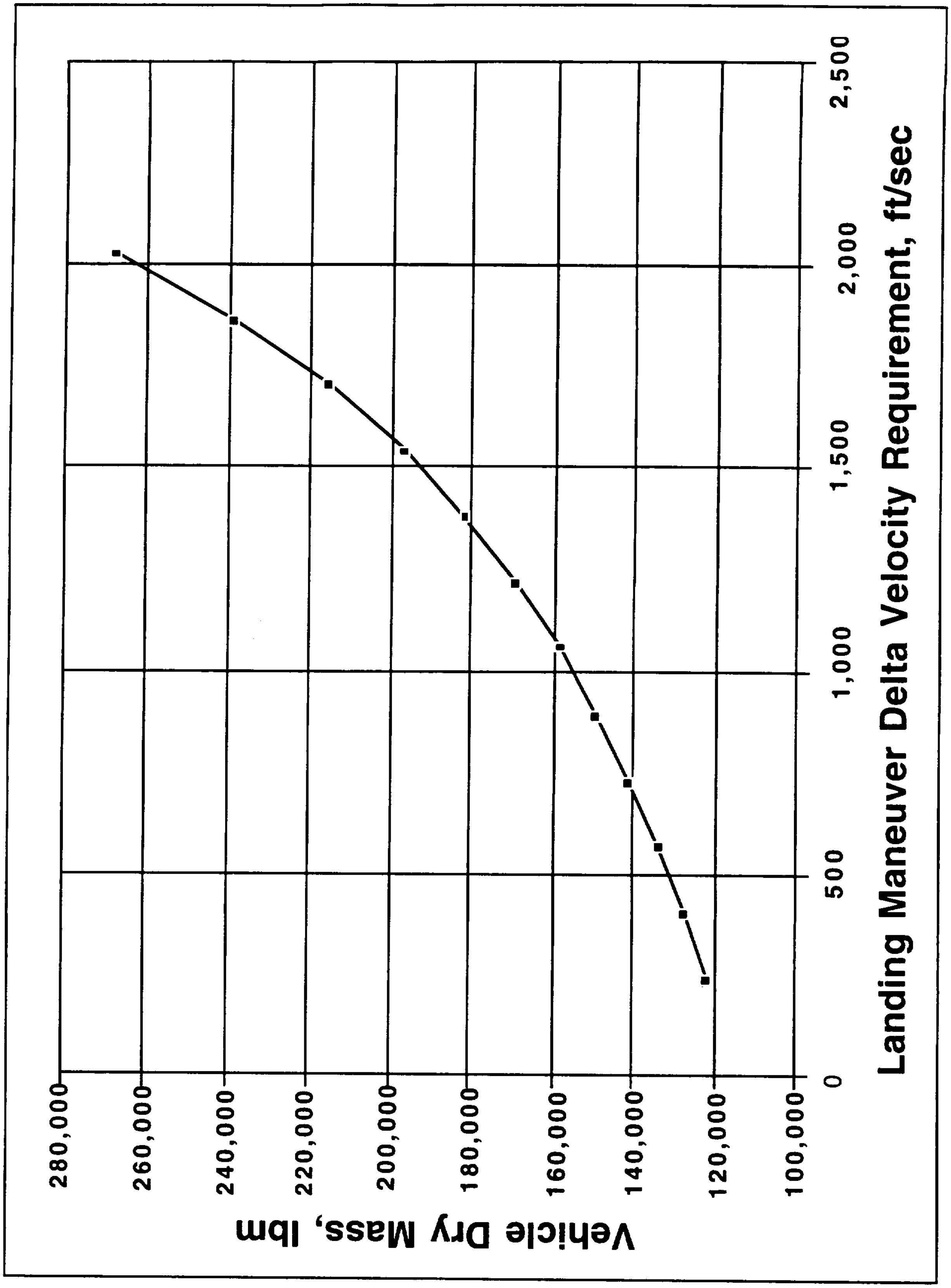



**Figure 5.5-1 Relationship between Vehicle Dry Mass and Landing Maneuver Velocity Requirement**

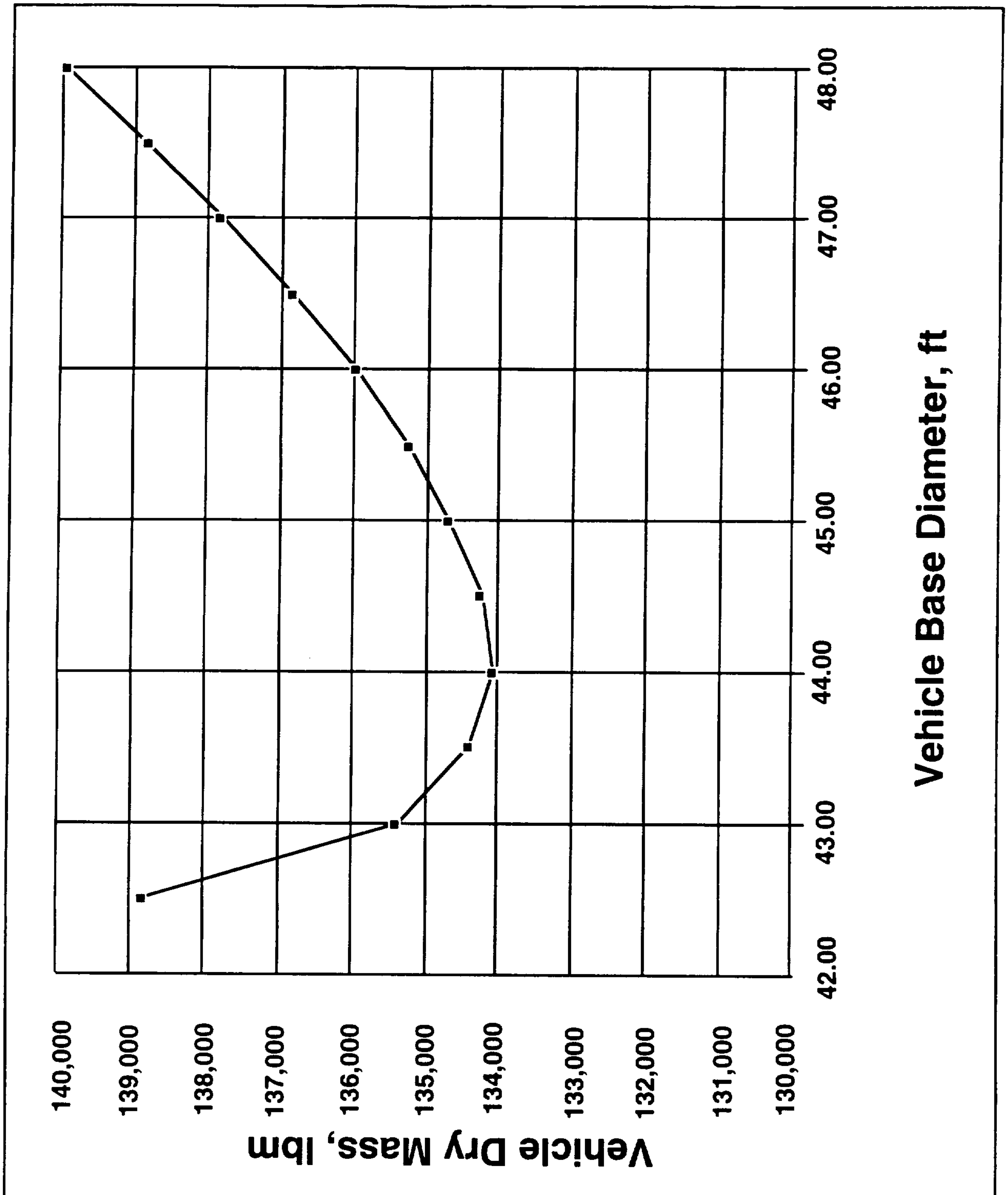



**Figure 5.5-2 Relationship Between Vehicle Dry Mass and Base Diameter**

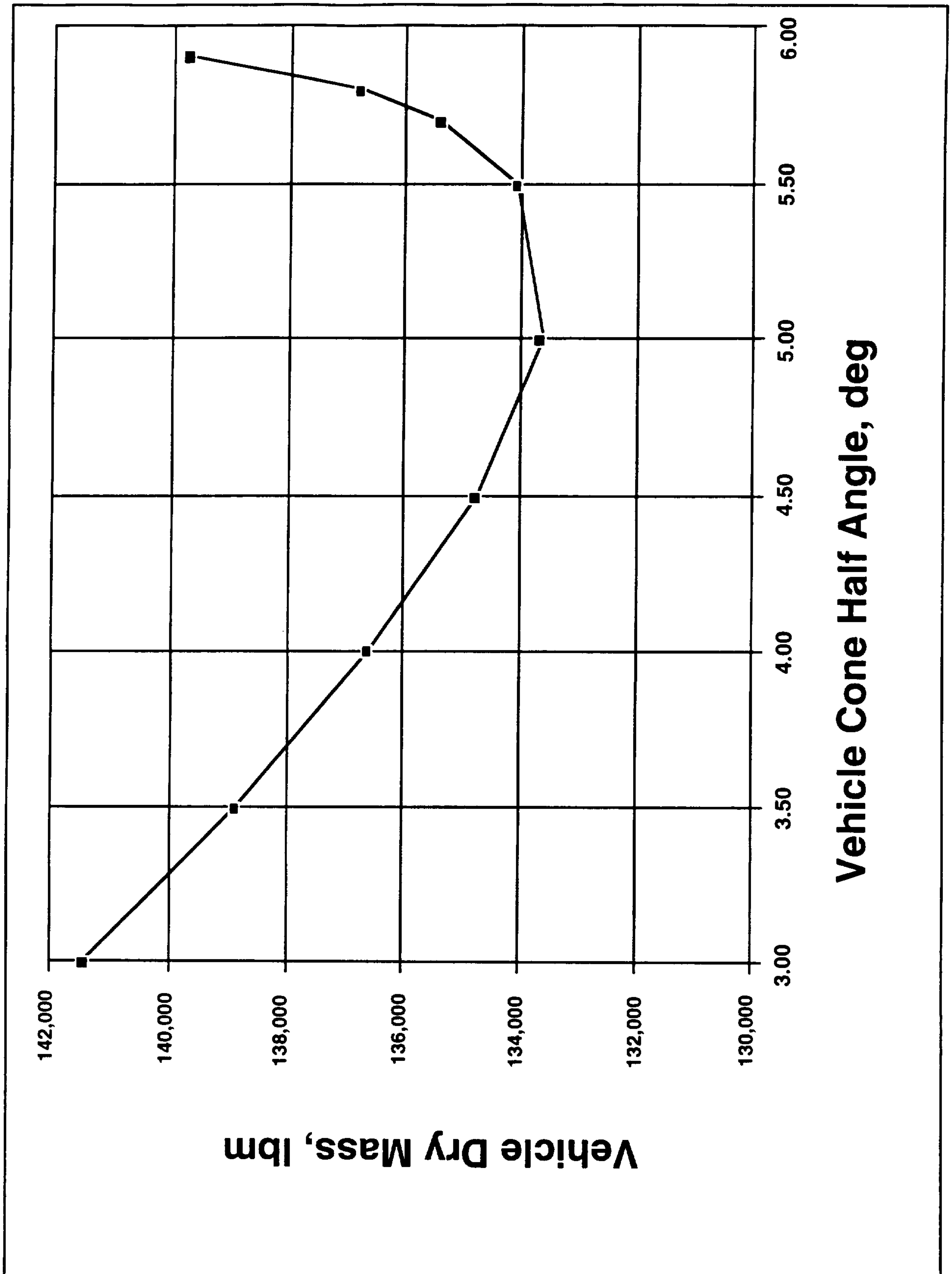



Figure 5.5-3 Relationship between Vehicle Dry Mass and Cone Half Angle

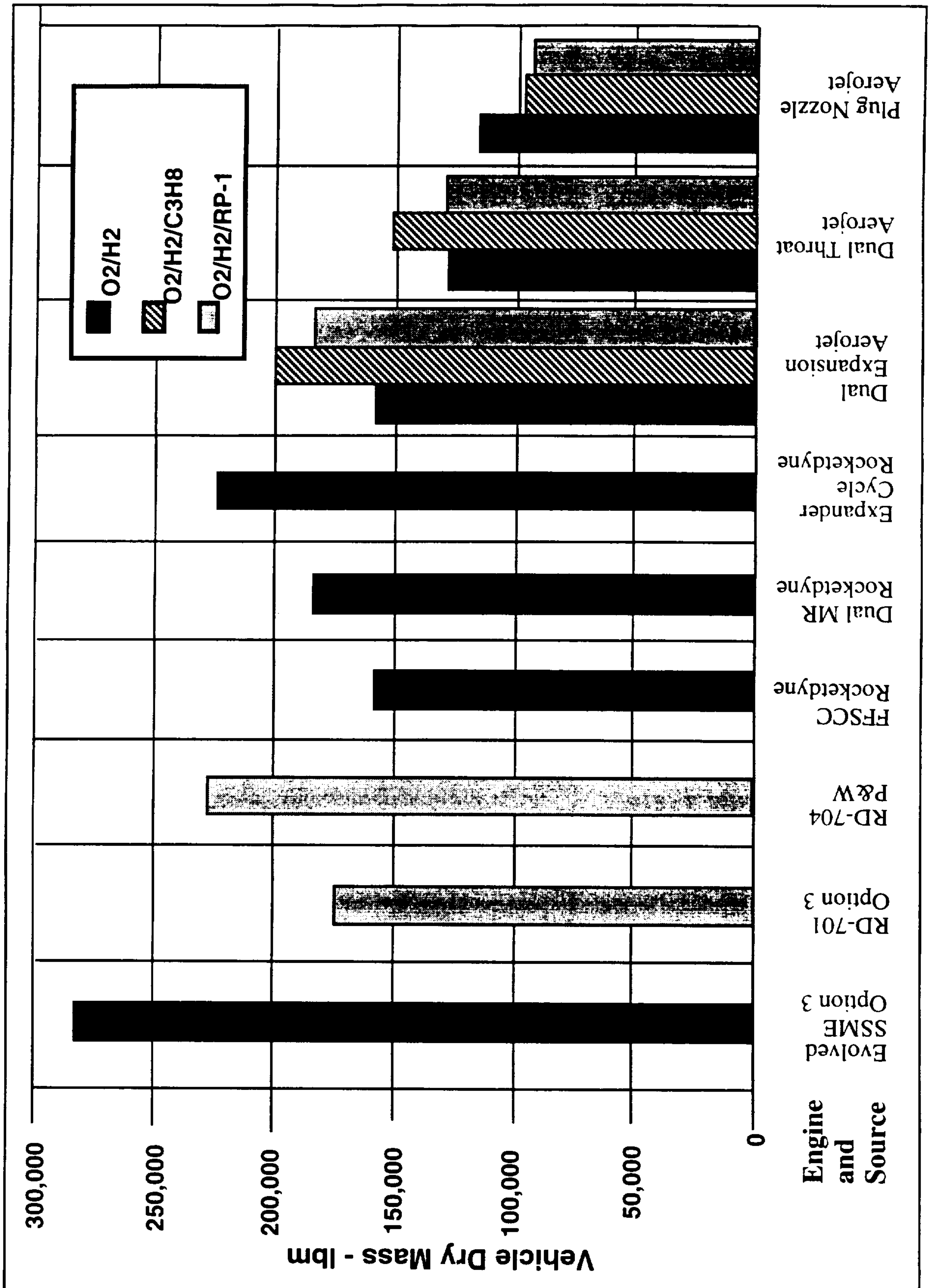



Figure 5.5-4 Side Entry Conical Vehicle Configuration Dry Mass as a Function of Engine and Propellant Used

| | |
|---|---|
| **GLOW:** | 3,273,748 lbm |
| **Length:** | 157 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 284,383 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,855,166 lbm |
| --Mode 2 | 1,026,322 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 14,996 lbm |
| OMS & RCS Propellant | 33,907 lbm |
| Landing Propellant | 33,975 lbm |
| Landing Specific Impulse | 390.4 sec |
| Main Engine Type/No.: | Evolved SSME/7 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 561,214 lbf |
| Sea Level Isp (@100 % RPL): | 390.4 sec |
| Vacuum Thrust per Engine (@100% RPL) | 643,010 lbf |
| Vacuum Isp (@100 % RPL): | 447.3 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 283,717 lbf |
| Vacuum Isp (@100 % RPL): | 447.3 sec |

**Figure 5.5-5 Side Entry Conical Vehicle Using Evolved SSMEs Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,409,834 lbm |
| **Length:** | 145 ft |
| | |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 176,196 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,376,847 lbm |
| --Mode 2 | 772,038 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 11,394 lbm |
| OMS & RCS Propellant | 22,429 lbm |
| Landing Propellant | 25,930 lbm |
| Landing Specific Impulse | 333.5 sec |
| Main Engine Type/No.: | RD-701/7 |
| | |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 413,114 lbf |
| Sea Level Isp (@100 % RPL): | 333.5 sec |
| Vacuum Thrust per Engine (@100% RPL) | 477,033 lbf |
| Vacuum Isp (@100 % RPL): | 385.1 sec |
| | |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 206,598 lbf |
| Vacuum Isp (@100 % RPL): | 452.7 sec |

**Figure 5.5-6 Side Entry Conical Vehicle Using RD-701 Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,876,495 lbm |
| **Length:** | 158 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 228,325 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,586,898 lbm |
| --Mode 2 | 964,294 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 13,391 lbm |
| OMS & RCS Propellant | 28,033 lbm |
| Landing Propellant | 30,553 lbm |
| Landing Specific Impulse | 356.0 sec |
| Main Engine Type/No.: | RD-704/7 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 493,113 lbf |
| Sea Level Isp (@100 % RPL): | 356.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 563,756 lbf |
| Vacuum Isp (@100 % RPL): | 407.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 257,919 lbf |
| Vacuum Isp (@100 % RPL): | 452.0 sec |

**Figure 5.5-7 Side Entry Conical Vehicle Using RD-704 Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,826,799 lbm |
| **Length:** | 136 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 159,467 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 928,685 lbm |
| --Mode 2 | 665,710 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 8,321 lbm |
| OMS & RCS Propellant | 20,140 lbm |
| Landing Propellant | 19,477 lbm |
| Landing Specific Impulse | 401.7 sec |
| Main Engine Type/No.: | Full Flow Staged Combustion/7 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 313,166 lbf |
| Sea Level Isp (@100 % RPL): | 401.7 sec |
| Vacuum Thrust per Engine (@100% RPL) | 358,850 lbf |
| Vacuum Isp (@100 % RPL): | 460.3 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 179,623 lbf |
| Vacuum Isp (@100 % RPL): | 460.3 sec |

Figure 5.5-8 Side Entry Conical Vehicle Using Full Flow Staged Combustion Engines Concept Summary

| | |
|---|---|
| **GLOW:** | 2,352,271 lbm |
| **Length:** | 136 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 186,688 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,289,698 lbm |
| --Mode 2 | 791,870 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 11,141 lbm |
| OMS & RCS Propellant | 23,328 lbm |
| Landing Propellant | 24,546 lbm |
| Landing Specific Impulse | 373.5 sec |
| Main Engine Type/No.: | Dual Mixture Ratio/7 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 403,246 lbf |
| Sea Level Isp (@100 % RPL): | 343.9 sec |
| Vacuum Thrust per Engine (@100% RPL) | 481,340 lbf |
| Vacuum Isp (@100 % RPL): | 410.5 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 212,515 lbf |
| Vacuum Isp (@100 % RPL): | 455.5 sec |

**Figure 5.5-9 Side Entry Conical Vehicle Using Dual Mixture Ratio Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,706,355 lbm |
| **Length:** | 150 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 225,781 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,409,245 lbm |
| --Mode 2 | 976,678 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 12,640 lbm |
| OMS & RCS Propellant | 27,663 lbm |
| Landing Propellant | 29,347 lbm |
| Landing Specific Impulse | 367.5 sec |
| Main Engine Type/No.: | Dual Expander Cycle Bell/7 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 463,947 lbf |
| Sea Level Isp (@100 % RPL): | 367.5 sec |
| Vacuum Thrust per Engine (@100% RPL) | 561,533 lbf |
| Vacuum Isp (@100 % RPL): | 444.8 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 259,422 lbf |
| Vacuum Isp (@100 % RPL): | 444.8 sec |

**Figure 5.5-10 Side Entry Conical Vehicle Using Dual Expander Cycle Bell Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,836,795 lbm |
| **Length:** | 129 ft |
| | |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 159,058 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 950,585 lbm |
| --Mode 2 | 651,884 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 8,616 lbm |
| OMS & RCS Propellant | 20,286 lbm |
| Landing Propellant | 21,366 lbm |
| Landing Specific Impulse | 366.0 sec |
| Main Engine Type/No.: | Dual Expanding Bell/7 |
| | |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 314,879 lbf |
| Sea Level Isp (@100 % RPL): | 366.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 385,426 lbf |
| Vacuum Isp (@100 % RPL): | 448.0 sec |
| | |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 177,242 lbf |
| Vacuum Isp (@100 % RPL): | 468.0 sec |

**Figure 5.5-11 Side Entry Conical Vehicle Using Dual Expanding Bell Engines and Oxygen/Hydrogen Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,681,674 lbm |
| **Length:** | 147 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 201,535 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,560,701 lbm |
| --Mode 2 | 829,845 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/C3H8 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 12,648 lbm |
| OMS & RCS Propellant | 25,029 lbm |
| Landing Propellant | 26,918 lbm |
| Landing Specific Impulse | 366.0 sec |
| Main Engine Type/No.: | Dual Expanding Bell/7 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 459,716 lbf |
| Sea Level Isp (@100 % RPL): | 329.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 522,596 lbf |
| Vacuum Isp (@100 % RPL): | 374.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 224,195 lbf |
| Vacuum Isp (@100 % RPL): | 462.0 sec |

**Figure 5.5-12 Side Entry Conical Vehicle Using Dual Expanding Bell Engines and Oxygen/Hydrogen /Propane Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,495,971 lbm |
| **Length:** | 142 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 185,347 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,455,061 lbm |
| --Mode 2 | 770,575 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 11,765 lbm |
| OMS & RCS Propellant | 23,239 lbm |
| Landing Propellant | 24,984 lbm |
| Landing Specific Impulse | 366.0 sec |
| Main Engine Type/No.: | Dual Expanding Bell/7 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 427,881 lbf |
| Sea Level Isp (@100 % RPL): | 329.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 485,105 lbf |
| Vacuum Isp (@100 % RPL): | 373.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 208,182 lbf |
| Vacuum Isp (@100 % RPL): | 462.0 sec |

**Figure 5.5-13 Side Entry Conical Vehicle Using Dual Expanding Bell Engines and Oxygen/Hydrogen/Kerosene Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,594,567 lbm |
| **Length:** | 118 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 129,834 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 832,800 lbm |
| --Mode 2 | 564,373 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 7,468 lbm |
| OMS & RCS Propellant | 17,070 lbm |
| Landing Propellant | 18,022 lbm |
| Landing Specific Impulse | 366.0 sec |
| Main Engine Type/No.: | Modular Dual Throat/1 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 1,913,481 lbf |
| Sea Level Isp (@100 % RPL): | 366.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 2,310,815 lbf |
| Vacuum Isp (@100 % RPL): | 442.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 1,066,474 lbf |
| Vacuum Isp (@100 % RPL): | 461.0 sec |

**Figure 5.5-14 Side Entry Conical Vehicle Using Modular Dual Throat Engine and Oxygen/Hydrogen Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,117,862 lbm |
| **Length:** | 131 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 153,726 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,230,508 lbm |
| --Mode 2 | 657,572 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/C3H8 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 10,035 lbm |
| OMS & RCS Propellant | 19,750 lbm |
| Landing Propellant | 21,271 lbm |
| Landing Specific Impulse | 366.0 sec |
| Main Engine Type/No.: | Modular Dual Throat/1 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 2,541,434 lbf |
| Sea Level Isp (@100 % RPL): | 326.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 2,923,429 lbf |
| Vacuum Isp (@100 % RPL): | 375.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 1,242,296 lbf |
| Vacuum Isp (@100 % RPL): | 461.0 sec |

**Figure 5.5-15 Side Entry Conical Vehicle Using Modular Dual Throat Engine and Oxygen/Hydrogen/Propane Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,806,057 lbm |
| **Length:** | 124 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 131,025 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,054,635 lbm |
| --Mode 2 | 551,177 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 8,525 lbm |
| OMS & RCS Propellant | 17,231 lbm |
| Landing Propellant | 18,464 lbm |
| Landing Specific Impulse | 366.0 sec |
| Main Engine Type/No.: | Modular Dual Throat/1 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 2,167,269 lbf |
| Sea Level Isp (@100 % RPL): | 325.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 2,480,689 lbf |
| Vacuum Isp (@100 % RPL): | 372.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 1,051,991 lbf |
| Vacuum Isp (@100 % RPL): | 471.0 sec |

**Figure 5.5-16 Side Entry Conical Vehicle Using Modular Dual Throat Engine and Oxygen/Hydrogen/Kerosene Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,428,257 lbm |
| **Length:** | 120 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 116,816 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 726,404 lbm |
| --Mode 2 | 520,395 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 6,845 lbm |
| OMS & RCS Propellant | 15,693 lbm |
| Landing Propellant | 17,104 lbm |
| Landing Specific Impulse | 354.0 sec |
| Main Engine Type/No.: | Modular Plug Nozzle/1 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 1,713,908 lbf |
| Sea Level Isp (@100 % RPL): | 354.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 2,227,112 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 982,594 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |

**Figure 5.5-17 Side Entry Conical Vehicle Using Modular Plug Nozzle Engine and Oxygen/Hydrogen Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,388,459 lbm |
| **Length:** | 123 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 98,717 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 772,959 lbm |
| --Mode 2 | 456,583 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/C3H8 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 6,562 lbm |
| OMS & RCS Propellant | 13,693 lbm |
| Landing Propellant | 14,945 lbm |
| Landing Specific Impulse | 354.0 sec |
| Main Engine Type/No.: | Modular Plug Nozzle/1 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 1,666,151 lbf |
| Sea Level Isp (@100 % RPL): | 344.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 1,942,228 lbf |
| Vacuum Isp (@100 % RPL): | 401.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 861,700 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |

**Figure 5.5-18 Side Entry Conical Vehicle Using Modular Plug Nozzle Engine and Oxygen/Hydrogen/Propane Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,357,839 lbm |
| **Length:** | 121 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 94,928 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 760,826 lbm |
| --Mode 2 | 442,869 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 6,423 lbm |
| OMS & RCS Propellant | 13,276 lbm |
| Landing Propellant | 14,517 lbm |
| Landing Specific Impulse | 354.0 sec |
| Main Engine Type/No.: | Modular Plug Nozzle/1 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 1,629,406 lbf |
| Sea Level Isp (@100 % RPL): | 340.5 sec |
| Vacuum Thrust per Engine (@100% RPL) | 1,899,778 lbf |
| Vacuum Isp (@100 % RPL): | 397.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 835,818 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |

**Figure 5.5-19 Side Entry Conical Vehicle Using Modular Plug Nozzle Engine and Oxygen/Hydrogen/Kerosene Propellants Concept Summary**

## 5.6 Winged Body VTHL Concept

This launch vehicle configuration is a cylindrical winged body VTHL design with integral propellant tanks. The small delta wings and tip fins are designed for the landing of the launch vehicle and payload after the ascent propellant has burned off. Since this configuration is a cylinder with integral propellant tanks, it has a load path going from the main engines through the skin of the vehicle to the propellant tanks and payload. Therefore, the cases using the dual throat engines and the plug nozzle engines have their engines on the periphery of the launch vehicle base. These cases use smaller thrust structure coefficients (see Section 5.4).

Table 5.6-1 shows the sizing groundrules that are specific to this launch vehicle configuration.

Table 5.6-1 Winged Body Configuration Specific Sizing Groundrule

- Maximum normal acceleration is 2.5 Gs (sensitivity trade study should be performed)
- Payload bay is mounted transverse to the vehicle long axis
- Payload bay weight is 5,786 lbm (Option 3 vehicle payload bay mass and mass of the fairing over the payload bay and crew cabin)
- The nose cone is a biconic with hemispherical nose tip; dimensions are defined by the user
- This launch vehicle configuration has six engines

This launch vehicle configuration was based on the winged SSTO rocket configuration in the Option Three Access to Space Study. One reason for including a winged body vehicle configuration in this trade study was to facilitate a comparison of the dry masses for a launch vehicle configuration as calculated by the sizing tools against the dry mass of a similar launch vehicle configuration using more elaborate design tools that was documented in the Option Three Access to Space final report.

The two cases that were run for this vehicle configuration used the Evolved SSME and RD-701 engines. The dry masses of these vehicle configurations using these engines can be seen in Figure 5.6-1. As shown in the Option Three Access to Space final report, a vehicle configuration using the RD-701 engines is significantly lighter than a vehicle configuration using the Evolved SSME. Descriptions of the resulting vehicle configurations and mass properties using these engines are shown in Figures 5.6-2 and 5.6-3.

The sizing tools estimated a dry mass of 251,480 lbm (see Figure 5.6-2) for a winged body launch vehicle configuration using Evolved SSME engines and a graphite epoxy liquid hydrogen tank. The corresponding vehicle configuration in the Option Three Access to Space final report had a dry mass of 198,980 lbm.

The sizing tool estimated a dry mass of 162,145 lbm (see Figure 5.6-3) for a winged body launch vehicle configuration using RD-704 engines and a graphite epoxy liquid hydrogen tank. The corresponding vehicle configuration in the Option Three Access to Space final report had a dry mass of 130,218 lbm.

A detailed comparison was made between the vehicle dry masses in the Option Three Access to Space final report and the vehicle dry masses calculated by the sizing tools. The major discrepancy found was the thrust structure mass calculations. If the option three Access to Space final report values for the thrust structure masses were used in the sizing tools and the vehicle configuration was resized, the sizing tools calculated a dry mass of 139,046 lbm for a vehicle configuration using RD-701 engines and a dry mass of 212,084 lbm for a vehicle configuration using Advanced SSMEs.

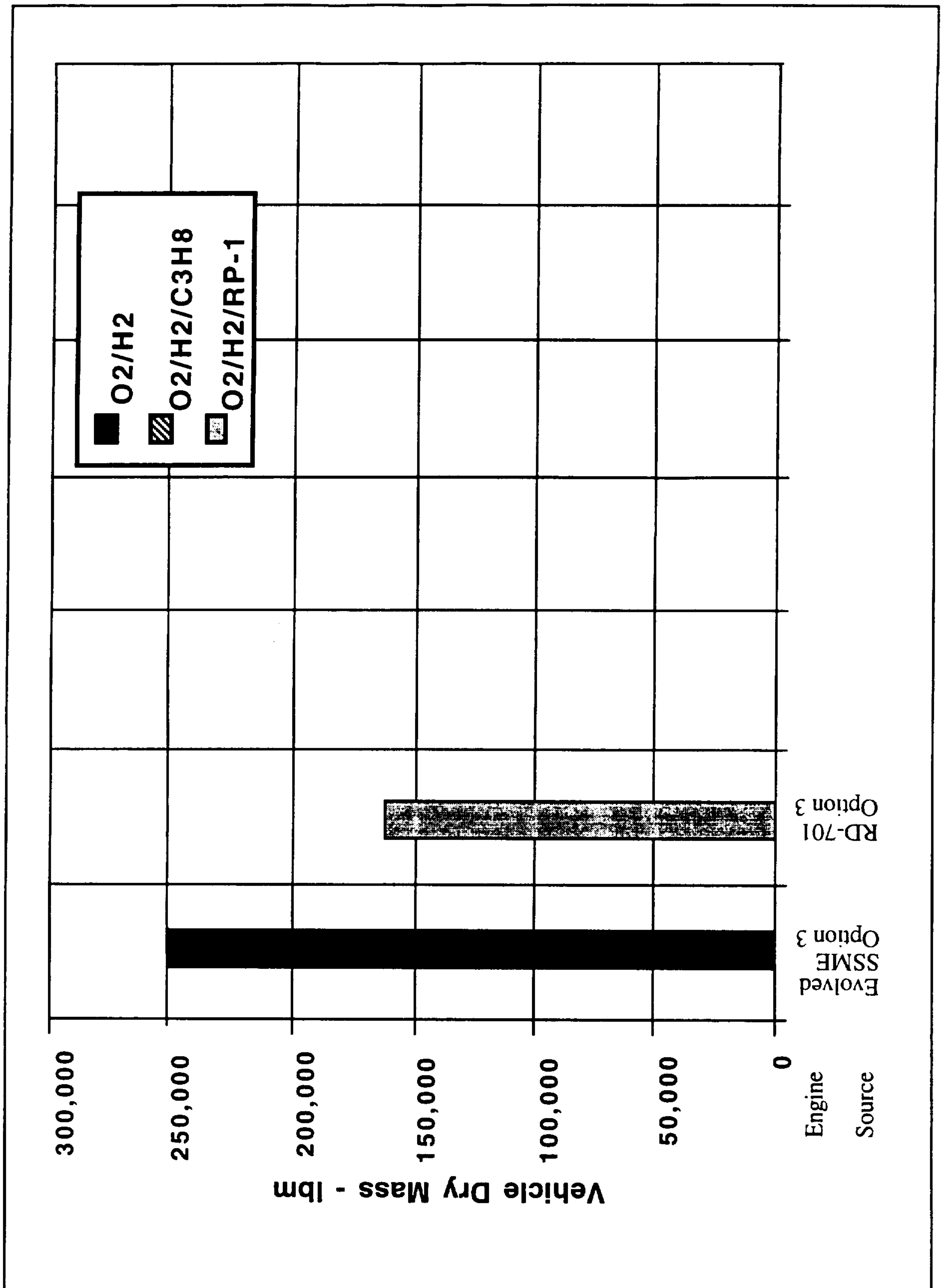



Figure 5.6-1 Vehicle Dry Masses

| | |
|---|---|
| **GLOW:** | 2,647,250 lbm |
| **Length:** | 179 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 251,480 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,382,943 lbm |
| --Mode 2 | 949,310 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 12,134 lbm |
| OMS & RCS Propellant | 26,383 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Evolved SSME/6 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 529,450 lbf |
| Sea Level Isp (@100 % RPL): | 390.4 sec |
| Vacuum Thrust per Engine (@100% RPL) | 606,616 lbf |
| Vacuum Isp (@100 % RPL): | 447.3 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 295,005 lbf |
| Vacuum Isp (@100 % RPL): | 447.3 sec |

**Figure 5.6-2 Winged Body Vehicle Using Evolved SSMEs Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,994,088 lbm |
| **Length:** | 151 ft |
| | |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 162,145 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,139,963 lbm |
| --Mode 2 | 639,688 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 9,434 lbm |
| OMS & RCS Propellant | 17,858 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | RD-701/6 |
| | |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 398,818 lbf |
| Sea Level Isp (@100 % RPL): | 333.5 sec |
| Vacuum Thrust per Engine (@100% RPL) | 460,524 lbf |
| Vacuum Isp (@100 % RPL): | 385.1 sec |
| | |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 199,296 lbf |
| Vacuum Isp (@100 % RPL): | 452.7 sec |

**Figure 5.6-3 Winged Body Vehicle Using RD-701 Engines Concept Summary**

The conclusions that were drawn from this comparison are that the sizing tools provide a conservative estimate of the launch vehicle configuration's dry mass and that the lighter thrust structure design used in the Option Three Access to Space Study would significantly reduce the size and mass of the launch vehicle configurations reported in this study.

## 5.7 Lifting Body VTHL Concept

This launch vehicle configuration is a lifting body VTHL design. The lifting body launch vehicle configuration has a smaller ballistic coefficient than the other configurations considered in this study. Therefore it encounters lower heat loads during reentry. The lifting body configuration has a lower wing loading and therefore a lower landing speed than the winged body configuration.

This launch vehicle configuration does not use integral propellant tanks. Therefore, the cases using the dual throat engines and the plug nozzle engines do not use smaller thrust structure coefficients (see Section 5.4).

Table 5.7-1 shows the sizing groundrules that are specific to this launch vehicle configuration.

**Table 5.7-1 Lifting Body Configuration Specific Sizing Groundrules**

| |
|---|
| • Maximum normal acceleration is 1.6 Gs (from unconstrained trajectory results) |
| • Payload bay is located parallel to the launch vehicle long axis |
| • Payload bay weight is 3,925 lbm (Option 3 vehicle payload bay mass ) |
| • Nose cap length is five feet |
| • Nose cap base is an ellipse: minor axis is five feet and major axis is eleven feet |
| • These launch vehicle configurations have five engines |

Figure 5.7-1 shows the layout of the propellant tanks, payload bay, and engine bay used by the lifting body launch vehicle configuration. The payload bay, the liquid oxygen propellant tank and the engine bay are mounted along the centerline of the launch vehicle configuration. The fuel is stored in the two outboard propellant tanks. The outboard fuel tanks are bent double cones. The bipropellant vehicle configurations have a total of three main propellant tanks, two liquid hydrogen tanks and a liquid oxygen tank. For the tripropellant launch vehicle configurations, the bent double cone fuel tanks are split into forward hydrocarbon fuel tanks and aft liquid hydrogen tanks. The tripropellant vehicle configurations have a total of five main propellant tanks, two hydrocarbon tanks, two liquid hydrogen tanks, and a liquid oxygen tank. Although it is not shown on this figure, there is space for a crew cabin and some of the vehicle subsystems forward of the payload bay.

The point of maximum fuel tank diameter is the dividing line between the forward and aft parts of the lifting body launch vehicle configuration body.

The lifting body launch vehicle configuration used in this study have three body parameters that have a large impact on the configuration mass and size. These parameters are the oxidizer tank radius, the forward fuel tank cone half angle, and the engine bay height.

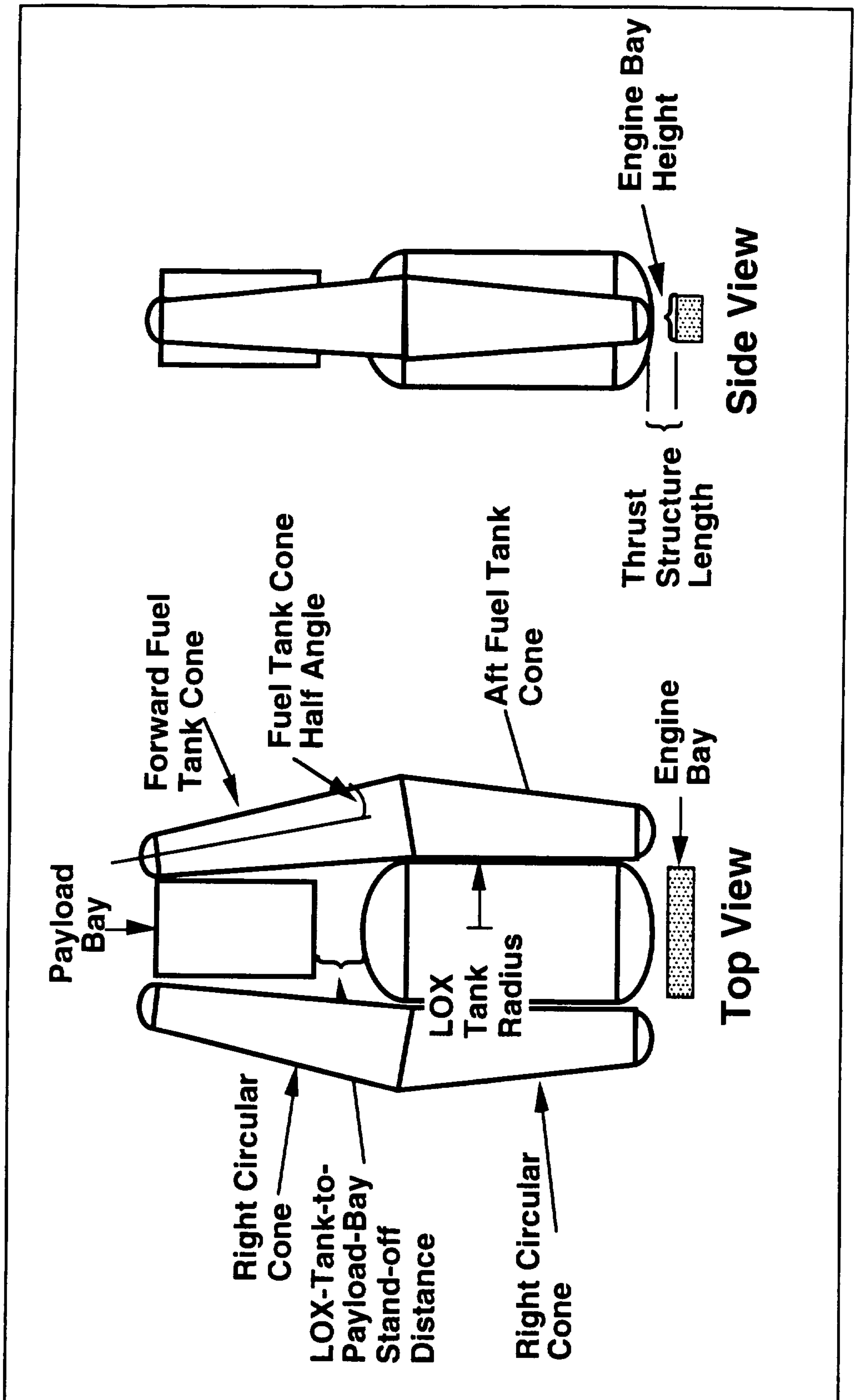



Figure 5.7-1 Lifting Body Configuration Propellant Tank, Payload Bay and Engine Bay Layout

Figure 5.7-2 shows the relationship between the oxidizer tank radius and the vehicle dry mass. As can be seen here, increasing the oxidizer tank diameter dramatically lowers the vehicle dry mass. Since the oxidizer tank volume is fixed, a larger oxidizer tank diameter reduces the oxidizer tank length. This reduces the aft lifting body surface area which lowers the mass of the vehicle skin and TPS.

Figure 5.7-3 shows the relationship between the forward fuel tank cone half angle and the vehicle dry mass. As can be seen here, decreasing the forward fuel tank cone half angle lowers the vehicle dry mass. Since the fuel tank volume is fixed, decreasing this half angle increases the forward fuel tank forward radius and reduces the forward fuel tank aft radius. This reduces the forward lifting body surface area which lowers the mass of the vehicle skin and TPS.

Figure 5.7-4 shows the relationship between the engine bay height and the vehicle dry mass. As can be seen here, increasing the engine bay height decreases the vehicle dry mass. Because of the aerodynamic affects, the lifting body configuration used in this study assumes a constant angle between the maximum fuel tank width and the engine bay height. Therefore increasing the engine bay height also increases the aft fuel cone aft radius and decreases the aft fuel tank forward radius. This reduces the aft lifting body surface area which lowers the mass of the vehicle skin and TPS.

If no constraints were imposed on the lifting body launch vehicle configuration body parameters, the minimum dry mass solution would result in a short, wide vehicle configuration with a small aft body length. Since the aerodynamics of this minimum dry mass solution was deemed unacceptable, it became necessary to impose constraints on the vehicle configuration body parameters to improve the launch vehicle configuration's aerodynamic characteristics. A set of constraints on the body launch vehicle body parameters were defined that resulted in acceptable aerodynamic characteristics. These constraints made the launch vehicle configuration heavier, longer, and slenderer. The lifting body launch vehicle configuration should have a trade study done to better define the relationship and tradeoffs between it's aerodynamic characteristics and dry mass.

Cases were run using the lifting body VTHL vehicle configuration and the engine and propellant combination shown in Table 5.3-1. The resulting vehicle dry masses are plotted in Figure 5.7-5. Descriptions of the resulting vehicle configuration mass properties and sizes for these cases are shown in Figures 5.7-6 through 5.7-20.

The first two cases in Figure 5.7-5 use the Evolved SSME and the RD-701 engines. As shown in the option three Access to Space final report, a vehicle configuration using the RD-701 engines is significantly lighter than a vehicle configuration using the Evolved SSME.

The next case uses the RD-704 engine. The dry mass of a vehicle configuration using the RD-704 engine is about half way between the dry mass of vehicle configurations using the Evolved SSME and the RD-701 engines. The reasons for the difference in the dry masses of vehicle configurations using the RD-701 engines and the RD-704 engines are the RD-704 engine is heavier and the extra hydrogen flow during the Mode 1 burn make the RD-704 vehicle configuration propellant tanks larger and therefore heavier.

The next three cases use the Rocketdyne SSTO study engines. A vehicle configuration using FFSCC engines are lighter than vehicle configurations using dual mixture ratio engines or expander cycle engines. The higher specific impulse of the FFSCC engines offset this engine cycle's higher weights.

The next nine cases used the three Aerojet engines with the three propellant combinations.

Vehicle configurations using the dual throat engine and the plug nozzle engines are lighter than vehicle configurations using the dual expansion cycle engines. The reason for this is thought to be the lighter thrust structures and the use of differential throttling for TVC used on the vehicle configurations using these engines. There was not enough time to check this hypothesis out.

Vehicle configurations using the plug nozzle cycle engines were lighter than vehicle configurations using dual throat cycle engines.

In these nine cases, the tripropellant hydrocarbon engine vehicle configurations were lighter than the bipropellant hydrogen engine vehicle configurations.

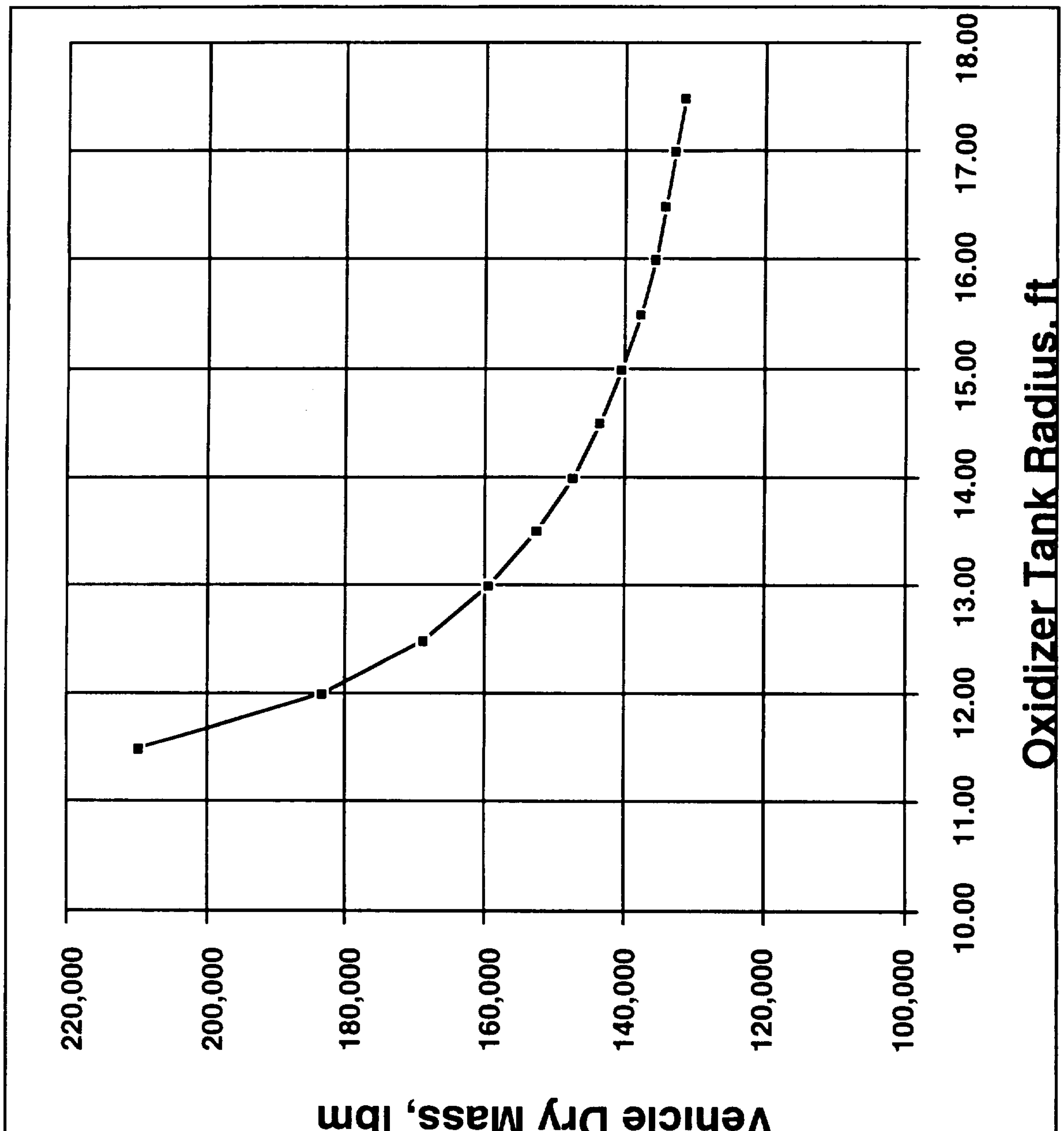



**Figure 5.7-2 Lifting Body Configuration Dry Mass as a Function of Oxidizer Tank Radius**

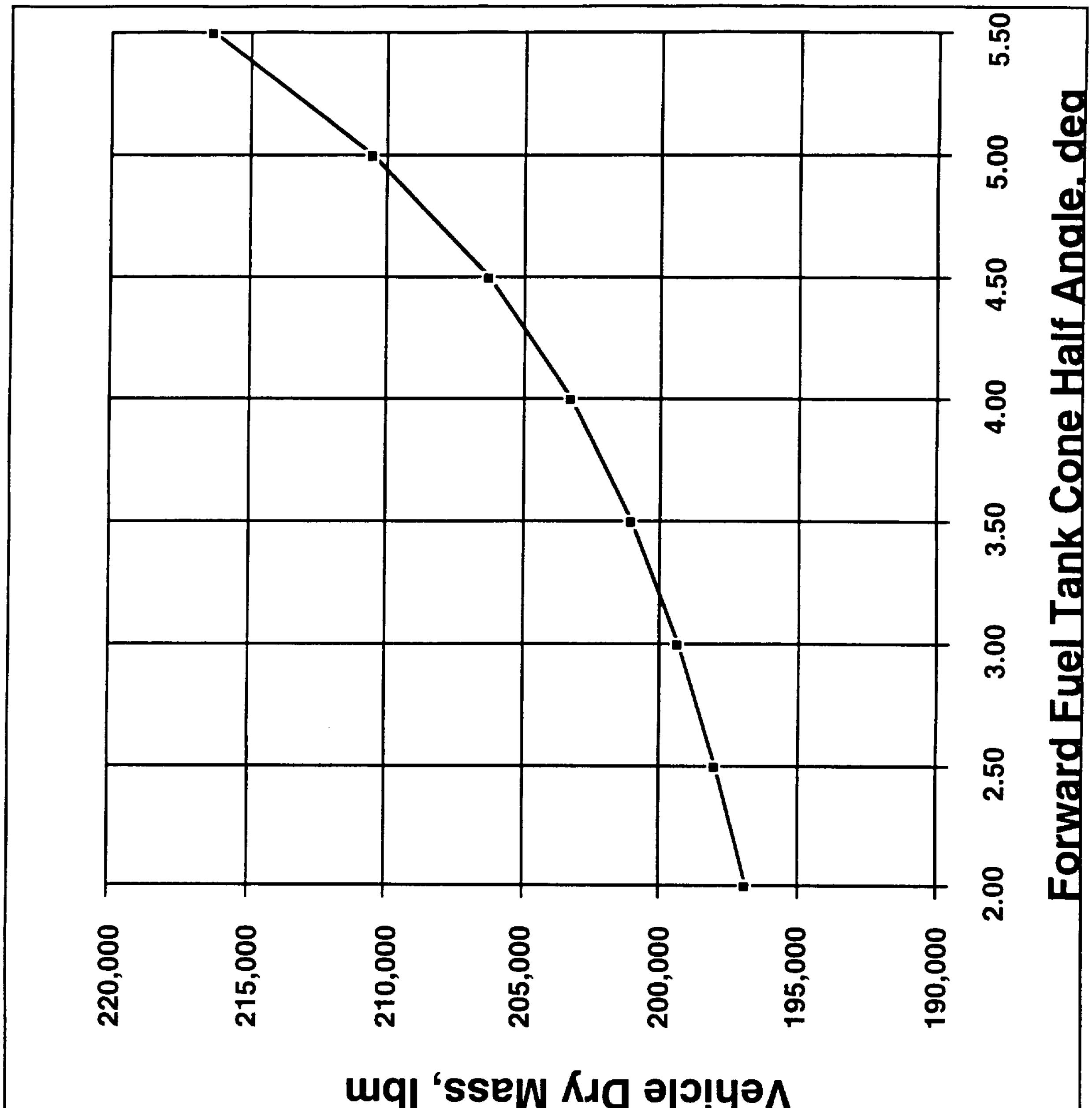



Figure 5.7-3 Lifting Body Configuration Dry Mass as a Function of Forward Fuel Tank Cone Half Angle

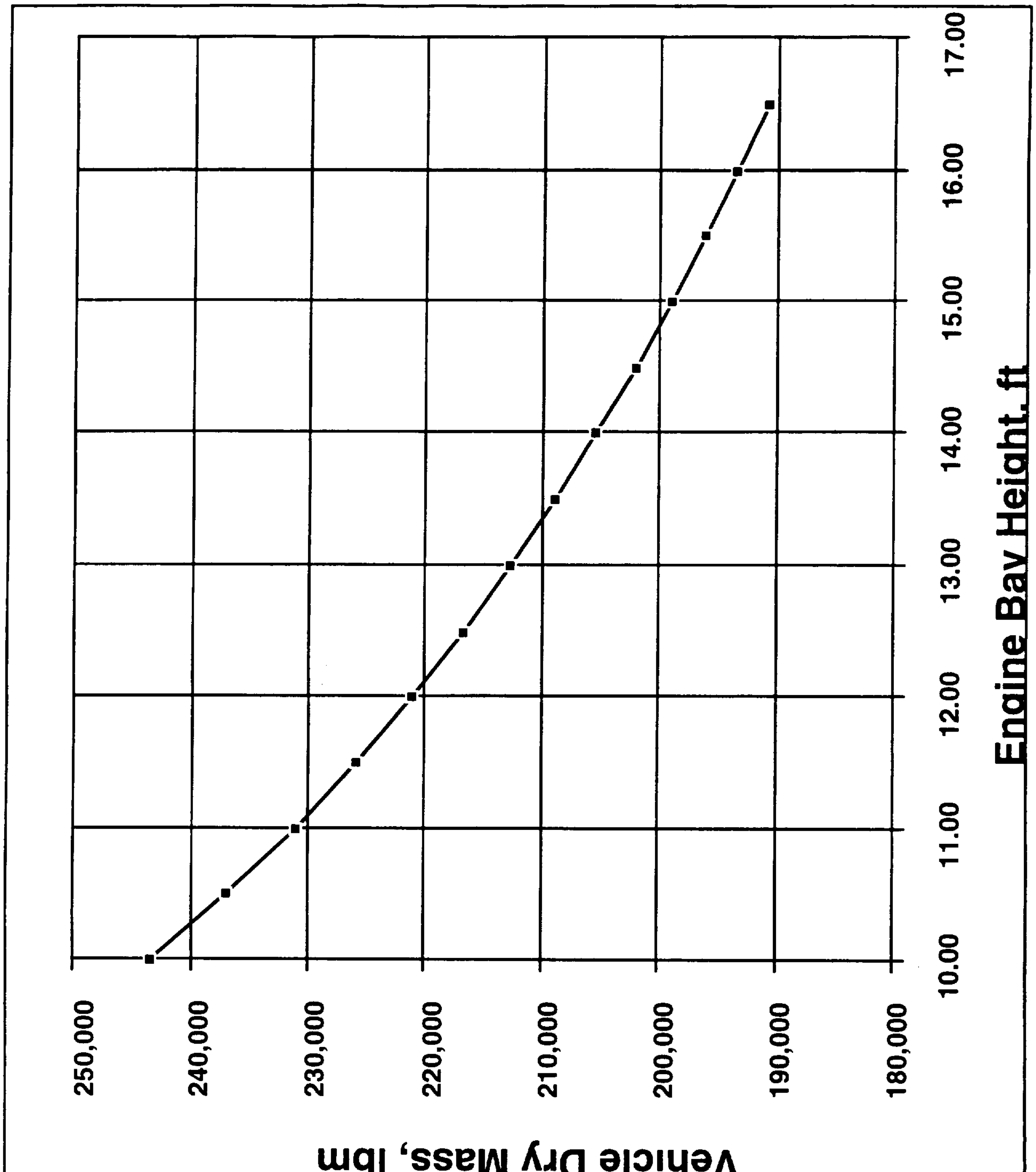



**Figure 5.7-4 Lifting Body Configuration Dry Mass as a Function of Engine Bay Height**

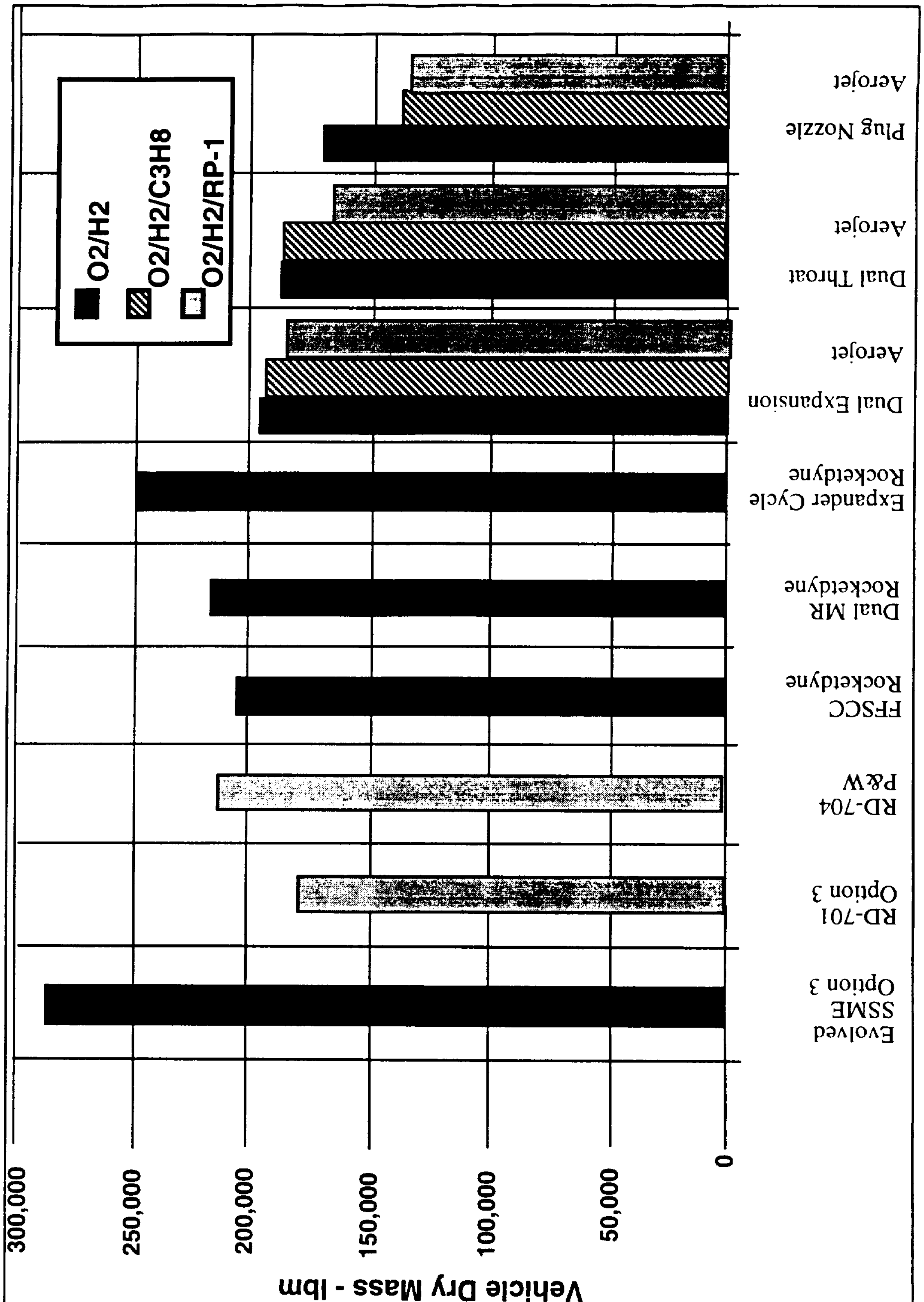



Figure 5.7-5 Lifting Body Vehicle Using Evolved SSMEs Concept Summary

| | |
|---|---|
| **GLOW:** | 2,919,677 lbm |
| **Length:** | 135 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 286,840 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,514,660 lbm |
| --Mode 2 | 1,050,064 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 13,356 lbm |
| OMS & RCS Propellant | 29,757 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Evolved SSME/5 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 700,723 lbf |
| Sea Level Isp (@100 % RPL): | 390.4 sec |
| Vacuum Thrust per Engine (@100% RPL) | 802,851 lbf |
| Vacuum Isp (@100 % RPL): | 447.3 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 393,405 lbf |
| Vacuum Isp (@100 % RPL): | 444.7 sec |

**Figure 5.7-6 Lifting Body Vehicle Using Evolved SSMEs Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,143,771 lbm |
| **Length:** | 131 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 180,737 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,224,895 lbm |
| --Mode 2 | 683,382 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 10,124 lbm |
| OMS & RCS Propellant | 19,632 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | RD-701/5 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 514,505 lbf |
| Sea Level Isp (@100 % RPL): | 333.5 sec |
| Vacuum Thrust per Engine (@100% RPL) | 594,111 lbf |
| Vacuum Isp (@100 % RPL): | 385.1 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 257,285 lbf |
| Vacuum Isp (@100 % RPL): | 452.7 sec |

**Figure 5.7-7 Lifting Body Vehicle Using RD-701 Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,388,519 lbm |
| **Length:** | 133 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 214,997 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,318,592 lbm |
| --Mode 2 | 795,922 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 11,106 lbm |
| OMS & RCS Propellant | 22,902 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | RD-704/5 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 573,245 lbf |
| Sea Level Isp (@100 % RPL): | 356.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 655,367 lbf |
| Vacuum Isp (@100 % RPL): | 407.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 299,580 lbf |
| Vacuum Isp (@100 % RPL): | 452.0 sec |

**Figure 5.7-8 Lifting Body Vehicle Using RD-704 Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,039,569 lbm |
| **Length:** | 131 ft |
| | |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 206,769 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,037,595 lbm |
| --Mode 2 | 738,814 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 9,276 lbm |
| OMS & RCS Propellant | 22,116 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | FFSCC/5 |
| | |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 489,497 lbf |
| Sea Level Isp (@100 % RPL): | 401.7 sec |
| Vacuum Thrust per Engine (@100% RPL) | 560,904 lbf |
| Vacuum Isp (@100 % RPL): | 460.3 sec |
| | |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 280,553 lbf |
| Vacuum Isp (@100 % RPL): | 460.3 sec |

**Figure 5.7-9 Lifting Body Vehicle Using Full Flow Staged Combustion Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,386,748 lbm |
| **Length:** | 133 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 219,040 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,308,669 lbm |
| --Mode 2 | 799,462 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 11,290 lbm |
| OMS & RCS Propellant | 23,287 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Dual Mixture Ratio/5 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 572,819 lbf |
| Sea Level Isp (@100 % RPL): | 343.9 sec |
| Vacuum Thrust per Engine (@100% RPL) | 683,752 lbf |
| Vacuum Isp (@100 % RPL): | 410.5 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 301,862 lbf |
| Vacuum Isp (@100 % RPL): | 455.5 sec |

**Figure 5.7-10 Lifting Body Vehicle Using Dual Mixture Ratio Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,604,058 lbm |
| **Length:** | 134 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 249,513 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,390,464 lbm |
| --Mode 2 | 900,739 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 12,147 lbm |
| OMS & RCS Propellant | 26,195 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Dual Expander Cycle/5 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 624,974 lbf |
| Sea Level Isp (@100 % RPL): | 367.5 sec |
| Vacuum Thrust per Engine (@100% RPL) | 756,431 lbf |
| Vacuum Isp (@100 % RPL): | 444.8 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 339,806 lbf |
| Vacuum Isp (@100 % RPL): | 444.8 sec |

**Figure 5.7-11 Lifting Body Vehicle Using Dual Expander Cycle Bell Engines Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,969,088 lbm |
| **Length:** | 131 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 199,104 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,018,482 lbm |
| --Mode 2 | 695,892 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 9,225 lbm |
| OMS & RCS Propellant | 21,385 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Dual Expansion Bell/5 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 472,581 lbf |
| Sea Level Isp (@100 % RPL): | 366.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 578,460 lbf |
| Vacuum Isp (@100 % RPL): | 448.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 266,170 lbf |
| Vacuum Isp (@100 % RPL): | 468.0 sec |

**Figure 5.7-12 Lifting Body Vehicle Using Dual Expanding Bell Engines and Oxygen/Hydrogen Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,304,262 lbm |
| **Length:** | 132 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 196,687 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,341,119 lbm |
| --Mode 2 | 709,446 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/C3H8 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 10,855 lbm |
| OMS & RCS Propellant | 21,154 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Dual Expansion Bell/5 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 553,023 lbf |
| Sea Level Isp (@100 % RPL): | 329.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 628,664 lbf |
| Vacuum Isp (@100 % RPL): | 374.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 269,680 lbf |
| Vacuum Isp (@100 % RPL): | 462.0 sec |

**Figure 5.7-13 Lifting Body Vehicle Using Dual Expanding Bell Engines and Oxygen/Hydrogen/Propane Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,218,593 lbm |
| **Length:** | 131 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 187,930 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,293,424 lbm |
| --Mode 2 | 681,474 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 10,446 lbm |
| OMS & RCS Propellant | 20,319 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Dual Expansion Bell/5 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 532,462 lbf |
| Sea Level Isp (@100 % RPL): | 329.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 603,673 lbf |
| Vacuum Isp (@100 % RPL): | 373.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 259,047 lbf |
| Vacuum Isp (@100 % RPL): | 462.0 sec |

**Figure 5.7-14 Lifting Body Vehicle Using Dual Expanding Bell Engines and Oxygen/Hydrogen/Kerosene Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,939,209 lbm |
| **Length:** | 129 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 188,976 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,012,237 lbm |
| --Mode 2 | 683,508 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 9,070 lbm |
| OMS & RCS Propellant | 20,418 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Modular Dual Throat/5 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 465,410 lbf |
| Sea Level Isp (@100 % RPL): | 366.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 562,053 lbf |
| Vacuum Isp (@100 % RPL): | 442.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 259,552 lbf |
| Vacuum Isp (@100 % RPL): | 461.0 sec |

**Figure 5.7-15 Lifting Body Vehicle Using Modular Dual Throat Engine and Oxygen/Hydrogen Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 2,202,834 lbm |
| **Length:** | 130 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 186,759 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,279,942 lbm |
| --Mode 2 | 680,500 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/C3H8 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 10,426 lbm |
| OMS & RCS Propellant | 20,207 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Modular Dual Throat/5 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 528,680 lbf |
| Sea Level Isp (@100 % RPL): | 326.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 608,144 lbf |
| Vacuum Isp (@100 % RPL): | 375.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 258,410 lbf |
| Vacuum Isp (@100 % RPL): | 461.0 sec |

**Figure 5.7-16 Lifting Body Vehicle Using Modular Dual Throat Engine and Oxygen/Hydrogen/Propane Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,962,870 lbm |
| **Length:** | 129 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 167,983 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 1,146,262 lbm |
| --Mode 2 | 595,955 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 9,255 lbm |
| OMS & RCS Propellant | 18,415 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Modular Dual Throat/5 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 471,089 lbf |
| Sea Level Isp (@100 % RPL): | 325.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 539,215 lbf |
| Vacuum Isp (@100 % RPL): | 372.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 228,650 lbf |
| Vacuum Isp (@100 % RPL): | 471.0 sec |

**Figure 5.7-17 Lifting Body Vehicle Using Modular Dual Throat Engine and Oxygen/Hydrogen/Kerosene Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,741,133 lbm |
| **Length:** | 129 ft |
| | |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 172,203 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 885,011 lbm |
| --Mode 2 | 631,768 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 8,333 lbm |
| OMS & RCS Propellant | 18,818 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Modular Plug/5 |
| | |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 417,872 lbf |
| Sea Level Isp (@100 % RPL): | 354.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 542,998 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |
| | |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 239,714 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |

**Figure 5.7-18 Lifting Body Vehicle Using Modular Plug Nozzle Engine and Oxygen/Hydrogen Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,606,145 lbm |
| **Length:** | 128 ft |
| **<u>Vehicle Specifications</u>:** | |
| Vehicle Dry Mass @ Liftoff: | 138,285 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 894,191 lbm |
| --Mode 2 | 525,506 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/C3H8 |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 7,581 lbm |
| OMS & RCS Propellant | 15,581 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Modular Plug/5 |
| **<u>Mode 1 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | 385,475 lbf |
| Sea Level Isp (@100 % RPL): | 344.0 sec |
| Vacuum Thrust per Engine (@100% RPL) | 449,347 lbf |
| Vacuum Isp (@100 % RPL): | 401.0 sec |
| **<u>Mode 2 Propulsion Specifications</u>:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 199,347 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |

**Figure 5.7-19 Lifting Body Vehicle Using Modular Plug Nozzle Engine and Oxygen/Hydrogen/Propane Propellants Concept Summary**

| | |
|---|---|
| **GLOW:** | 1,580,604 lbm |
| **Length:** | 127 ft |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 134,314 lbm |
| Usable Propellant Mass (including FPR): | |
| --Mode 1 | 885,692 lbm |
| --Mode 2 | 512,927 lbm |
| Propellant Combination: | |
| --Mode 1 | LOX/LH2/Kerosene |
| --Mode 2 | LOX/LH2 |
| Ascent Residuals | 7,468 lbm |
| OMS & RCS Propellant | 15,202 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | Modular Plug/5 |
| **Mode 1 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | 379,345 lbf |
| Sea Level Isp (@100 % RPL): | 340.5 sec |
| Vacuum Thrust per Engine (@100% RPL) | 442,291 lbf |
| Vacuum Isp (@100 % RPL): | 397.0 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@100% RPL) | N/A |
| Sea Level Isp (@100 % RPL): | N/A |
| Vacuum Thrust per Engine (@100% RPL) | 194,575 lbf |
| Vacuum Isp (@100 % RPL): | 460.0 sec |

**Figure 5.7-20 Lifting Body Vehicle Using Modular Plug Nozzle Engine and Oxygen/Hydrogen/Kerosene Propellants Concept Summary**

## 5.8 Study Conclusions

Based on a comparison of the dry masses of vehicle configurations using the Option Three Access to Space Study versions of evolved SSMEs and RD-701 engines and the groundrules used in this study, the winged body launch vehicle configuration had the lowest dry mass. The lifting body launch vehicle configuration had the largest dry mass.

The side entry cone launch vehicle configuration becomes the lowest dry mass vehicle configuration if the rotation/landing maneuver velocity requirement drops below 900 ft/sec.

The lifting body launch vehicle configuration has the most flexibility in it's body parameters and has the potential for a much larger dry mass reduction than the other vehicle configurations.

The winged body and side entry cone launch vehicle configurations in this study used payload bays transverse to the vehicle configuration's long axis. The reason for this decision was the winged body configuration in the Option Three Access to Space Study used this payload bay orientation. The lifting body launch vehicle configuration in this study used a payload bay oriented along the vehicle configuration's long axis. If the winged body and side entry cone configurations were required to have payload bays oriented along the vehicle configuration's long axis, their size and dry mass would increase. Since the lifting body configuration already has this payload bay orientation, its relative ranking would be improved.

The version of the evolved SSMEs used in Option Three Access to Space Study did not result in satisfactory SSTO launch vehicle configurations.

The lightest vehicle configurations used plug nozzle engines.

The payoff of using tripropellant engines is a function of both the engine type used and the vehicle configuration the engine is used on.

The reduction in vehicle configuration dry mass from the use high thrust-to-weight and high performance engines is a function of the vehicle configuration used.

# 6.0 SSTO Simulation Results

The candidate SSTO vehicle configurations that were defined and sized on TA-2, as described in Section 5, were simulated in three-degrees-of-freedom (3-DOF) nominal ascent trajectories in order to verify that the vehicle concepts were sized sufficiently to meet the Access to Space Option 3 team's payload delivery goal of 25 Klbm to an International Space Station Alpha orbit. Nominal entry trajectories were also simulated to verify crossrange capability and stagnation heating conditions. An innovative approach-and-landing trajectory simulation was also developed for the candidate VTOL concept in order to verify the propellant required to perform the vertical landing maneuver. An LMMS-modified version of the Simulation and Optimization of Rocket Trajectories (SORT) 3-DOF simulation tool was used to perform all of the trajectory analyses. LMMS' version of SORT was licensed to use the NPSOL optimizer that was developed by Stanford University, and was specially modified prior to the TA-2 contract for the assessment of advanced transportation system concepts. LMMS' SORT was derived from the standard version of SORT which is publicly available through NASA's Computer Software Management and Information Center (COSMIC) at the University of Georgia. Previous versions of SORT have been baselined for use in performing nominal and abort flight design for the Space Shuttle Program.

## 6.1 Simulation Groundrules and Assumptions

As was mentioned in Section 5, the Access to Space Option 3 Team's groundrules and assumptions were used for TA-2's SSTO sizing efforts. Similar groundrules and assumptions were used from Option 3 for TA-2's SSTO simulation and performance analyses. Figure 6.1-1 summarizes the groundrules that were used for the ascent trajectory simulations.

Ascent Groundrules

- KSC Launch, KSC atmosphere and winds
- 50 x 100 nautical mile MECO, 51.6° orbital inclination
- Maximum acceleration of 3 Gs
- 1% Delta-V reserves for flight performance reserves
- 900 psf maximum dynamic pressure
- 3500 psf*degree maximum dynamic pressure*alpha
- Pitch rate optimization for endo and exoatmospheric phases
- Continuous throttle for acceleration limiting
- Hohmann transfer post-MECO for final circularization
- Configuration specific forebody and power-on base aerodynamics generated for the lifting-body and conical configurations
- LaRC aerodynamic coefficients utilized for winged-body configuration

**Figure 6.1-1 Ascent Trajectory Groundrules**

Figure 6.1-2 summarizes the groundrules that were used for the entry trajectory simulations.

Entry Groundrules

- 1962 U.S. Standard atmosphere (non-site-specific)
- Maximum lift trajectory (minimum heat rate) transitioning into maximum range trajectory
- Bank angle and angle of attack used for heat rate modulation for the winged-body & lifting body configurations and conical configuration, respectively
- Bank angle and sideslip angle used for crossrange capability for the winged-body & lifting body configuration and conical configurations, respectively
- Configuration specific forebody and base aerodynamics generated for the lifting-body and conical configurations
- LaRC aerodynamic coefficients utilized for winged-body configuration

**Figure 6.1-2 Entry Trajectory Groundrules**

Figure 6.1-3 summarizes the groundrules that were used to simulate the approach-and-landing "pull-up" maneuver for the VTOL concept assessments. The purpose of the pull-up maneuver was to null out the VTOL vehicle's horizontal velocity component prior to performing a controlled rate-of-descent landing in the vertical plane. Prior to the initiation of the pull-up maneuver, the main propulsion system were activated and any requisite pre-chill performed. The main engines were ignited at a geodetic altitude of 1,000 feet and the engines were throttled up at a level that would allow a constant 2Gs normal acceleration pull-up flight path arc, with no more than 500 feet of altitude having been lost from the point of engine ignition to the point that a positive rate-of-climb (or positive h-dot) is achieved. The load factor during the pull-up was limited to 2Gs as a conservative limit, consistent with the 2G limit that is observed by the Shuttle Orbiter during entry and landing. The pull-up arc was flown, utilizing the main propulsion for total control authority, until a 90 degree flight path angle was achieved, and the main engines were throttled to null the vertical velocity component and initiate a controlled rate-of-descent landing. The use of aerodynamic control surface deflections to augment control authority during the pull-up maneuver was not investigated, due to time and budget constraints, therefore the worst-case scenario was assessed for identifying landing propellant requirements.

- **Conical landing maneuver initiated at 1,000 ft altitude with a 500 ft minimum droop altitude**
- **2 G constant normal load during propulsive pull-up maneuver to 90 degree flight path angle**
- **Strive for minimum H-dot value and minimum safe altitude at end of pull-up maneuver to minimize propellant requirements**

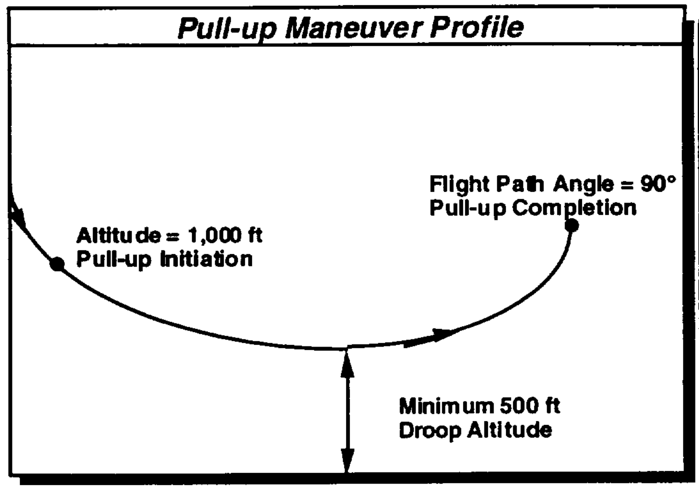



Figure 6.1-3 VTOL Approach-and-Landing Groundrules

Prior to the ascent and entry trajectory simulations being executed, longitudinal forebody and power-on/power-off base aerodynamic data were generated by LMMS as a function of angle of attack and Mach number, for the VTOL and lifting body configurations. The aerodynamic data were computed from a synthesis of existing wind tunnel and flight test data for vehicles having comparable outer moldline shapes. Longitudinal aerodynamic data generated by LaRC for NASA's wing-body concept were used for TA-2's wing-body configuration. Sideslip was presumed to be controlled to maintain a zero value, since the no-wind case was being assumed. Protuberance effects were not modeled for this first-order performance assessment. Point-mass mass properties were used, ignoring the effect of thrust vector losses due to a time-varying static margin.

## 6.2 Simulation Results

Figures 6.2-1 through 3 illustrate the vehicle specifications and injected payload mass that were achieved in the SORT ascent simulations for the tri-propellant VTOL, Wing-Body, and Lifting Body SSTO configurations that were defined by TA-2, respectively. The tri-propellant SSTO configurations are shown to illustrate an apples-to-apples comparison across the three configuration types, since the NASA wing-body SSTO concept (and thus that of TA-2) was only designed to use the NPO Energomash RD-704 tri-propellant main engine. A comparison of the various bi-propellant VTOL and Lifting Body concepts defined on TA-2 may be found in Section 5 of this volume.

| | |
|---|---|
| **Payload:** | 24,800 lbm |
| **Final Position:** | 220 nm Cic. Orbit, i= 51.6 deg |

| | |
|---|---|
| **GLOW:** | 1,815,965 lbm |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 134,075 lbm |
| Usable Propellant Mass (including FPR): | |
| — Mode 1 | 1,037,840 lbm |
| — Mode 2 | 581,558 lbm |
| Propellant Combination: | |
| — Mode 1 | LOX/LH2/Kerosene |
| — Mode 2 | LOX/LH2 |
| Ascent Residuals | 8,587 lbm |
| OMS & RCS Propellant | 16,375 lbm |
| Landing Propellant | 12,530 lbm |
| Landing Specific Impulse | 333.5 sec |
| Main Engine Type/No.: | RD-701A/7 |
| **Mode 1 Propellant Specifications:** | |
| Sea Level Thrust per Engine (@ 100 % RPL) | 311,308 lbf |
| Sea Level Isp (@ 100 % RPL): | 333.5 sec |
| Vacuum Thrust per Engine (@ 100 % RPL) | 359,475 lbf |
| Vacuum Isp (@100 % RPL): | 385.1 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@ 100 % RPL) | N/A |
| Sea Level Isp (@ 100 % RPL): | N/A |
| Vacuum Thrust per Engine (@ 100 % RPL) | 151,178 lbf |
| Vacuum Isp (@100 % RPL): | 452.7 sec |

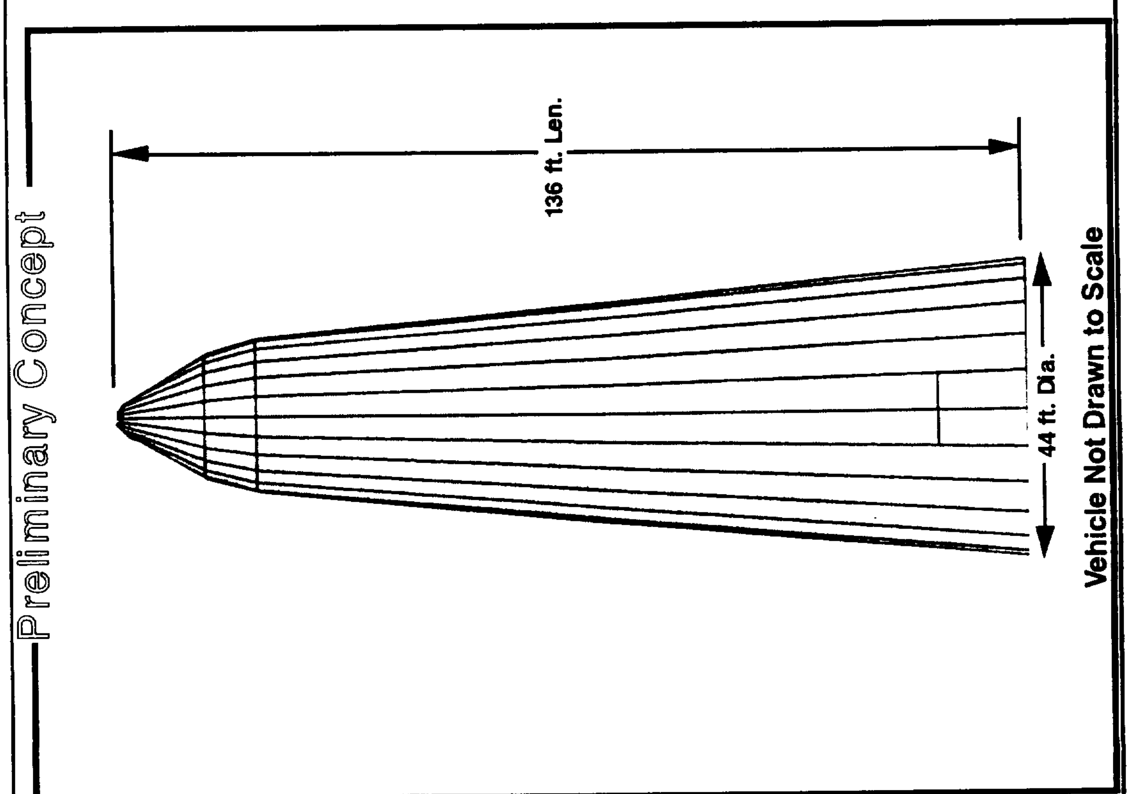
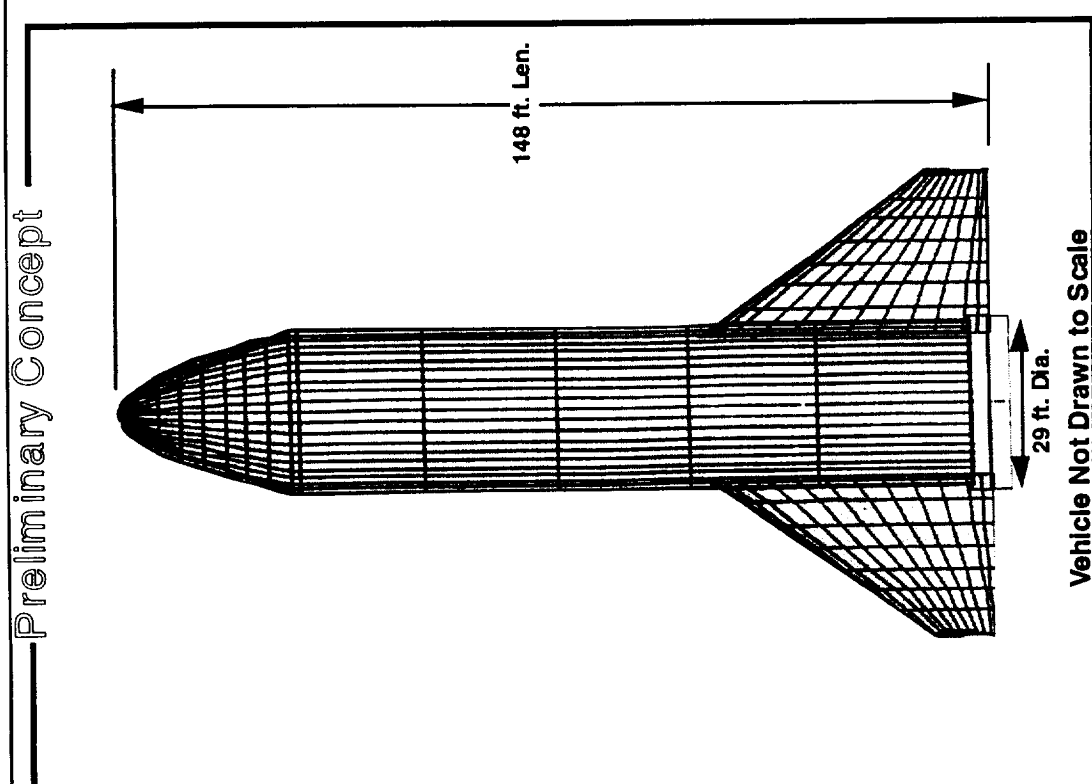



Figure 6.2-1 VTOL Configuration Summary

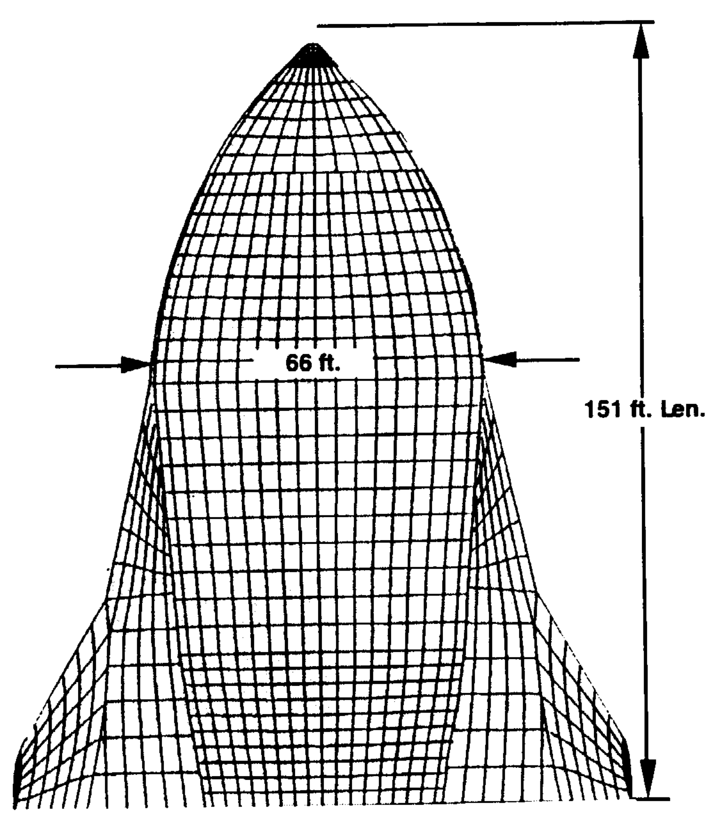
Preliminary Concept




| | |
|---|---|
| **Payload:** | 25,300 lbm |
| **Final Position:** | 220 nm Cic. Orbit, I= 51.6 deg |

| | |
|---|---|
| **GLOW:** | 1,857,714 lbm |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 148,884 lbm |
| Usable Propellant Mass (including FPR): | |
| — Mode 1 | 1,061,829 lbm |
| — Mode 2 | 596,618 lbm |
| Propellant Combination: | |
| — Mode 1 | LOX/LH2/Kerosene |
| — Mode 2 | LOX/LH2 |
| Ascent Residuals | 8,790 lbm |
| OMS & RCS Propellant | 16,593 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | RD-701A/6 |
| **Mode 1 Propellant Specifications:** | |
| Sea Level Thrust per Engine (@ 100 % RPL) | 371,543 lbf |
| Sea Level Isp (@ 100 % RPL): | 333.5 sec |
| Vacuum Thrust per Engine (@ 100 % RPL) | 429,029 lbf |
| Vacuum Isp (@100 % RPL): | 385.1 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@ 100 % RPL) | N/A |
| Sea Level Isp (@ 100 % RPL): | N/A |
| Vacuum Thrust per Engine (@ 100 % RPL) | 180,401 lbf |
| Vacuum Isp (@100 % RPL): | 452.7 sec |

Figure 6.2-2 Wing-Body VTHL Configuration Summary




| | |
|---|---|
| **Payload:** | 25,200 lbm |
| **Final Position:** | 220 nm Cic. Orbit, i= 51.6 deg |

| | |
|---|---|
| **GLOW:** | 2,439,123 lbm |
| **Vehicle Specifications:** | |
| Vehicle Dry Mass @ Liftoff: | 209,569 lbm |
| Usable Propellant Mass (including FPR): | |
| — Mode 1 | 1,394,274 lbm |
| — Mode 2 | 776,386 lbm |
| Propellant Combination: | |
| — Mode 1 | LOX/LH2/Kerosene |
| — Mode 2 | LOX/LH2 |
| Ascent Residuals | 11,511 lbm |
| OMS & RCS Propellant | 22,384 lbm |
| Landing Propellant | N/A |
| Landing Specific Impulse | N/A |
| Main Engine Type/No.: | RD-701A/5 |
| **Mode 1 Propellant Specifications:** | |
| Sea Level Thrust per Engine (@ 100 % RPL) | 585,390 lbf |
| Sea Level Isp (@ 100 % RPL): | 333.5 sec |
| Vacuum Thrust per Engine (@ 100 % RPL) | 675,963 lbf |
| Vacuum Isp (@100 % RPL): | 385.1 sec |
| **Mode 2 Propulsion Specifications:** | |
| Sea Level Thrust per Engine (@ 100 % RPL) | N/A |
| Sea Level Isp (@ 100 % RPL): | N/A |
| Vacuum Thrust per Engine (@ 100 % RPL) | 284,144 lbf |
| Vacuum Isp (@100 % RPL): | 452.7 sec |

Figure 6.2-3 Lifting Body VTHL Configuration Summary

## Ascent

Figures 6.2-4 through 8 illustrate the resulting ascent trajectory time histories for the three vehicle types for geodetic altitude, thrust-acceleration, dynamic pressure, product of dynamic pressure and angle of attack (also known as q-alpha), and angle of attack, respectively. All three vehicle types were shown to exhibit similar ascent trajectory profiles due to similar assumptions being used for vehicle sizing, thrust-to-weight ratios, and main engine performance. The lifting body configuration was able to tolerate a higher ascent dynamic pressure profile, for the same q-alpha load constraint, because of its favorable lift-to-drag ratio at low angles of attack, thereby flying a lifting trajectory. Further trajectory shaping could have been performed, if time had allowed, to further smooth the q-alpha profiles for all three vehicle configurations.

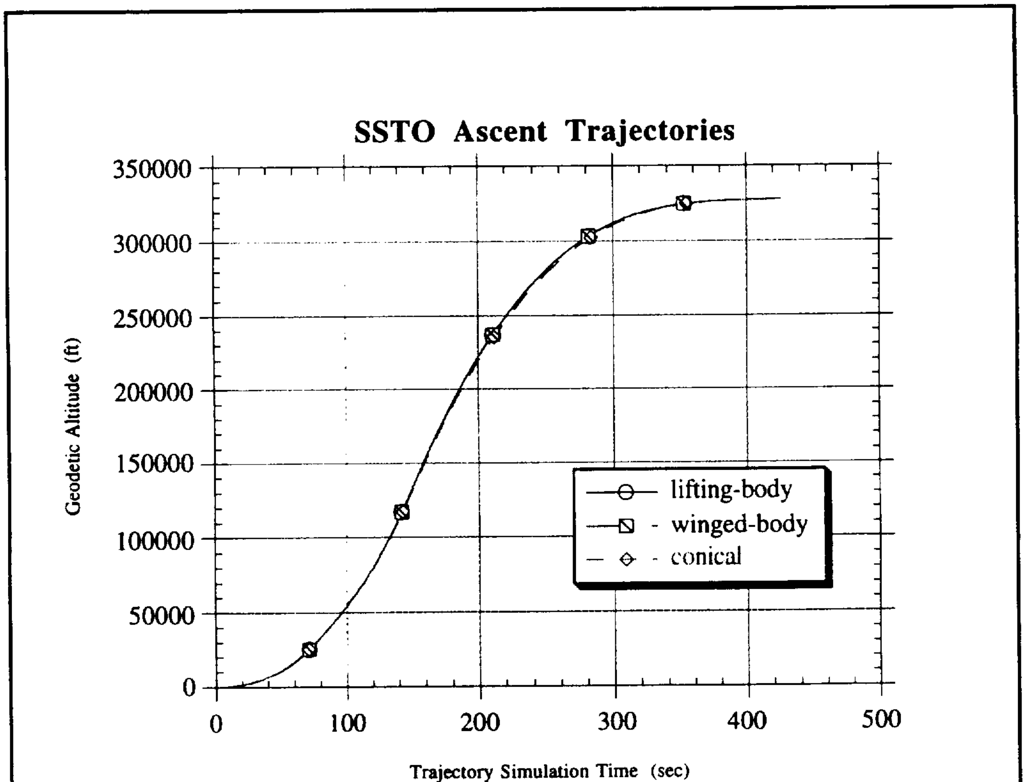



**Figure 6.2-4 Ascent Trajectory Geodetic Altitude Time History**

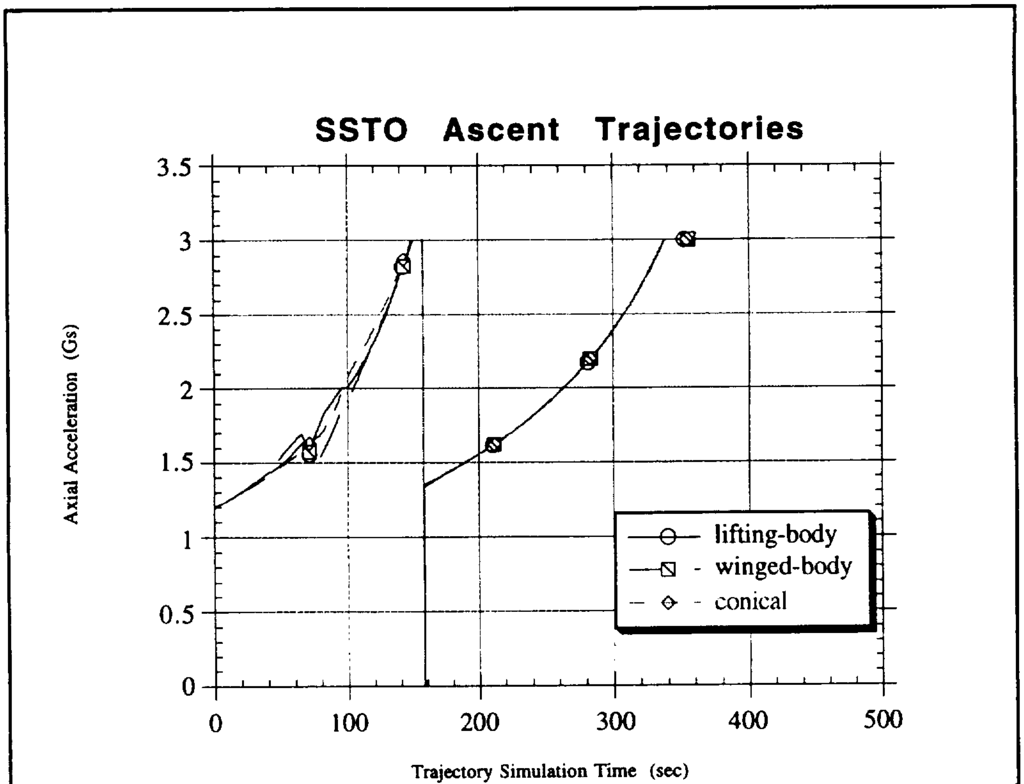



**Figure 6.2-5 Ascent Trajectory Thrust-Acceleration Time History**

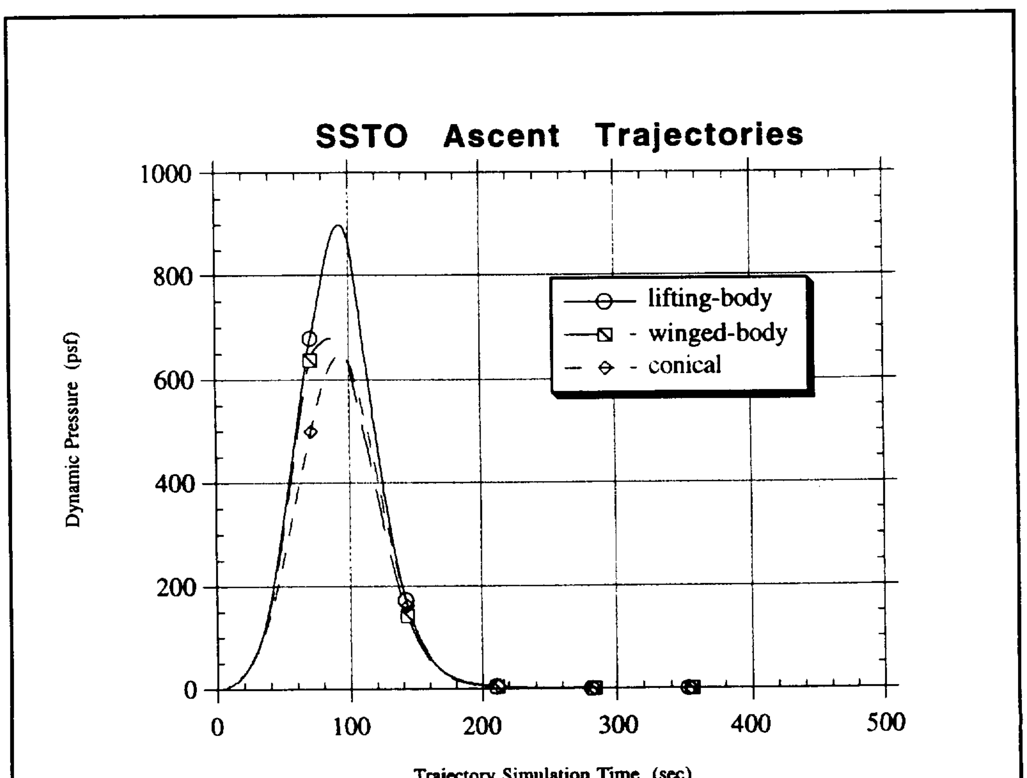



**Figure 6.2-6 Ascent Trajectory Dynamic Pressure Time History**

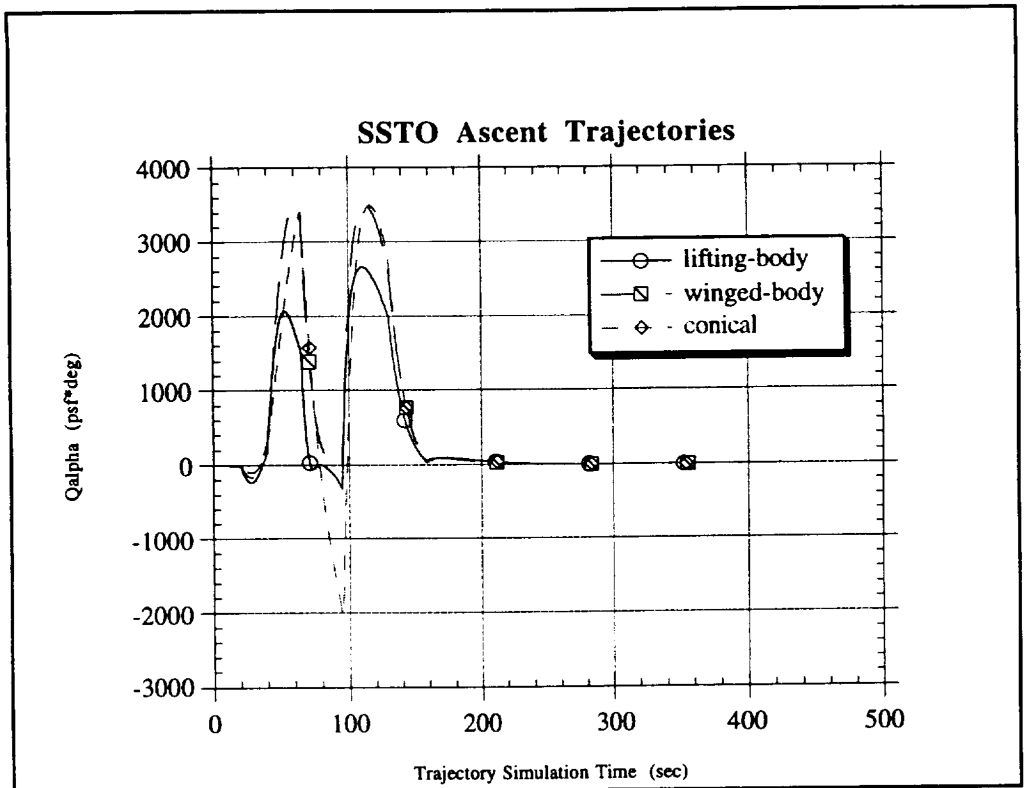



**Figure 6.2-7 Ascent Trajectory Q-Alpha Time History**

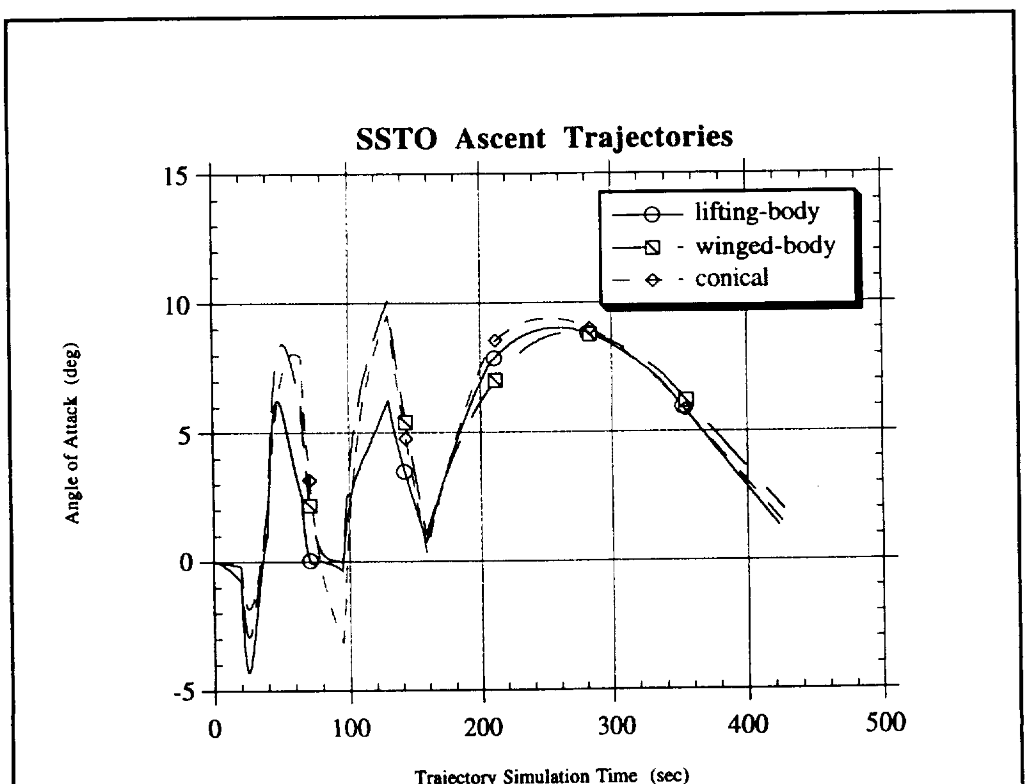



**Figure 6.2-8 Ascent Trajectory Angle-of-Attack Time History**

## Entry

Figures 6.2-9 through 13 illustrate the resulting entry trajectory time histories for the three vehicle types for geodetic altitude, stagnation heat rate, crossrange, normal load factor, and Mach number, respectively. During the entry performance assessments, a conscious decision was made to not treat crossrange capability as a leading performance metric, since all three vehicles were found to be capable of generating at a minimum, the roughly 700 nm capability needed for an abort-once-around scenario. With crossrange being held as an equal across the three configurations, the primary entry metrics became stagnation heat rate (which drives thermal protection system technology) and total heat load (which drives the technologies for thermal protection system, internal structure, and main propellant tank insulation). As seen from Figures 6.2-9 and 10, the aerodynamics associated with the lifting body allowed it to descend much more quickly into the thicker part of the atmosphere at maximum stagnation heating rates similar in value to the VTOL and wing-body configurations, which, when integrating the area under the heat rate curves to compute total stagnation heat load, results in the lifting body having significantly lower total heat load (and resulting heat soak) on the vehicle thermal protection system and internal structural elements; thus requiring less internal insulation. Figure 6.2-13 illustrates clearly the more rapid deceleration rate of the lifting body versus the other two configurations, over the same period of time, while maintaining similar maximum normal load factors as evidenced in Figure 6.2-12. Thus, the lifting body appears to exhibit better overall entry characteristics, which factor more favorably into the vehicle design.

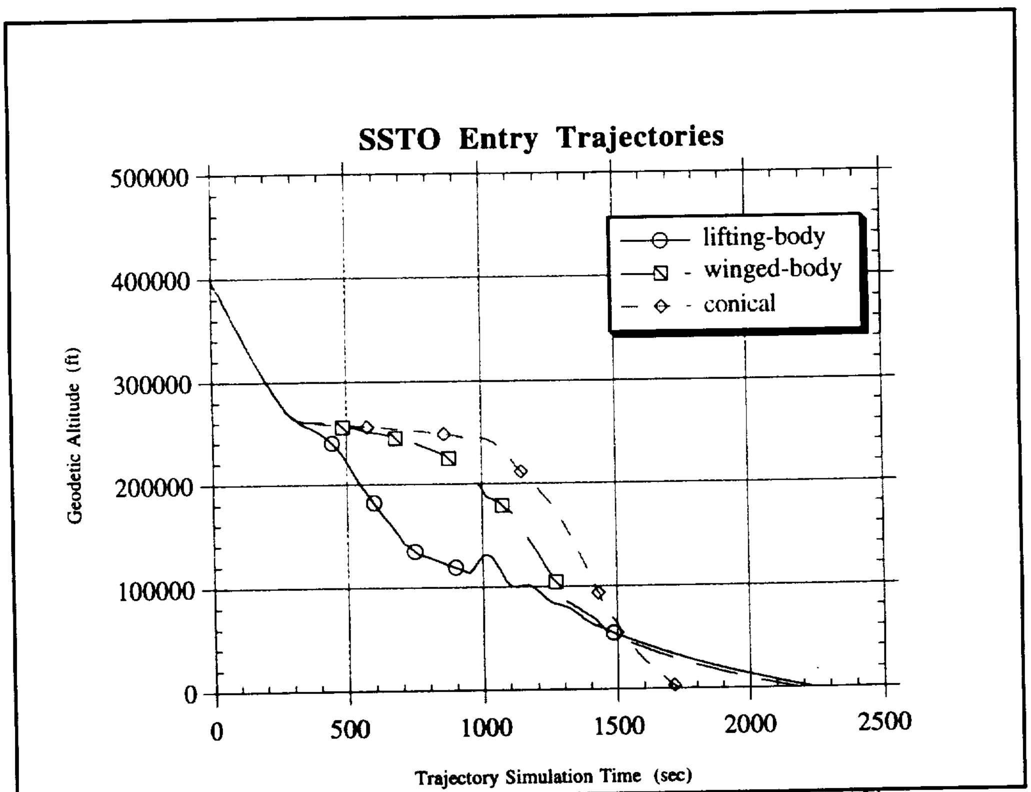



Figure 6.2-9 Entry Trajectory Geodetic Altitude Time History

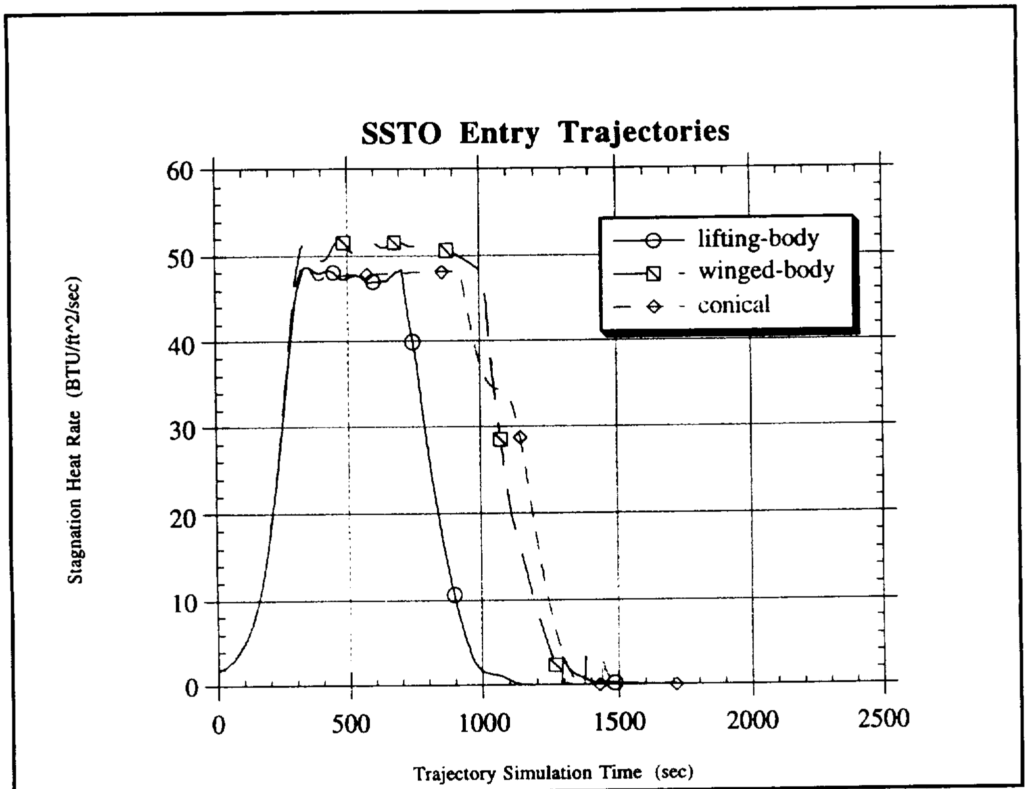



**Figure 6.2-10 Entry Trajectory Stagnation Heat Rate Time History**

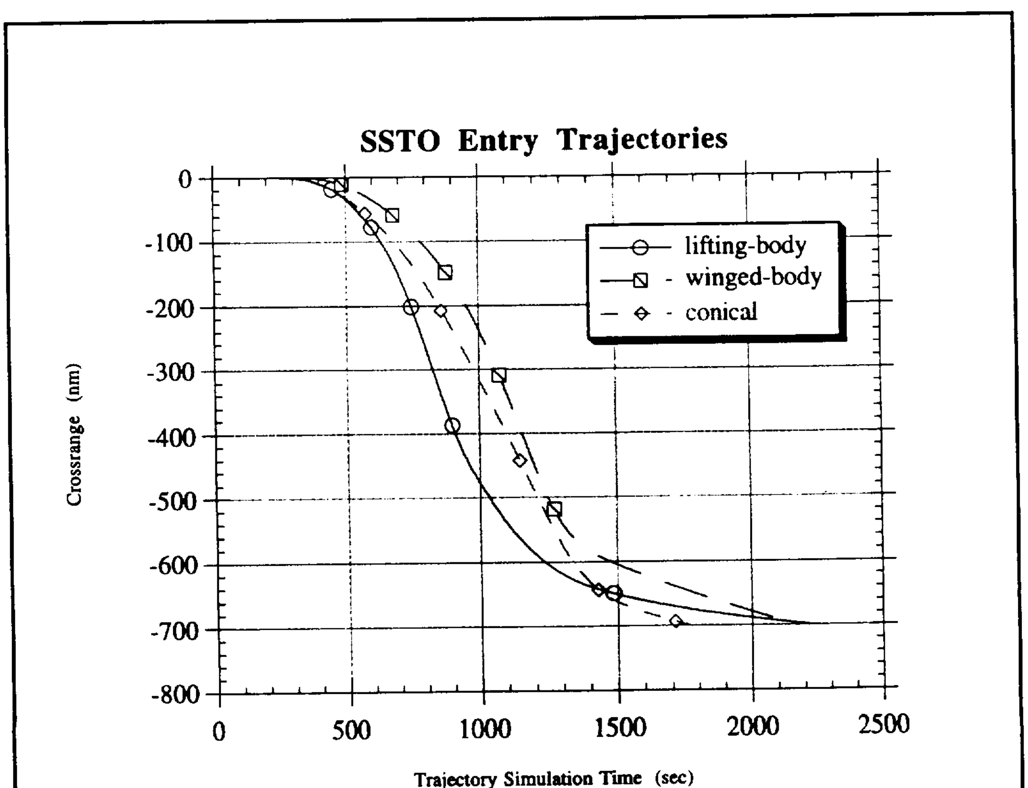



**Figure 6.2-11 Entry Trajectory Crossrange Time History**

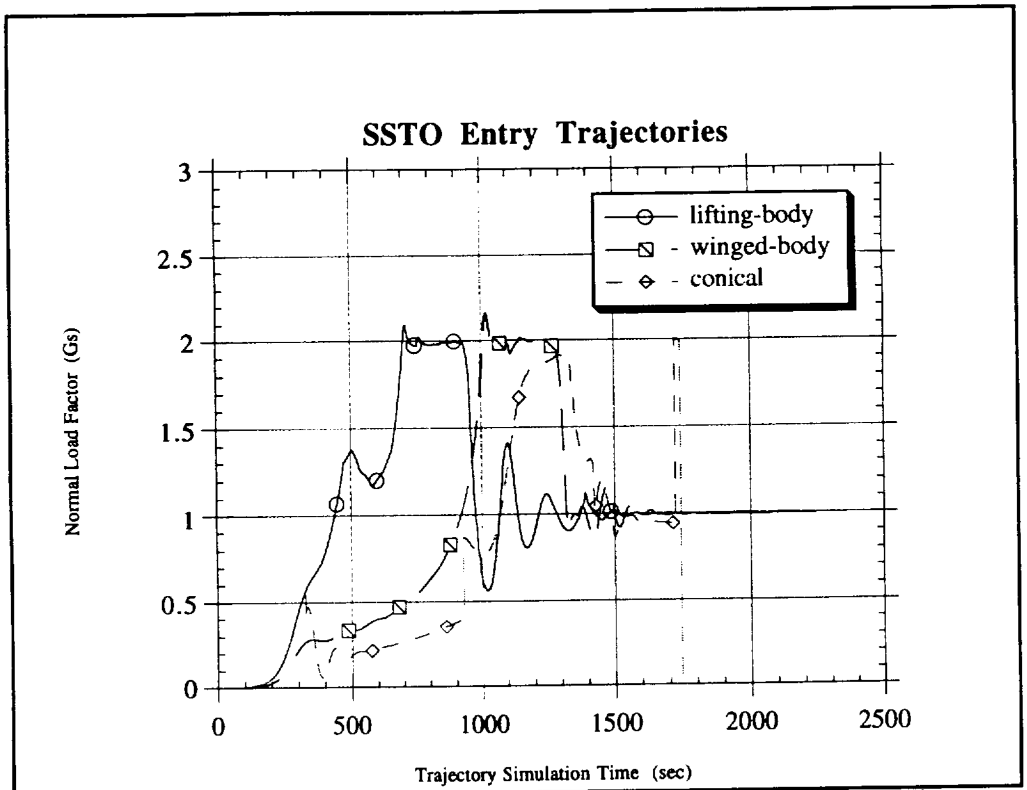



**Figure 6.2-12 Entry Trajectory Normal Load Factor Time History**

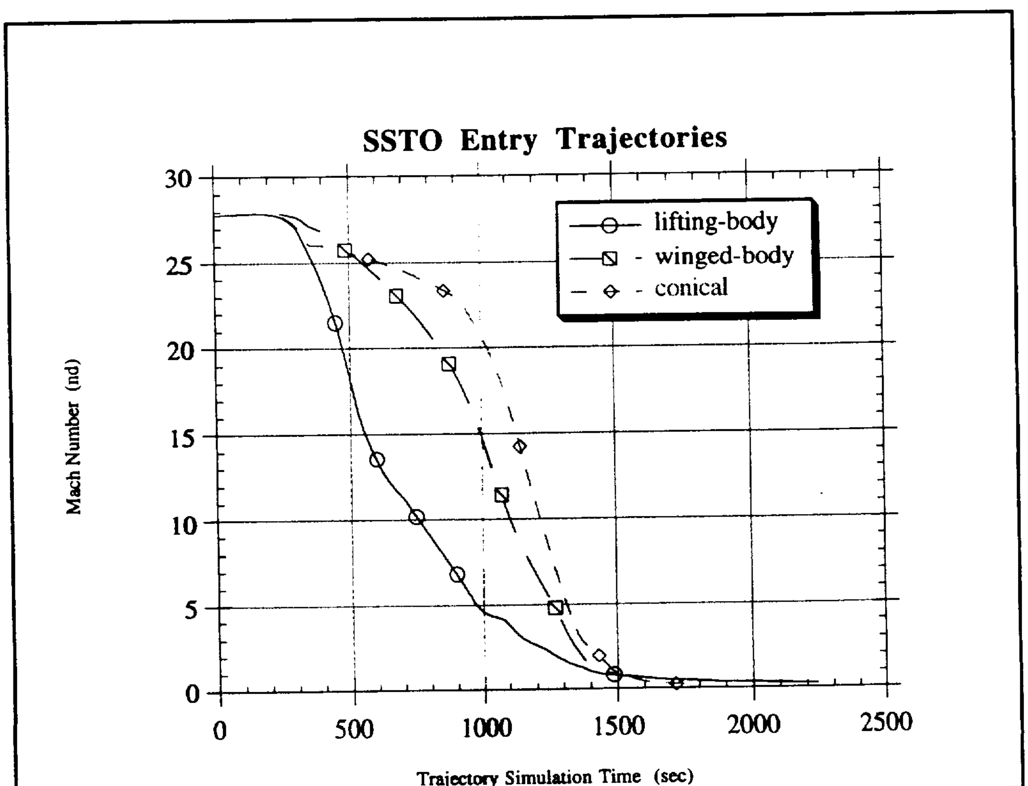



**Figure 6.2-13 Entry Trajectory Mach Number Time History**

## VTOL Pull-Up and Landing Maneuver

The assessment of the VTOL configuration included an initial analysis of the feasibility of using the main propulsion in a "propulsive pull-up" maneuver, as was previously described in Section 6.1 above. A sizing sensitivity assessment was first performed on the VTOL concept, as described in Section 5, to understand the sensitivity of the VTOL's dry mass (which ultimately drove vehicle cost) to the amount of main propellants required to perform the landing maneuver. The result of the sensitivity assessment indicated that total landing maneuver delta velocity ($\Delta V$) requirements greater to or equal to approximately 1,000 fps started having a significant deleterious effect on vehicle dry mass.

Figures 6.2-14 through 18 illustrate the resulting time histories for geodetic altitude, normal load factor, q-alpha, angle of attack, and relative flight path angle during the pull-up and landing maneuvers for the VTOL configuration. As seen from the figures, a feasible pull-up maneuver was developed that only utilized the main propulsion to provide flight path control authority. The DV required to perform the maneuver however, was above 1,000 fps, without considering additional propellant requirements for performance dispersions and a fuel bias (to avoid LOX-rich fuel-depletion cut-off conditions that would destroy the engines). A more optimal solution to the pull-up and landing maneuver would have been to utilize aerosurfaces on the vehicle to provide additional flight path control authority, and lessen the propulsive requirement.

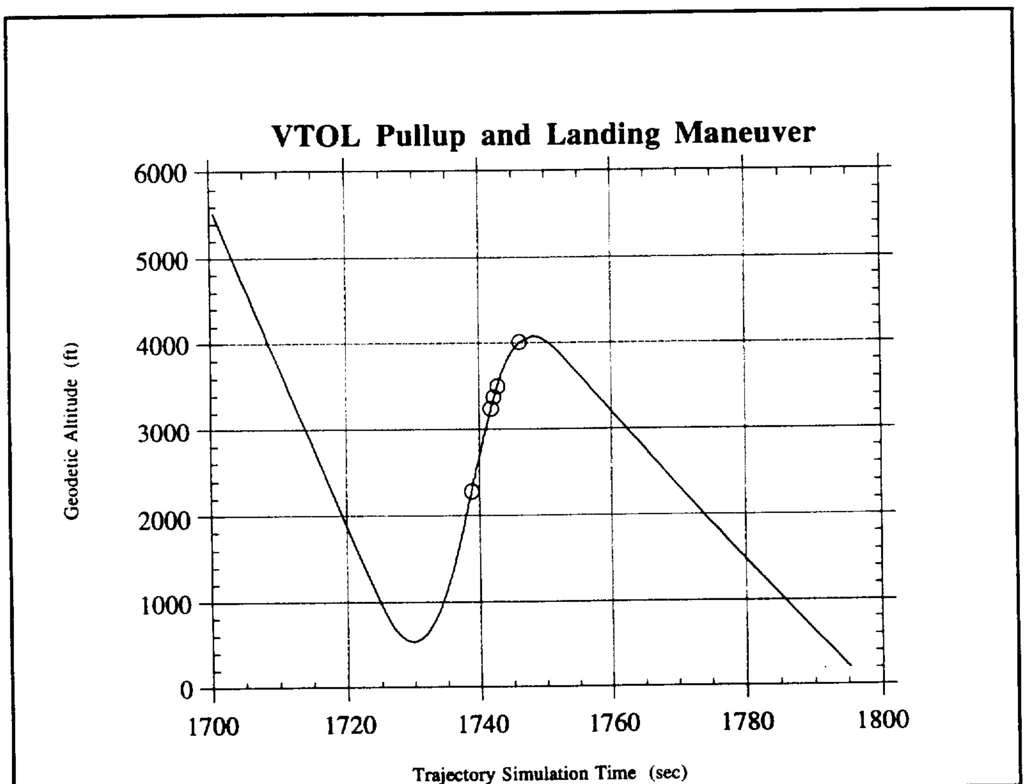



**Figure 6.2-14 VTOL Pull-up Geodetic Altitude Time History**

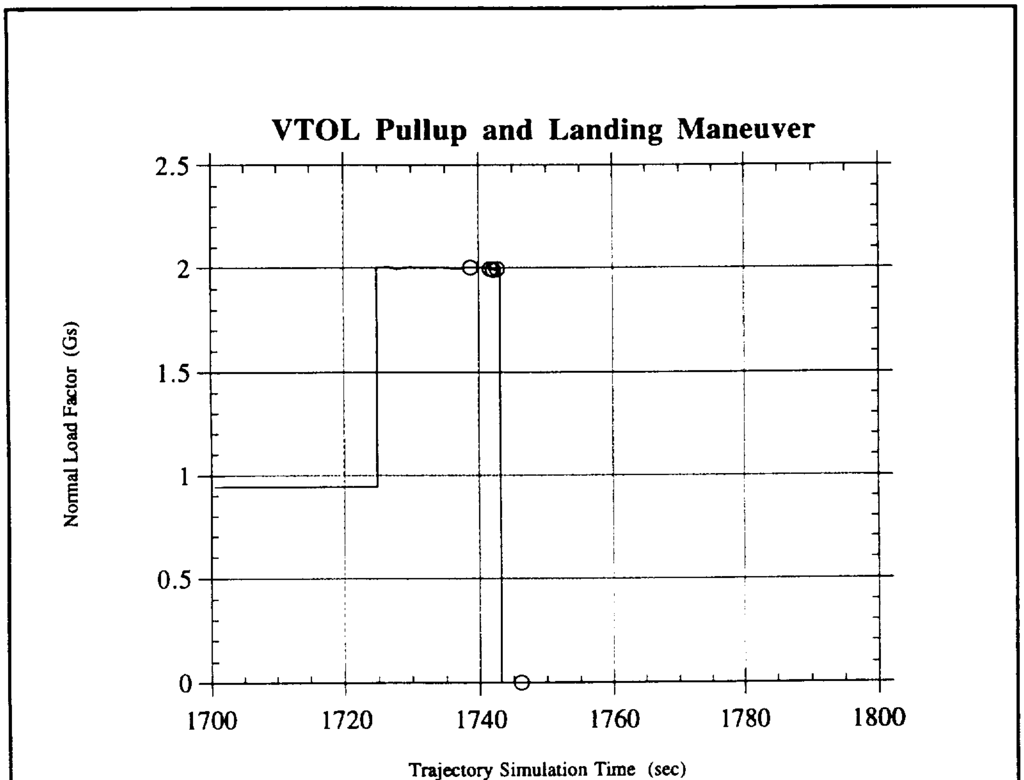



**Figure 6.2-15** VTOL Pull-up Normal Load Factor Time History

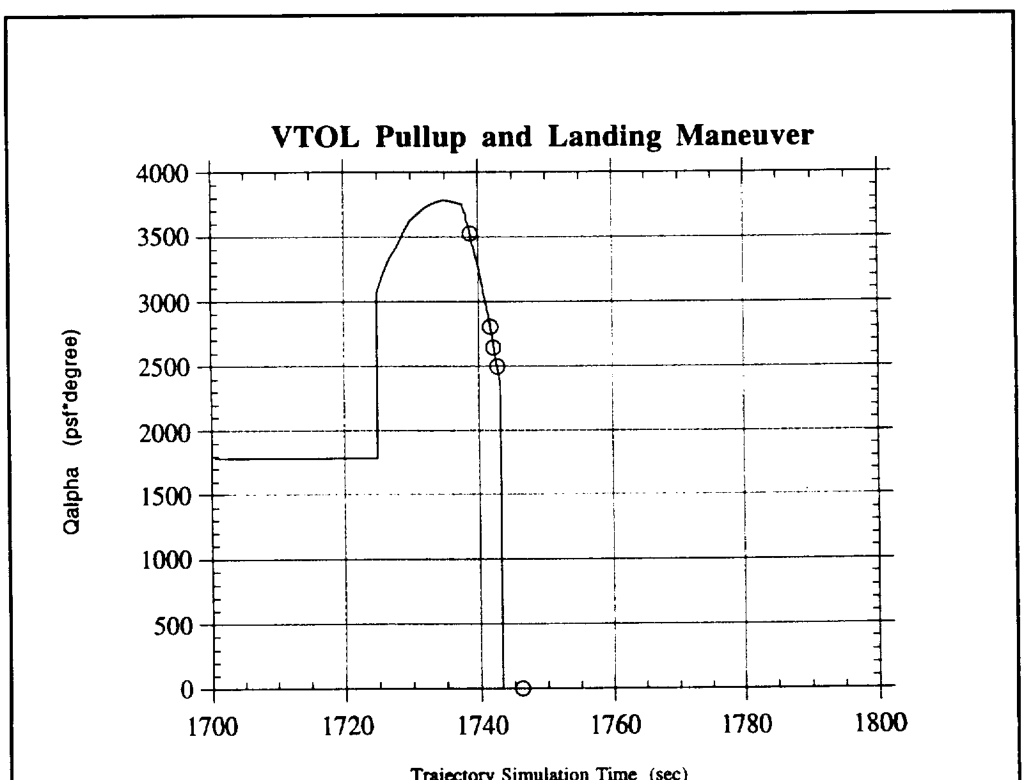



**Figure 6.2-16** VTOL Pull-up Qalpha Time History

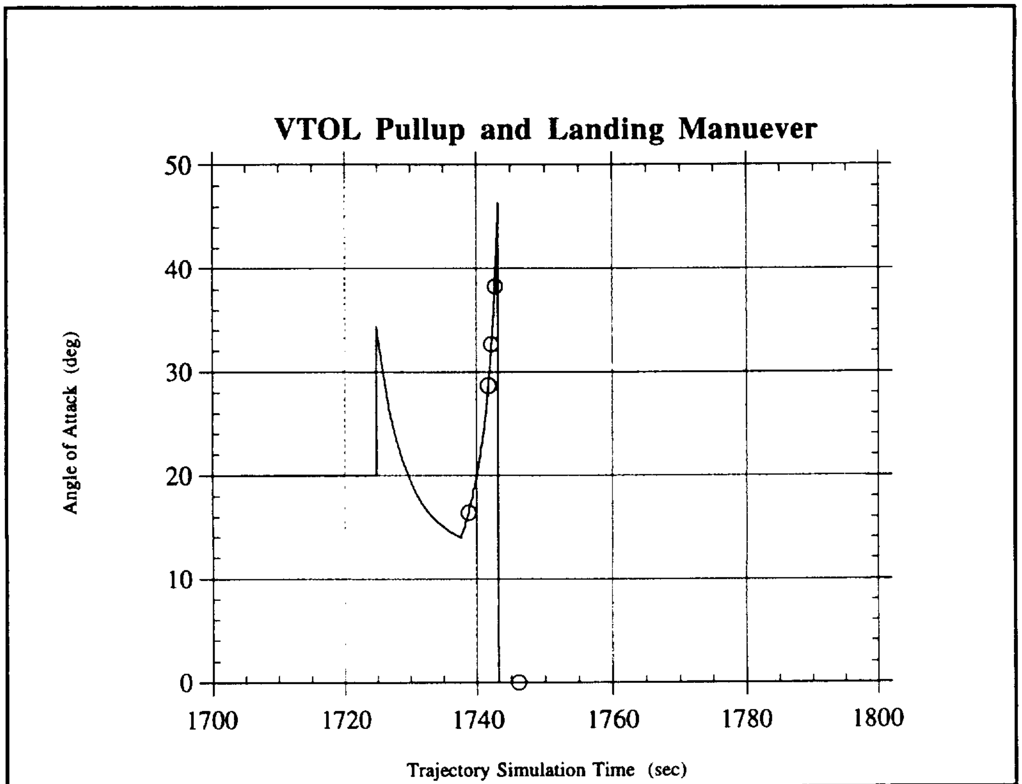



**Figure 6.2-17 VTOL Pull-up Angle-of-Attack Time History**

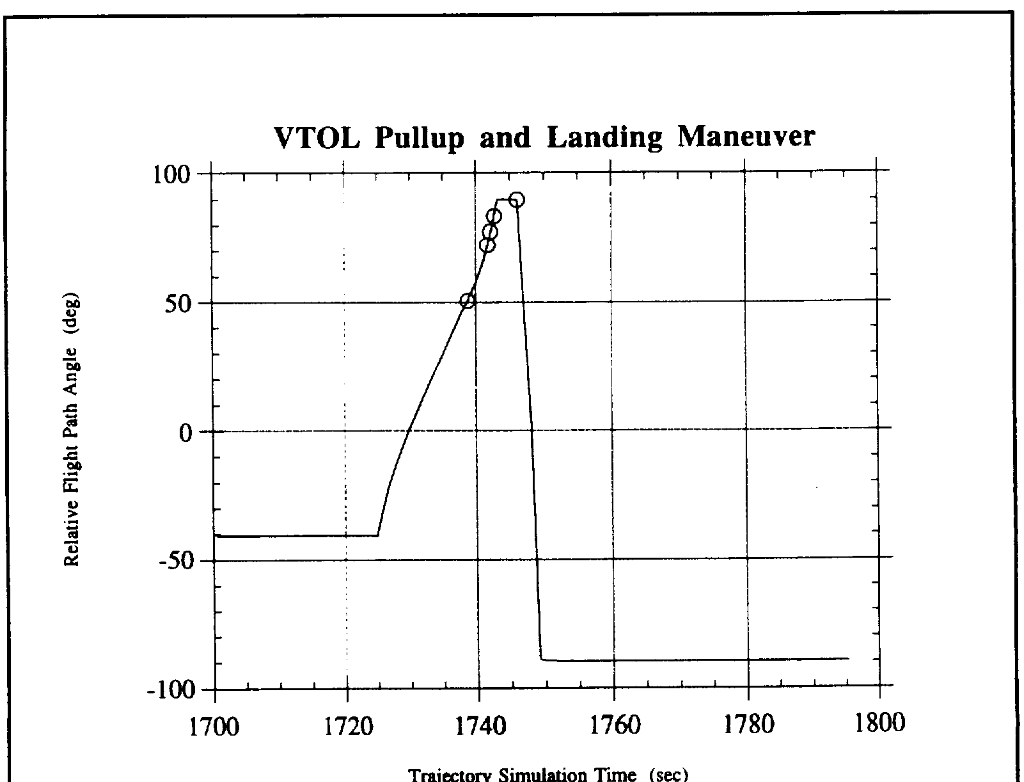



**Figure 6.2-18 VTOL Relative Flight Path Angle Time History**

# 7.0 SSTO Assessment Conclusions

## Vehicle Sizing

As was mentioned in Section 1, contractual time and budget limitations constrained the scope of the SSTO assessments that were performed on the TA-2 contract. Design closure was reached for the first iteration of the VTOL and lifting body VTHL SSTO concepts prior to depletion of the TA-2 funding resources. Second-order fidelity, down to the subsystem definition level, was achieved in the SSTO sizing tools and sizing benchmarks were achieved with very good comparisons with the LaRC Option 3 wing-body SSTO results. There were some differences in the subsystem sizing assumptions used between LaRC and TA-2, but those were isolated. The principal difference was in the sizing of the thrust structure, which has since been resolved (LaRC modified their method to reflect the method used by TA-2). TA-2's SSTO assessment demonstrated that three major SSTO configuration types could be sized with commonalty of groundrules and assumptions, and that the effect of different technology assumptions could also be introduced. The wide variety of propulsion concepts that were collected, providing an excellent matrix from which to chose from for future advanced transportation system studies. A major legacy of the TA-2 tract is the collection of SSTO sizing tools and the associated user's guide documentation, contained in Section 8.

## General Observations

SSTO vehicle stability during unpowered flight was found to be a first-order driver on the outer moldline and internal subsystem and structure layout. Ascent flight mechanics and performance were fundamentally the same for similar thrust-to-weight profiles between the three SSTO configuration types. Entry flight mechanics and performance were different between the configurations when trying to produce similar entry heat and heat rate profiles, and crossrange. The next step in the design process would have been the influence of second-order performance variations in the main propulsion system, such as installed thrust-to-weight, Isp, chamber pressure, and mixture ratio variations, on vehicle operability and dry mass.

## VTOL

The VTOL SSTO configurations will involve large diameter propellant tanks (>27 ft.), raising manufacturability issues. Multi-cell propellant tank designs may be required to minimize propellant tank diameter and mass. The flight mechanics assessed for the final landing maneuver were complex and were found to greatly influence the structural design requirements of the vehicle. There was insufficient time during the study to fully optimize an entry trajectory profile, but a very good preliminary trajectory was developed. An aerospike main propulsion concept will likely be required to sufficiently lower the inherently high power-on and power-off base drag associated with the VTOL configurations. The alternative, utilizing a large number of engines, addresses power-on base drag alleviation, but results in a tremendous operations and reliability burden.

The lifting body configuration provided the largest degree of control over entry stagnation heat rate and total heat load while also providing an acceptable crossrange capability (greater than 700 nm).

## Wing-Body VTHL

The wing-body configuration was found to be less robust in the ascent and entry trajectory profile definition due to wing bending, shear, and torsion loads design constraints. Designing a wing that was tolerant to ascent and entry loads provided a contrary solution to the natural SSTO design to keep vehicle dry mass as low as possible, for performance reasons. The wing-body configurations also required more high-temperature thermal protection system coverage due to the high degree of small radii-of-curvature on the vehicle. The entry heat loads were found to be better than the VTOL concepts, but worse than the lifting body concepts.

**Lifting Body VTHL**

The lifting body configurations provided a more robust load path and higher inherent body stiffness, at the price of a slightly higher dry mass, due to the aeroshell that must wrap around the propellant tanks, subsystems, and payload bay. The entry heating load was found to be significantly lower than either of the two other SSTO types, which will result in a significant savings in hot structure and high-temperature TPS requirements. A significant finding was that the parameterization of the lifting body design resulted in a much greater number of design options being available, as compared to the other two concepts. With a larger number of significant design parameters available for variance, the lifting body concept lends itself to providing greater ease of design closure, when design issues arise. The only downside of the lifting body concept was a greater dry weight sensitivity to aerodynamic and stability considerations.

# 8.0 SSTO Sizing Tool User's Guide

This section contains a copy of the user's guide that was written by Mr. Keith A. Holden of LMSC to accompany the set of three SSTO sizing tools that he developed on the TA-2 contract. An electronic copy of the sizing tools may be obtained from the TA-2 COTR, Gary Johnson, at the Marshall Space Flight Center.

LMSC P096611

# Single Stage to Orbit Launch Vehicle Sizing Tool User's Guide

developed for the
Advanced Transportation System Studies
TA-2
Heavy Lift Launch Vehicle Development

Contract NAS8-39208

June 1994

Submitted in Accordance with the
Requirements of
Contract NAS8-39208, DPD 756
For Data Requirement #3

By

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# FOREWORD

During the course of the Advanced Transportation System Studies (ATSS) Heavy Lift Launch Vehicle Development contract, NAS8-39208, Lockheed was requested to assist in the development and assessment of Single Stage-To-Orbit (SSTO) concepts. During the course of this assignment, three SSTO launch vehicle sizing tools were developed to model three different SSTO launch vehicle configurations: a conical side entry vertical takeoff and landing (VTOL) configuration, a winged body vertical takeoff and horizontal landing (VTHL) configuration, and a lifting body VTHL configuration. This document is a user's guide for the three sizing tools.

## CONTENTS

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# 1. SIZING TOOL DESCRIPTION

This section of the user's guide includes a description of three sizing tool models developed on NASA Contract NAS8-39208, a description of how the sizing tool models are installed on a Macintosh computer, and a description of the input data needed to run the sizing tool models.

## 1.1 Introduction

Lockheed was tasked on Contract NAS8-39208 to design and assess three different classes of single-stage-to-orbit (SSTO) vehicle concepts: a side-entry, conically shaped vertical take-off and landing (VTOL) concept; a winged body vertical take-off/horizontal landing (VTHL) concept; and a lifting body VTHL concept. Because of the significant design differences between the three vehicle classes, a separate vehicle sizing tool was developed for each class, with some aspects of commonalty between the tools. A description of each of the three tools is provided herein. The sizing tools are intended for use in the preliminary design phase of a project where the effects of various vehicle configuration alternatives are compared.

Section 1 of the user's guide provides a description the sizing tools. Section 2 provides a description of the equations used in the sizing tools and lists of typical values of the constants that the user will put into the input data files. Section 3 provides test case printouts for the three launch vehicle configuration sizing tool models.

A Macintosh IIci computer was used during the development of the sizing tools. Microsoft Excel 4.0 was the program used.

## 1.2 General Layout

The sizing tools described herein model a conical side entry VTOL SSTO launch vehicle configuration, a winged body VTHL SSTO launch vehicle configuration, and a lifting body VTHL SSTO launch vehicle configuration. The launch vehicle sizing tools use an iterative approach to calculate the size and mass of a launch vehicle configuration. Figure 1.1 illustrates the functional flow within each of the sizing tools. The sizing tools use an iterative approach that estimates the velocity required to reach the mission orbit, the propellant load required to reach the velocity requirement, and the vehicle structural mass required to contain the amount of propellant required. Each iteration drives these three parameters closer to a converged solution. After the vehicle sizing calculations have converged, the estimate of the velocity required to reach the mission orbit should be checked against a trajectory analysis on the resulting vehicle configuration. This trajectory analysis should then be used to calculate a temperature map of the launch vehicle configuration during reentry. The temperature map will be used to check the initial assumptions on the thermal protection system (TPS) required to protect the launch vehicle configuration during ascent and reentry.

The mission requirements are entered into the sizing tool input data file. These requirements include the payload mass, payload bay size, acceleration limits, destination orbit inclination, apogee, perigee, on-orbit mission velocity requirements, number of crew, time spent on-orbit and the average on-orbit power and heat rejection requirements. The ascent dynamic pressure and loads constraints are not used by the sizing tools. However, they will come into play when the mission velocity requirement is refined by a trajectory analysis.

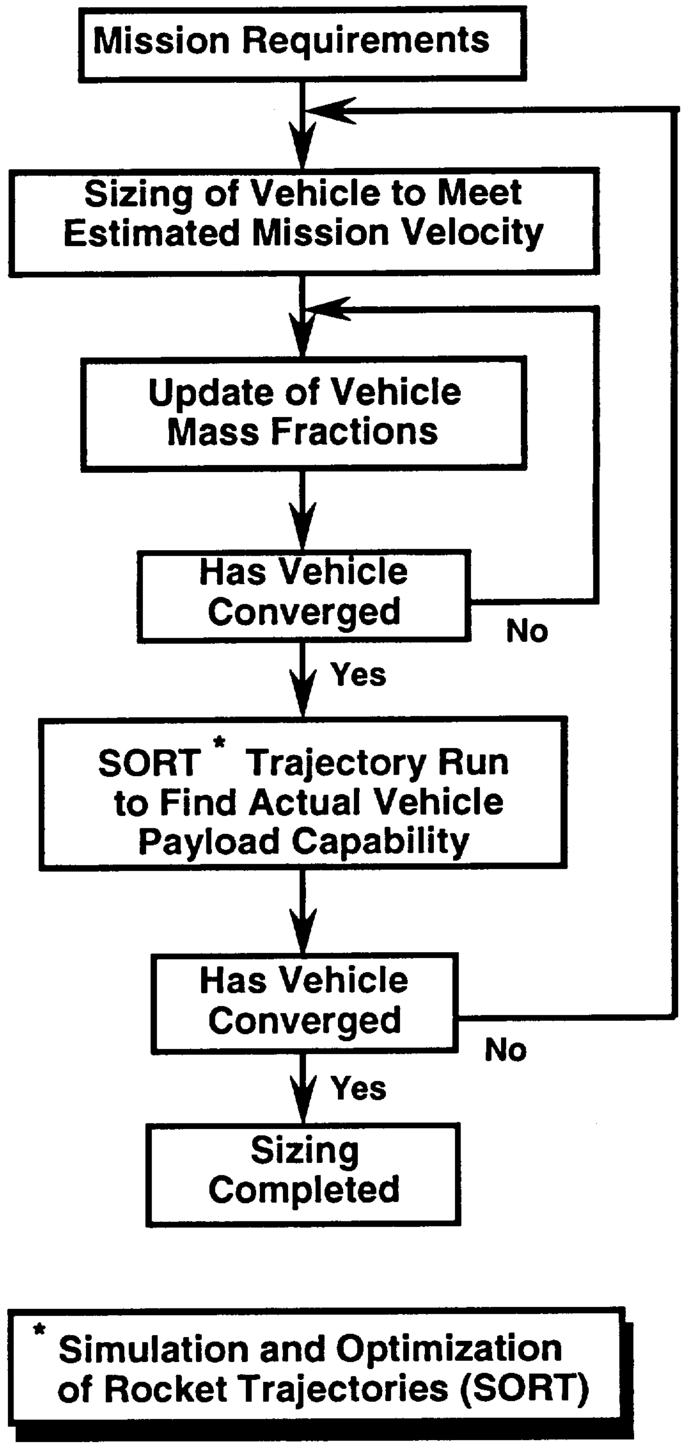



**Figure 1.2-1 Launch Vehicle Sizing Tools Iterative Solution Schematic**

The side entry cone VTOL sizing tool, the winged body VTHL sizing tool, and the lifting body VTHL sizing tool were developed from a generic SSTO sizing tool. Separate sizing tools were developed because these launch vehicle configurations were too different to be covered by a single general purpose sizing tool. The three sizing tools have a performance spreadsheet and a weights spreadsheet. Figure 1.2-2 illustrates the submodules that are interfaced between the two common spreadsheets. Information flows both ways between these spreadsheets until the sizing tool has converged on a solution.

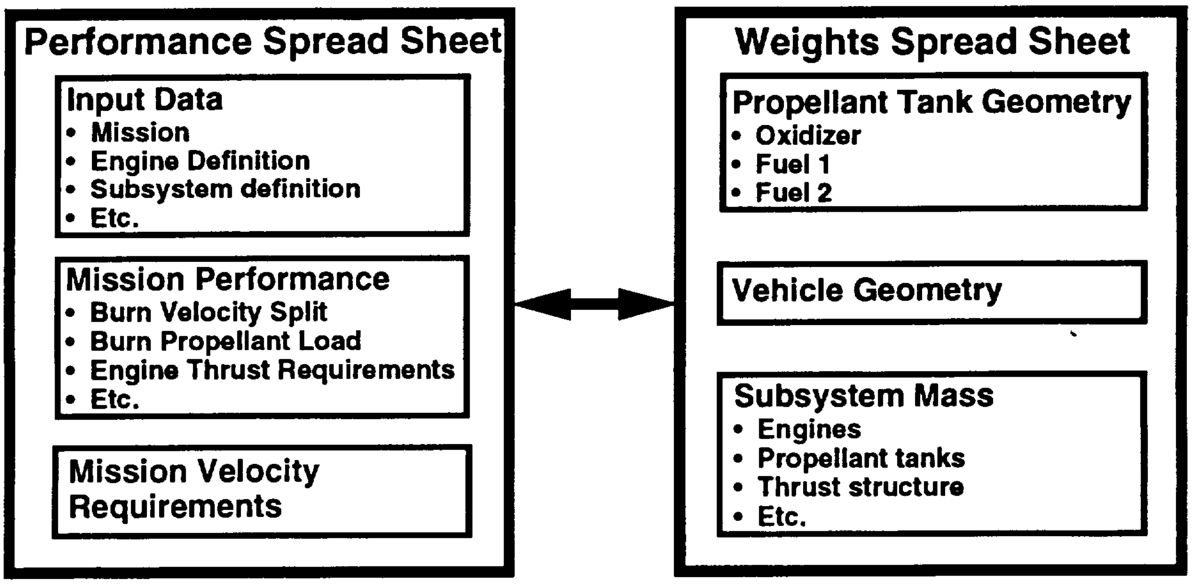



**Figure 1.2-2 Common Sizing Tool Features**

The approach used in the sizing tools is to split the mission velocity required to reach orbit into endoatmospheric (Mode 1) and exoatmospheric (Mode 2) velocity segments. The sizing program calculates the total velocity requirement. The user supplies a Mode 2 velocity estimate. The sizing program then calculates the Mode 1 velocity requirement, the Mode 1 and Mode 2 propellant requirements, and the resulting vehicle masses. The Mode 2 delta velocity capability is varied by the user to find the Mode 1 and Mode 2 propellant loads that result in the total minimum structural mass. This approach is used to allow the use of rocket engines that have different performance characteristics in the two moves, such as mixture ratio, expansion ratio, propellant configuration, etc.

The performance spreadsheet has an input data section, a mission performance section, and a mission velocity requirements section. The input data section contains all of the data used by the sizing tool to define the launch vehicle configuration model. The mission performance section uses the vehicle masses supplied by the weights spreadsheet, the calculated mission velocity requirement, and the rocket equation to calculate the amount of propellant needed by the vehicle to reach main engine cutoff (MECO) conditions. The mission velocity requirements section calculates the delta velocity required as a function of the Mode 1 burn and the Mode 2 burn initial thrust-to-weight ratios.

The weights spreadsheet has a propellant tank geometry section, a vehicle geometry section and a subsystem mass section. The propellant tank geometry is calculated from the propellant load requirement supplied by the performance spreadsheet. The vehicle geometry is calculated from the tank geometry. The vehicle masses are calculated from the vehicle geometry, the propellant masses, and the engine thrust are supplied from the performance spreadsheet. The vehicle masses are then supplied back to the performance spreadsheet for the next pass through the iterative loop. This iterative loop continues until the model has converged onto a solution.

The vehicle sizing tools will converge on a single point design. Each of the three launch vehicle configurations have their own configuration-specific geometrical sizing parameters. In addition, the Mode 2 velocity split, the initial vehicle liftoff thrust-to-weight ratio, and the Mode 2 initial thrust-to-weight ratio are general launch vehicle sizing parameters. The configuration can be optimized by varying the sizing parameters.

The initial vehicle liftoff thrust-to-weight ratio trades off the velocity required to reach orbit against the propulsion system (Section 2.5) mass. The propulsion system includes the main engines (Section 2.5.1), the thrust structure (Section 2.5.4), and the propellant feed (Section 2.5.2) systems. If the propulsion system thrust-to-weight ratio is high enough, the optimum initial vehicle liftoff thrust-to-weight ratio will be larger than the minimum acceptable vehicle liftoff thrust-to-weight ratio.

The Mode 2 initial thrust-to-weight ratio is a function of both the initial Mode 2 vehicle mass and the initial Mode 2 engine thrust (see the engine control flags discussed in Section 2.5.1). "Mixed mode" propulsion concepts may be used, in which Mode 1 constitutes an engine configuration that is initially tailored for the atmospheric portion of flight (such as a tripropellant mode, or a low expansion ratio mode) and then reconfigured for exoatmospheric flight (such as a bipropellant, high specific impulse mode). If the mixed mode option is chosen (burn flag = 3) and Modes 1 and 2 initial thrust-to-weight ratios are specified (engine flag = 2), the user will need to check the Mode 1 engine mass against the Mode 2 engine mass in the mission performance section of the performance spreadsheet for a given Mode 2 initial thrust-to-weight ratio to confirm that the Mode 1 engine mass is equal to the Mode 2 engine mass.

The use of a mixed mode propulsion system opens up the question of the proper split of thrust between the Mode 1 and Mode 2 propulsion phases. The Mode 1 engine performance, Mode 2 engine performance, total vehicle propellant density, choice of the velocity and the initial thrust-to-weight ratio are all affected by the choice of the thrust split. To evaluate this question, the user will need engine data for several thrust splits. The initial thrust-to-weight ratio will then be chosen to match the engine thrust for each velocity requirement.

The engine lengths will change as the engine thrust changes. Engine length is one of the user vehicle inputs. The lifting body configuration is more sensitive to the value used for the engine length than the winged body configuration or the side entry cone configuration.

The Excel program has a table function where one or two variables can be varied at a time. If the table function is used, check to be sure that the range of parameters covered by the table is in the valid design space. If the solution blows up at any of the points, all of the data in the table will be lost.

Optimizing a vehicle requires the choice of some parameters to optimize the vehicle against. The three classical parameters that launch vehicles are optimized against are the vehicle gross liftoff weight, the vehicle propellant mass, and the vehicle dry (structural) mass. Thought should be used in the selection of an optimization parameter. The optimization of the vehicle using any one of these three parameters (or any other parameter) will result in a different vehicle. Classically, vehicle dry mass has a more direct correlation to vehicle cost than the other two.

The payload of a launch vehicle is the difference between the mass put into orbit and the vehicle mass. As the vehicle mass becomes a larger fraction of the total mass put into orbit, the vehicle size and mass grow. At some point, there will not be a possible solution.

The three vehicle configurations have an engine bay heat shield (Section 2.2.5). This shields the engine bay from plume heating and the recircularization of hot gasses. If a plug nozzle is used, the base of the plug will cover the engine bay such that an engine bay heat shield will not be needed.

The number of engines is a factor in the sizing thrust structure (Section 2.5.4) mass. Because of the close integration between the body and engine possible with a plug nozzle engine, treat it as one engine with reduced thrust structure coefficients. This argument will also apply to an engine using a large number of small modular thrust cells if the thrust cells are wrapped around the edge of the vehicle aft skirt.

Because the sizing tools are computer programs written as Excel spreadsheets, the values in the cells in a spreadsheet are changed sequentially as the iteration proceeds. The value in any given cell is a function of values in other cells that may or may not have already changed such that the values in some of the cells will hunt at the start of a new design point. If you are near the edge of the configuration design space, it will be necessary to make small changes from the previous design point to keep the solution from blowing up.

If you are stuck near the edge of the configuration design space and are having trouble backing away, try making small reductions in the vehicle dry mass contingency (Section 2.12).

It is strongly recommended that an input data file be used when a new case is being setup. After the new input data file has been setup, the input data can then be copied onto the performance spreadsheet input data section. This technique allows quality control checks to be made to insure that the new case is correctly setup, provides a record of what is in the new case and starts the new case with all of the changes applied at the same time.

### 1.2.1 VTOL Side Entry Cone Configuration

This sizing tool model assumes a vertical takeoff and landing configuration, where the tradeoff is between the mass of the wings required for a winged body configuration against the extra propellant required to propulsively land the vehicle vertically. The sizing tool model assumes a conical body with integral propellant tanks instead of a squared off vehicle with non integral propellant tanks. The tradeoff here is that an axisymmetrical vehicle with integral propellant tanks will have a lower vehicle dry mass at the cost of a smaller cross range capability.

This sizing tool model assumes atmospheric reentry is performed on the vehicle's side instead of the base reentry of the classical VTOL configurations. The side entry configuration has a lower heat load on its thermal protection system (TPS) (Section 2.3.1) and a larger cross range capability at the cost of having a larger surface area exposed to high heat loads, require a tip-over maneuver from its horizontal equilibrium gliding flight to vertical for landing that will require propellant, and must be designed for stable flight and loads in both the vertical and horizontal orientations.

All vertical landing configurations require a propulsive maneuver for landing. This requirement opens up several problems. The vehicle is dependent upon the engines igniting for the landing maneuver. Engine-out protection can be provided by igniting more engines than are required and running them at less than full rated thrust. A bell rocket engine at sea level will have a larger throttle range as the engine area ratio is decreased, while conversely it will have better altitude performance as the nozzle area ratio increases. A two-position nozzle bell rocket engine design will be heavier and more complex than a single nozzle position design. Because the mass ratio of an SSTO launch vehicle will be approximately ten to one. The thrust required for the landing maneuver will be about ten percent of the liftoff thrust. The weight and complexity of the engine bay will increase as the number of independent bell rocket engines increases. A vertical landing launch vehicle configuration must have an acceptable compromise among these conflicting factors. This compromise must answer the questions of the use of a bell rocket engine or the use of an unconventional nozzle rocket engine, the number of engines used, and will the engines use nozzle extensions.

The sizing tool model assumes all of the Mode 1 engines require a pre-chill cycle prior to ignition, which is factored into calculating the engine restart propellant (Section 2.14.5) requirement. It is also assumed that none of this propellant is used propulsively. If these assumptions do not apply, the coefficients in this equation should be adjusted.

Since this sizing tool model is a side entry VTOL vehicle configuration, the landing maneuver includes a vehicle tip-over phase and a touch-down phase. The tip-over maneuver will include some mixture of using the landing engines and using the body flaps (Section 2.1.3). The body flaps are modeled as four flaps with a total planform area equal to a user input constant times the vehicle base cross section area. The propellant required for the landing maneuver (Section 2.14.4) is defined by the velocity required to slow the vehicle from its terminal velocity in its vertical orientation plus the velocity used as the vehicle hovers at one g for a user-input number of seconds of hover time. The user needs to establish a landing maneuver definition that is covered by the body flap size and vehicle hover time and perform a trade study to find the best split between the flap size and vehicle hover time.

The vehicle aft skirt extends aft to cover the length of the rocket engines. This choice was made to protect the vehicle's rocket engines from the heat loads of reentry.

The mass of a piece of unpressurized structure is a function of the mass of everything above it and of the aero loads imposed on it during entry and the vehicle factor of safety.

The sizing tool model assumes the vehicle has one oxidizer tank and one or two fuel tanks. The user defines the propellant tank arrangement and layout by use of the propellant tank flags (Section 2.2.3). These choices include the forward-to-aft arrangement of the propellant tanks, the propellant tank endcap type, the use of toroidal propellant tanks, and the payload bay location in the forward or aft intertank.

The weights spreadsheet splits the wetted body area into zones to calculate the TPS mass (Section 2.3.1). The user inputs TPS unit masses for each zone based on the initial estimate on the temperatures and heat loads that the zone will see.

The vehicle base diameter and vehicle side half angle are the primary geometrical configuration sizing parameters.

This sizing tool model has a performance spreadsheet, weights spreadsheet, loads spreadsheet, and an airloads spreadsheet (Figure 1.2.1-1).

The airloads spreadsheet calculates the bending moments imposed on the vehicle during entry and sends the information to the loads spreadsheet. The user consults a vehicle entry trajectory analysis and picks the most heavily loaded part of the reentry trajectory. At this point, the user inputs Cn and Ca data as a function of vehicle station into the airloads spreadsheet. The vehicle configuration geometry and mass information are supplied to the airloads spreadsheet by the weights spreadsheet.

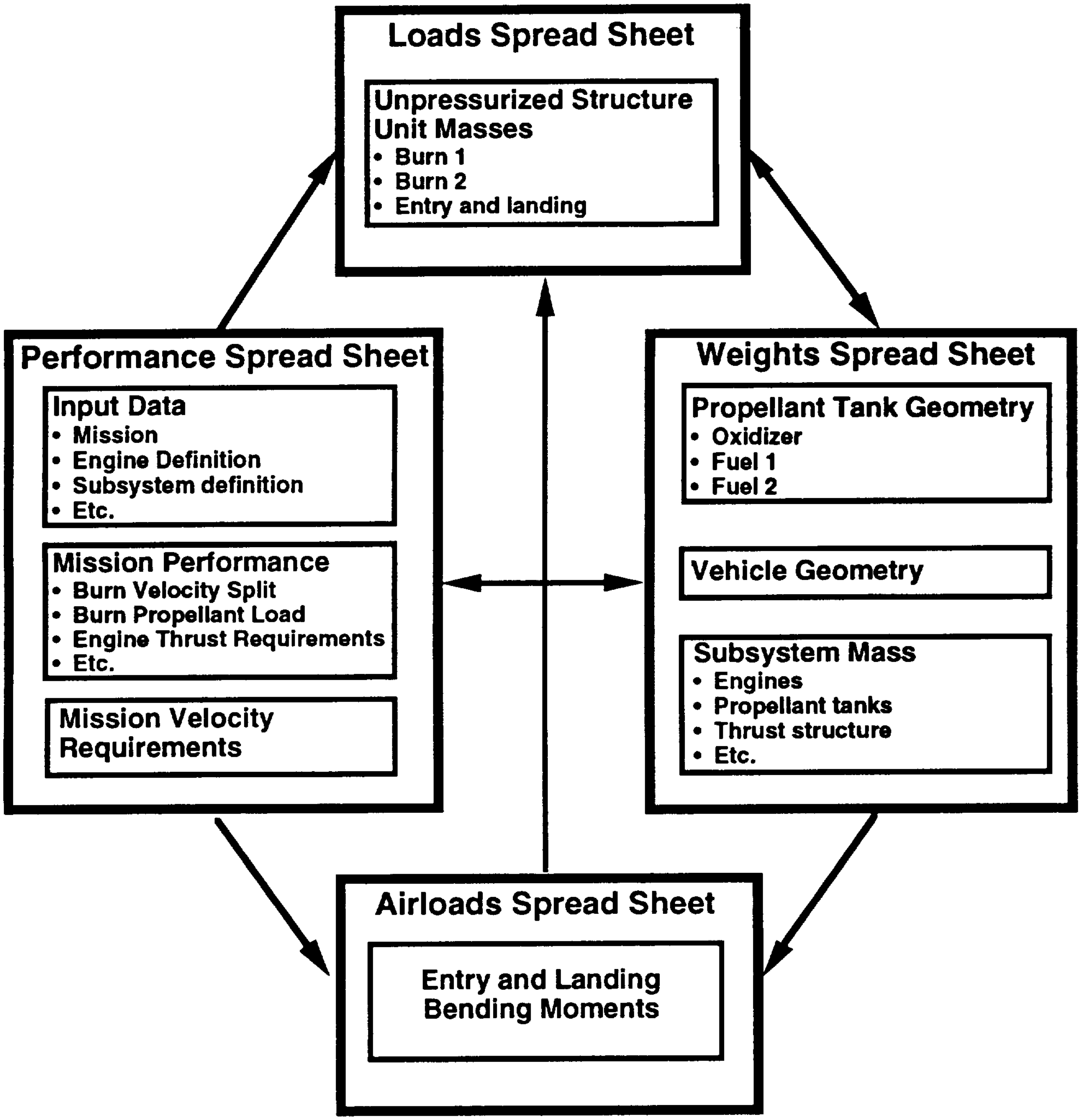



**Figure 1.2.1-1 Conical Side Reentry VTOL Configuration Model**

Axial loads are calculated on the vehicle elements based on the mass of everything above it.

The axial load conditions are looked at during Modes 1 and 2. These load conditions are at the end of the burn or at the point the vehicle reaches maximum acceleration during that burn. The axial loads imposed on the vehicle aft skirt as the vehicle is setting on the pad are also looked at. The vehicle element unit masses are calculated for the axial load cases and the bending moments case from the airloads spreadsheet.

The maximum value of the axial loads, the bending moments and a user-supplied minimum acceptable unit mass for the unpressurized structure are compared and the maximum value of the unpressurized structure unit masses are sent to the weights spreadsheet.

### 1.2.2 VTHL Winged Body Configuration

This sizing tool model assumes a cylindrical winged body configuration. The vehicle fuselage uses an integral tank design. This choice was made because an integral tank design is more structurally efficient and is therefore a lighter design.

The wing size is chosen to give a reasonable compromise between the additional mass on the vehicle because of the wings, a lower ballistic coefficient because of the additional wing planform area, and an acceptable vehicle cross range capability and landing speed because of the resulting vehicle lift to drag ratio. The user defines the wing mass by the choice of several wing coefficients (Section 2.1.1).

The unpressurized structure unit mass is a function of the maximum normal acceleration and the safety factor.

The body flap length is sized by the larger of enough length to shield the engines from the reentry heat loads and being long enough to have adequate control authority. This length is defined by a user input body flap length-to-width ratio. The body flap width is the width of the vehicle body.

The sizing tool model assumes the vehicle has one oxidizer tank and one or two fuel tanks. The user defines the propellant tank arrangement and layout by use of the propellant tank flags (Section 2.2.3). These choices include the forward-to-aft arrangement of the propellant tanks, being located in the forward or aft intertank.

The weights spreadsheet splits the wetted body area into zones to calculate the TPS mass (Section 2.3.1). The user inputs TPS unit masses for each zone based on the initial estimate on the temperatures and heat loads that the zone will see.

This configuration was modeled to provide a calibration of the sizing tool vehicle mass estimates against a known vehicle design. A comparison was made of the vehicle subsystem masses between the vehicles published in the Access to Space Option 3 Final Report and comparable vehicles modeled by this sizing tool. The dry mass comparison for these vehicle configurations were very close for all of the vehicle subsystems except the vehicle thrust structure. Due to Lockheed's sizing tools factoring in the effect of number of engines on thrust structure sizing, which was not directly done in the Option 3 effort, the thrust structure masses predicted by this sizing tool was a factor of two heavier than the thrust structures of the vehicles in the Access to Space Option 3 Final Report.

The vehicle diameter is the primary geometrical configuration sizing parameter. If the vehicle configuration has a body length-to-diameter ratio constraint, there is only one vehicle diameter that will meet this constraint for any given vehicle configuration.

This sizing tool model has a performance spreadsheet and a weights spreadsheet (Figure 1.2.2-1). There are no additional spreadsheets.

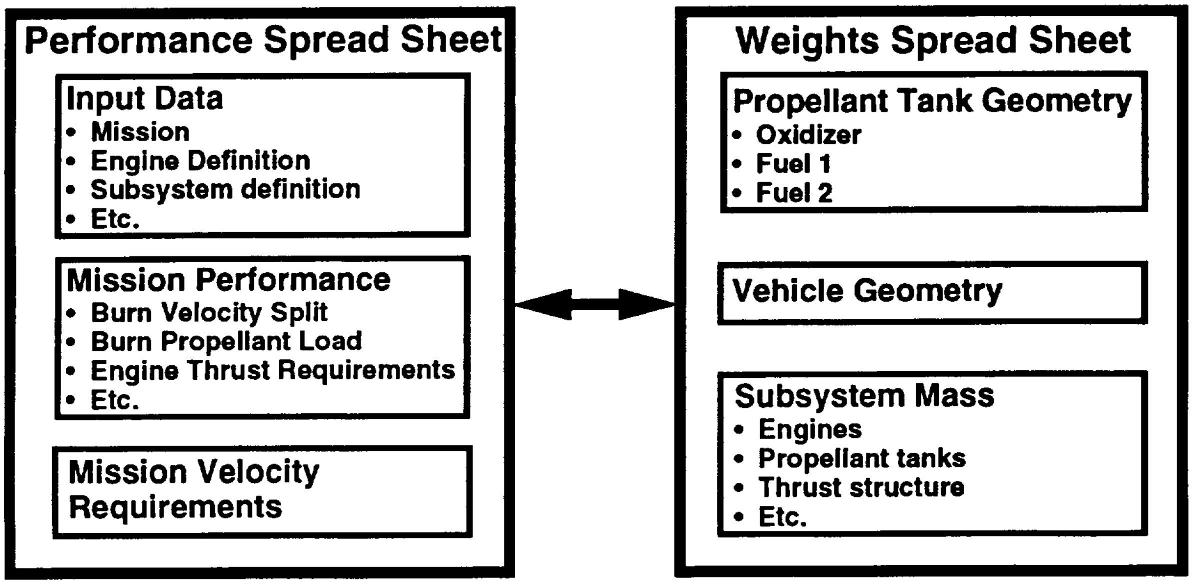



**Figure 1.2.2-1 Winged Body VTHL Configuration Model**

### 1.2.3 VTHL Lifting Body Configuration

This sizing tool model assumes a lifting body configuration. Because of the higher structural efficiencies of cylindrical and conical pressurized propellant tanks, the configuration's propellant tanks are buried inside the configuration aero shell. The tradeoff here is the additional complexity and mass of having an aero shell against the reduction in mass in not having wings.

A lifting body configuration should have a lower ballistic coefficient than a winged body configuration. This lower ballistic coefficient should translate into lower peak heating rates during reentry and therefore lower TPS unit mass (Section 2.3.1).

The unpressurized structure unit mass is a function of the maximum normal acceleration and the safety factor.

The body flap length is defined to be equal to the width of the engine bay, which is long enough to shield the engines from the reentry heat loads.

This sizing tool model has a performance spreadsheet, a weights spreadsheet and a tank sizing spreadsheet (Figure 1.2.3-1). The propellant tank geometry function was taken out of the weights spreadsheet and put into its own spreadsheet. Because of the large number of possible propellant tank and payload bay geometry choices, the tank sizing spreadsheet was written with a specific tank geometry (Figure 1.2.3-2). If a different propellant tank and payload bay geometry is needed, a new tank sizing spreadsheet must be written.

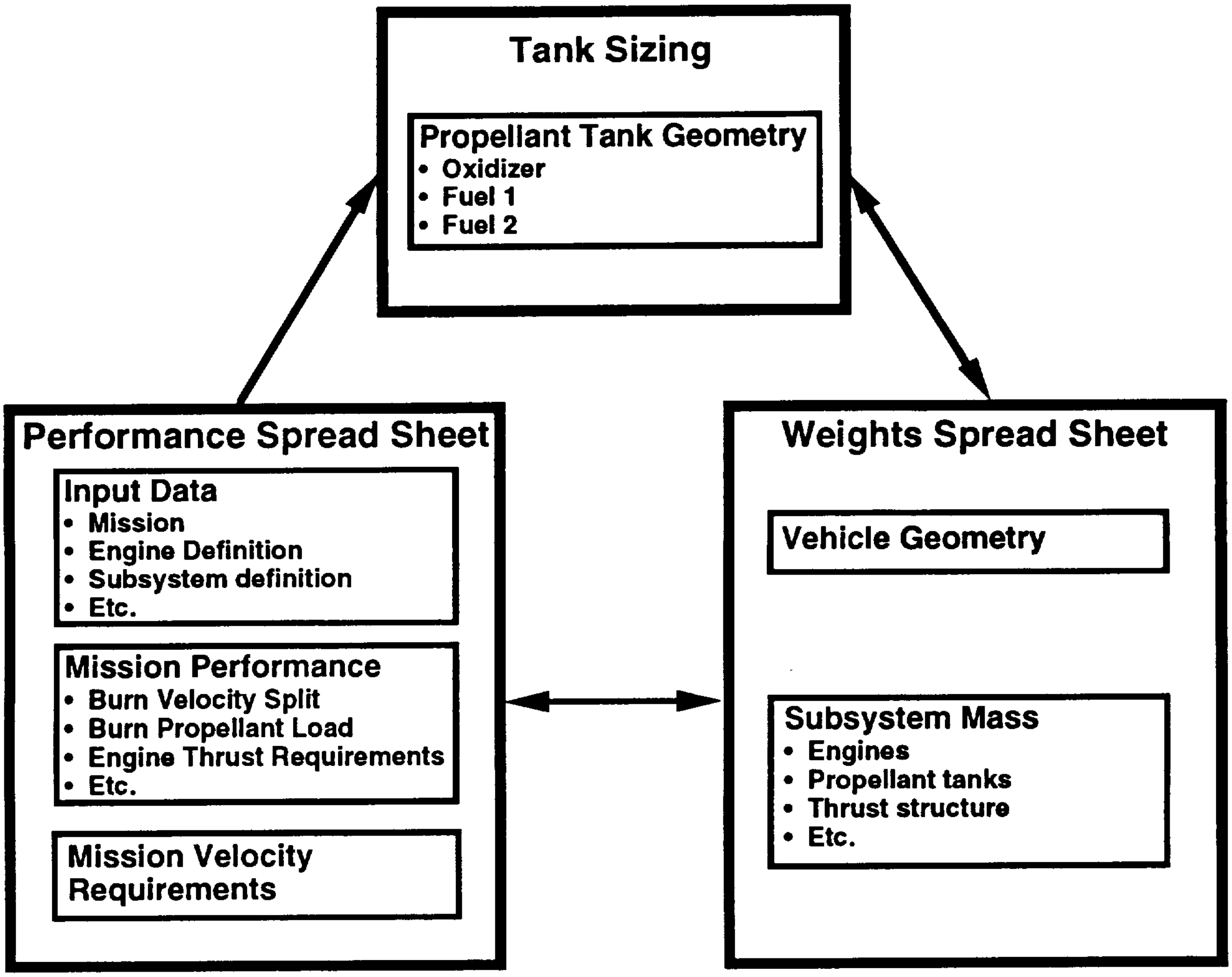
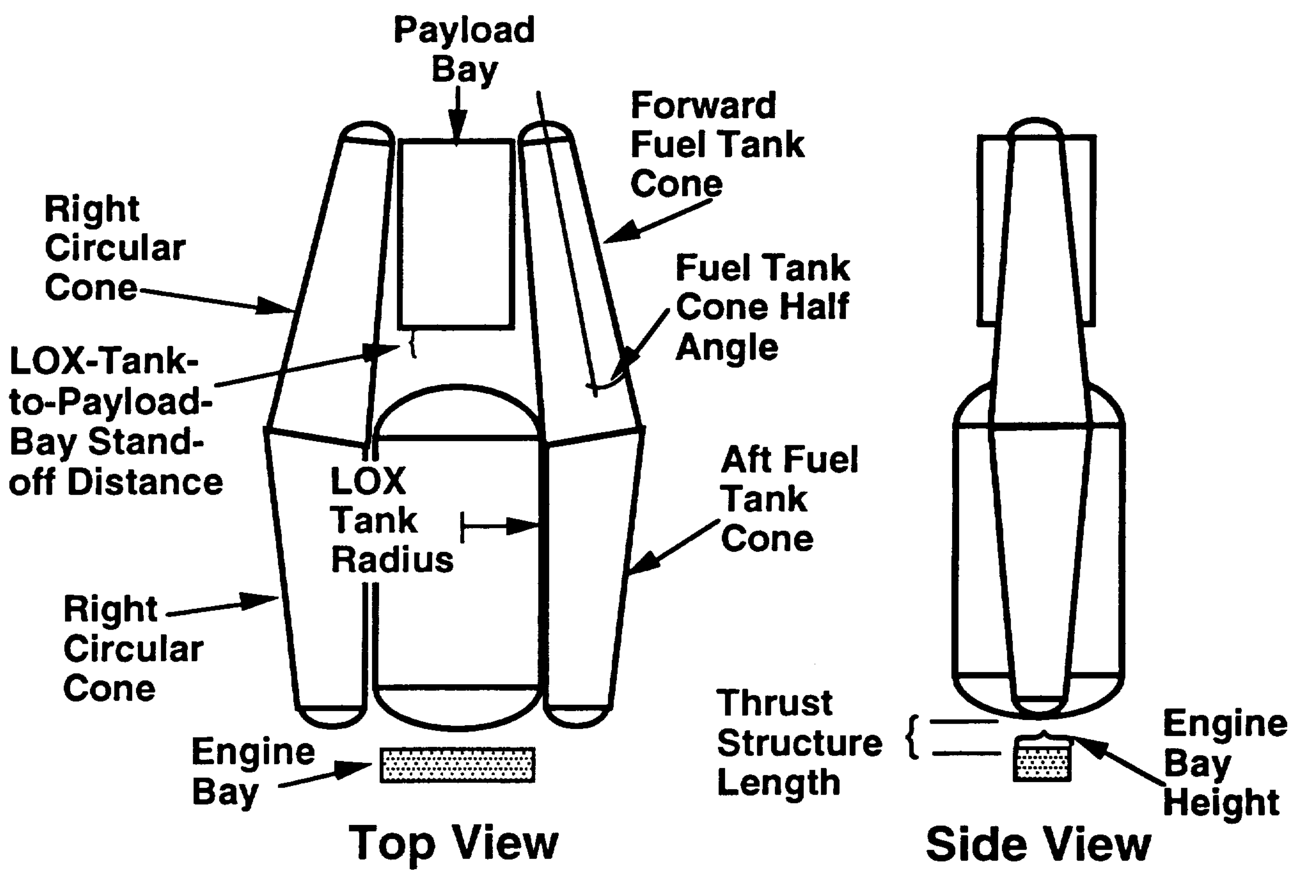



**Figure 1.2.3-1 Lifting Body VTHL Configuration Model**




**Propellant Tank and Payload Bay Layout**

**Sizing Parameters**

- LOX Tank Radius
- Forward Fuel Tank Cone Half Angle
- Main Engine Bay Height
- Thrust Structure Length
- LOX-Tank-to-Payload-Bay Stand-off Distance

Figure 1.2.3-2 Lifting Body Configuration Propellant Tank and Payload Bay Layout

The payload bay and propellant tank geometry has the payload bay forward of the oxidizer tank with the fuel split into port and starboard tanks on the sides of the oxidizer tank and payload bay.

If a tripropellant configuration is used, there will be two fuels used, with the fuel one tanks smaller and forward of the fuel two tanks.

The sizing tool was written to allow the oxidizer tank forward and aft radii to be independently varied. However, care needs to be used in keeping the slope of the oxidizer tank barrel section reasonable.

The weights spreadsheet uses the propellant tank and payload bay geometry from the tank sizing spreadsheet to calculate a vehicle planform area. User defined inputs are then used to go from vehicle planform area to the vehicle wetted body area.

The weights spreadsheet splits the wetted body area into zones to calculate the TPS mass (Section 2.3.1). The user inputs TPS unit masses for each zone based on the initial estimate on the temperatures and heat loads that the zone will see.

The vehicle geometry is defined by several factors. The oxidizer tank radius defines the oxidizer tank length and therefore the length of the fuel tank. With the assumption that there is a straight line from the mid fuel tank radius to the aft of the engine bay, the engine bay height defines the half angle of the aft fuel tank cone. When the forward fuel tank cone half angle, the aft fuel tank cone half angle, and the fuel tank length are defined, the fuel tank forward, mid and aft radii are determined. The vehicle planform geometry does have other inputs such as the payload bay geometry, the crew cabin geometry (which is presumed to be located forward of the payload bay for crewed missions), the thrust structure length and the standoff distances between the vehicle parts. However, these other inputs in the vehicle geometry are not likely to change from one case to the next. Therefore, the forward and aft oxidizer tank radii, the half angle of the forward fuel tank cone and the engine bay height are the primary configuration sizing parameters that change the tank planform area and therefore the vehicle geometry and mass.

The use of this sizing tool model is more complex and labor intensive than the use of the other two sizing tool models. The larger number of vehicle geometry parameters gives this sizing tool model a larger range of possible vehicle configuration options. All of these possible configurations will have their own mass and aero characteristics. Because of this, the vehicle design will need input from the aero section to arrive at a good compromise between the configuration aerodynamics and mass. Also, since the model body geometry is calculated from the vehicle planform and a table of user input coefficients, the user will need to generate a new set of body geometry coefficients for each new configuration geometry.

### 1.3 Parentage of the Sizing Model

- Most of the equations and some of the technology coefficients were from NASA TM 78661, "Techniques for the Determination of Mass Properties of Earth-to-Orbit Transportation Systems," by I. O. MacConochie and P. J. Klich, June 1976.

- Additional technology coefficients were from "Space Transportation Architecture Study Special Report - Final Phase, Book 3," General Dynamics Space System Division, November 1987, Contract NAS8-36615.

- Residual propellant equation and the data that was used to develop the thrust structure equations were from "Space Shuttle Synthesis Program (SSSP), Volume II, Weight/Volume

Handbook Final Report," General Dynamics Convair Aerospace Division, December 1970, Contract NAS9-11193.

- An equation to calculate the non-optimum weight factors on the design of propellant tanks was from "A Semi-Empirical Method for Propellant Tank Weight Estimation," L. A. Willoughby, 27th Annual Conference of the Society of Aeronautical Weight Engineers, May 1968

- Space Shuttle Orbiter component mass information was from "Orbiter Detail Weight Statement (OV-103)," SD75-SH-0116-216, Rockwell International, August 2, 1993 and "Press Information, Space Shuttle Transportation System," Rockwell International, January 1984

- The SSTO(R) component mass information was from "Access to Space Study, Advanced Technology Team (Option 3) Final Report," July 1993

- Equations to calculate the unpressurized structure unit mass for the side entry conical configuration were from "Aerospace Vehicle Design, Volume II, Spacecraft Design," by K. D. Wood

- TPS unit mass coefficients are from "Reusable Surface Insulations for Reentry Spacecraft," AIAA 91-0695, S. Amanda Chiu and William C. Pitts; and "Assessment of Alternative Thermal Protection Systems for the Space Shuttle Orbiter," AIAA 82-0899.

### 1.4 Sizing Tool Installation

These sizing tool models were developed on a Macintosh IIci computer using Microsoft Excel 4.0.

The software included with this user's guide includes the three files containing the three sizing tool models and a global macro sheet containing the TABLE_READ macro. This macro is used in the performance and loads spreadsheets and must be installed for the sizing tools to work. If you already have a global macro sheet, copy the TABLE_READ macro to it. If you do not already have a global macro sheet, copy this global macro sheet to the Excel Startup Folder under the Systems Folder/Preferences/Excel Startup Folder files.

### 1.5 Sizing Tool Input Data Files

This section of the user's guide contains an explanation of the user inputs necessary to run the sizing tool models. These inputs are configured for the Option 3 Access to Space Final Report version of the RD-701 engines. Therefore, these are tripropellant vehicle configurations. See also the listings of the input data files and the performance spreadsheets in Section 3. Where appropriate, cross references will be made back to the sizing tool model equation descriptions in Section 2.

The convention used in these sizing tool models is that data can be put into an uncolored data cell, data is locally calculated in lightly colored data cells and that data is transferred from another spreadsheet in heavily colored data cells.

The mission velocity requirements section of the performance spreadsheets have two data tables describing changes in mission velocity requirements for differing initial Mode 1 and Mode 2 initial thrust-to-weight ratios. Although these data tables can be changed, it should not be necessary to change them.

### 1.5.1 Side Entry Conical VTOL Launch Vehicle Configuration

The following 12 blocks of data are from the RD-701 input data file for the side entry conical VTOL launch vehicle configuration sizing tool model. This input data is also in the input data section of the sizing tool performance spreadsheet. After each block of information, an explanation of the inputs will be given with cross references back to the equation descriptions in Section 2 where appropriate. The last table is from the airloads spreadsheet. Aerodynamic coefficients as a function of body station are entered into this table. A new set of aerodynamic coefficients should be calculated for a new vehicle configuration.

| Input Data: | |
|---|---|
| Payload (Wpay) | 25,000 lbm |
| Payload bay diameter | 15.00 ft |
| Number of crew | 0.00 |
| Crew cabin volume | 0.00 ft^3 |
| Number of days on-orbit | 7.00 |
| Average on-orbit power usage | 5.00 kw |
| Average on-orbit heat rejection requirement | 10.00 kw |
| Maximum acceleration (No) | 3.000 g |
| Factor of safety | 1.40 |
| Orbit inclination | 51.60 deg |
| Orbit perigee | 50.00 NM |
| Orbit apogee | 100.00 NM |

The vehicle mission is defined in this first block of user input data.

The first entry is the mission payload (Section 2.18). The sizing tool model assumes the payload is both carried up to orbit and landed.

The next entry is the payload bay diameter.

The next two entries are the number of crew on the vehicle (Sections 2.2.1 and 2.11) and the pressurized structural volume on the vehicle (Section 2.11).

The next three entries are the number of days on-orbit, the average on-orbit power requirement and the average on-orbit heat rejection requirement (Sections 2.8, 2.11). These three entries size the fuel cells supplying the on-orbit power, the mass of the fuel cell reactants and reactant tanks required to supply this amount of energy and the heat rejection system mass required to dissipate the waste heat. The reason the average heat rejection requirement is larger than the average power usage is there will be waste heat from the fuel cells in generating the required amount of power and the crew and possibly the payload will also be generating waste heat that needs to be removed.

The next entry is the maximum axial acceleration of the vehicle during ascent. This is one of the vehicle ascent trajectory constraints used in the trajectory analysis to verify the estimate of the mission velocity required to reach orbit and is used to define the axial loads on the vehicle and therefore the unpressurized structure unit masses (Section 2.2.2).

The next entry is the vehicle safety factor. The safety factor is an allowance for the vehicle seeing loads that may be larger than the design loads during the vehicle's operational life. The safety factor increases the vehicle's load tolerance and therefore the vehicle's mass. The safety factor is applied to the axial loads on the vehicle and therefore the unpressurized structure unit masses (Section 2.2.2) and the propellant tank wall thickness (Sections 2.2.3).

The last three entries are the main engine cut off (MECO) orbit conditions of inclination, perigee and apogee. This information is used by the mission velocity requirements section of the performance spreadsheet to estimate the mission velocity required to reach the mission orbit and is also used in the trajectory analysis to verify the mission velocity requirements. The orbital maneuvering system (OMS) velocity budget (Section 2.14.3) includes allowances for the on-orbit transfer from the MECO orbit to the mission orbit, the de-orbit burn and any other on-orbit maneuvering burns required.

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank |
|---|---|---|---|
| Position= | 3 | 2 | 1 |
| Ullage= | 0.05 | 0.05 | 0.05 |
| Density= | 71.20 | 50.50 | 4.43 lbm/ft^3 |
| Residual A= | 0.0038 | 0.0038 | 0.0016 lbm/lbm |
| Residual B= | 0.0010 | 0.0010 | 0.0012 lbm/lbf |
| Ullage pressure= | 35.00 | 35.00 | 50.00 psi |
| TPS unit mass= | 0.250 | 0.000 | 0.250 lbm/ft^2 |
| | Fwd Tank | Mid Tank | Aft Tank |
| Forward endcap height coefficient= | 0.7071 | 0.7071 | 0.7071 |
| Aft endcap height coefficient= | 0.7071 | 0.7071 | 0.7071 |
| Tank design= | 3 | 2 | 3 |
| Upper endcap flag= | 1 | 1 | 1 |
| Lower endcap flag= | 1 | 1 | 1 |

Propellant tank data is defined in this block of user input data. The first section refers to the oxidizer tank and the fuel one and fuel two tanks. The last section refers to the forward, mid and aft propellant tanks. The fuel one tank and the mid tank do not apply to a bipropellant vehicle configuration. The oxidizer in this table is liquid oxygen. The fuel one in this example is kerosene. The fuel two in this example is liquid hydrogen.

The first entry is the propellant tank position flag (Section 2.2.3). The options are the propellant tank is in the forward position, the middle position or the aft position.

The next entry is the propellant tank ullage factor (Section 2.2.3). This factor increases the volume in the propellant tank to account for the ullage space required for the pressurization of the tank, the propellant burned by the vehicle's main engines prior to liftoff and the volume of the residual ascent main propellant.

The next entry is the propellant density (Section 2.2.3).

The next two entries are ascent residual propellant coefficients (Section 2.14.2). The first coefficient is a factor for the ascent propellant mass. The second coefficient is a factor for the main engine vacuum thrust.

The next entry is the propellant tank ullage pressure (Section 2.2.3).

The next entry is the unit mass of the propellant tank's cryogenic insulation (Section 2.3.3). A room temperature fluid, such as kerosene, does not require cryogenic insulation.

The next two entries are the forward and aft endcap height coefficients (Section 2.2.3). Only elliptical endcaps require inputs in this section. Endcap heights for hemispherical endcaps and toroidal endcaps are hardwired into the program.

The next entry is the tank design flag (Section 2.2.3). The options are a common bulkhead propellant tank design, a nested bulkhead propellant tank design, or a separate propellant tank design. If a nested bulkhead design is selected for a pair of tanks, the aft tank will have a concave forward endcap. If a common bulkhead is selected for a pair of tanks, the aft tank will not have a forward endcap. For both of these cases, the forward tank of this pair will have a convex aft endcap.

The last two entries are the propellant tank forward and aft endcap flags (Section 2.2.3). The options are an ellipsoidal endcap, a hemispherical endcap, or a toroidal endcap.

| Engine Restart Propellant Coefficients: | |
|---|---|
| Engine conditioning coefficient, fuel= | 0.000620 |
| Engine conditioning coefficient, oxidizer= | 0.000920 |

The above block of user input data contains the coefficients defining the amount of propellant required to thermally condition the engines prior to starting them for the landing maneuver (Section 2.14.5). The program assumes the mode one engines at full rated thrust will be used for the landing maneuver and that none of this propellant is used propulsively. The first entry is the fuel engine conditioning coefficient. The last entry is the oxidizer engine conditioning coefficient. If all of the mode one engines are not restarted for the landing maneuver, or if some of the engine cooldown propellant is used propulsively, the values of these coefficients should be adjusted.

| Vehicle Materials: | Fwd Tank | Mid Tank | Aft Tank | Unpressurized Structure | |
|---|---|---|---|---|---|
| Stiffener constant A= | 2.3 | 2.3 | 2.3 | 0.52 | |
| Stiffener constant B= | 0.44 | 0.44 | 0.44 | 0.83 | |
| Density= | 0.098 | 0.098 | 0.057 | 0.057 | lbm/in^3 |
| Ftu= | 65,600 | 65,600 | 90,400 | | psi |
| Modulus of elasticity= | 9,900,000 | 9,900,000 | 9,450,000 | 9,450,000 | psi |

This block of user input data contains the coefficients used in defining the materials and types of structure used in the propellant tanks and the unpressurized structures. The middle tank is not used in a bipropellant vehicle configuration. The first two entries are stiffener coefficients used in defining the structure types used (Section 2.2.2). The next entry is the density of the material used (Section 2.2.2). The next entry is the ultimate strength of the material used (Section 2.2.3). The last entry is the modulus of elasticity (Young's modulus) of the material used. In the Access to Space Option 3 final report, the materials strength was reduced by 20% and Young's modulus was reduced by 10% to account for fatigue. Similar knockdowns over the handbook material properties are recommended. The propellant tanks and unpressurized structures stiffener coefficients and modulus of elasticity are used in the loads spreadsheet calculations of the vehicle's surface unit masses. However, only the unpressurized structures surface unit masses are currently passed from the loads spreadsheet to the weights spreadsheet for the calculation of the aft skirt, aft intertank and the forward intertank masses.

| Mode 2 burn: | | |
|---|---|---|
| Mode two mission velocity | 20,040 | ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 | sec |
| Iv2 | 452.70 | sec |
| Mixture ratio (% Oxidizer) | 85.70 | % |
| Mixture ratio (% Fuel 1) | 0.00 | % |
| Mixture ratio (% Fuel 2) | 14.30 | % |
| Engine height | 11.67 | ft |
| Number of engines | 7 | |
| Engine flag | 2 | |
| Engine mass (if engine flag= 1) | 0 | lbm |
| Engine vac thrust (if engine flag= 1) | 0 | lbf |
| Engine unit mass (if engine flag= 2) | 40.26 | lbf/lbm(vac) |
| No2 (if engine flag= 2) | 1.400 | g |

This block of user input data defines the Mode 2 engines.

The first entry is the Mode 2 mission velocity. This parameter is varied by the user to find the best velocity split between Modes 1 and 2.

The next entry is the Mode 2 sea level specific impulse. This parameter is used only if burn flag (Section 2.5.1) is set to two.

The next entry is the Mode 2 engine vacuum specific impulse.

The next three entries are the Mode 2 mixture ratios. The sum of these entries is 100%. For a bipropellant vehicle configuration, the fuel one mixture ratio is set to zero.

The next entry is the engine height. This parameter will need to be changed to reflect the changes in engine height as the engine thrust is changed in a vehicle configuration optimization. The side entry conical vehicle configuration is not sensitive to this parameter.

The next entry is the number of engines. This parameter is used in the thrust structure mass calculation (Section 2.5.4).

The next entry is the engine flag (Section 2.5.1) definition.

The next two entries are used if the engine flag is set to one. The first parameter is the mass of one engine. The second parameter is the vacuum thrust of one engine.

The last two entries are used if the engine flag is set to two. The first parameter is the engine vacuum thrust-to-weight ratio. The second parameter is the Mode 2 initial thrust-to-weight ratio.

| Mode 1 burn: | |
|---|---|
| Isl1 | 333.50 sec |
| Iv1 | 385.10 sec |
| Mixture ratio (% Oxidizer) | 76.80 % |
| Mixture ratio (% Fuel 1) | 20.20 % |
| Mixture ratio (% Fuel 2) | 3.00 % |
| Engine height | 11.87 ft |
| Number of engines | 7 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine sl thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 82.90 lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 g |

This block of user input data defines the Mode 1 engines.

The first entry is the Mode 1 sea level specific impulse.

The next entry is the Mode 1 engine vacuum specific impulse.

The next three entries are the Mode 1 mixture ratios. The sum of these entries is 100%. For a bipropellant vehicle configuration, the fuel one mixture ratio is set to zero.

The next entry is the engine height. This parameter will need to be changed to reflect the changes in engine height as the engine thrust is changed in a vehicle configuration optimization. The side entry conical vehicle configuration is not sensitive to this parameter.

The next entry is the number of engines. This parameter is used in the thrust structure mass calculation (Section 2.5.4).

The next entry is the engine flag (Section 2.5.1) definition.

The next two entries are used if the engine flag is set to one. The first parameter is the mass of one engine. The second parameter is the sea level thrust of one engine.

The last two entries are used if the engine flag is set to two. The first parameter is the engine sea level thrust-to-weight ratio. The second parameter is the Mode 1 initial thrust-to-weight ratio.

| | |
|---|---|
| Burn flag= | 3 |
| Number of fuel tanks= | 2 |
| Payload bay location flag= | 1 |

The next block of user input data contains several of the vehicle configuration flags. The first entry is the burn flag (Section 2.5.1). The next entry is the number of fuel tanks. For a bipropellant vehicle configuration, there will be one fuel tank. For a tripropellant vehicle configuration, there will be two fuel tanks. The last entry is the payload bay location flag (Section 2.2.3).

| Vehicle sizing coefficients: | |
|---|---|
| Main propellant feed line and press sys= | 55.00 lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 |
| Avionics= | 710 lbm/lbm^(1/8) |
| Range safety= | 0 lbm |
| Crew Cabin Body Constant (Bo)= | 0 lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.035 lbm/lbm |
| Body insulation constant (Kbi)= | 0 lbm/ft^2 (if hot structure is used) |
| Base engine heat shield unit mass= | 1.64 lbm/ft^2 |
| | |
| Gimbal actuator unit mass= | 0 lbm/lbf |
| Thrust structure (Kts1)= | 0.00207 lbm/lbf |
| Thrust structure (Kts2)= | 0.00039 lbm/lbf |
| | |
| Prime Power (PWc) (aero surface)= | 0.274 lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.0000000 lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface) | 0.000 lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 lbm/kw |
| Fuel cell reactant unit mass (FCc)= | 29.26 lbm/kw-day |
| | |
| ECLSS crew cabin constant (Ec)= | 0.00 lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 lbm/kw |
| | |
| Personnel waste systems (PPf)= | 0.00 lbm |
| Personnel seats and crew related (PPs)= | 0.00 lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 lbm |
| Personnel weight (Pp)= | 0.00 lbm/crew member |
| | |
| Body Flap Unit Area= | 0.25 ft^2/ft^2 four flaps are assumed |
| Body flap constant (Bbf)= | 1.17 lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 lbm/ft^2 |
| Control surface miscellaneous (Spc)= | 200 lbm |
| | |
| Payload bay mass= | 5,786 lbm (from the Langley SSTO(R) case) |
| Vehicle base diameter= | 48.40 ft |
| Vehicle cone angle= | 5.50 deg |
| Minimum gage factor= | 1.00 lbm/ft^2 |
| | |
| Maximum normal load case: Angle of attack= | 20.00 deg |
| Maximum normal load case: Qbar= | 95.10 psf |

The above block of user input data are the coefficients used in most of the vehicle subsystem mass calculations.

The first entry is the coefficient used in the calculation of the main propellant feed line and pressurization system mass (Section 2.5.2).

The next entry is the coefficient used in calculating the vehicle dry mass contingency factor (Section 2.12). The contingency factor is an allowance for an increase in vehicle mass during the program.

The next entry is the coefficient used in calculating the avionics system mass (Section 2.10).

The next entry is the coefficient used in calculating the range safety system mass (Section 2.2.6). A range safety system is used to destroy ta launch vehicle if it approaches the sides of the firing range. Reusable vehicles may or may not have range safety systems.

The next entry is the coefficient used in calculating the crew cabin mass (Section 2.2.1).

The next entry is the coefficient used in calculating the landing gear mass (Section 2.4).

The next entry is the coefficient used in calculating the body insulation mass (Section 2.3.2). The need for body insulation is a function of the thermal protection system (TPS) used on the vehicle. If the body insulation is used, it is assumed to cover the fuselage wetted area. A hot structure design will probably require body insulation.

The next entry is the coefficient used in calculating the vehicle engine bay heat shield (Section 2.2.5). This engine bay heat shield keeps the hot engine plume gas out of the engine bay. If the vehicle configuration uses an engine configuration, such as a plug nozzle, that keeps engine recirculating gas out of the engine bay, this system will not be needed.

The next entry is the coefficient used in calculating the main engine gimbal actuator mass (Section 2.5.3). This calculation is only the gimbal actuator mass. The power system mass used to run the main engine gimbals is calculated elsewhere.

The next two entries are the coefficients used in calculating the main engine thrust structure mass (Section 2.5.4). The first coefficient factors in the main engine vacuum thrust. The second coefficient factors in the number of engines used.

The next three entries are coefficients used in calculating the vehicle's prime power system mass (Section 2.8) for the power requirements during ascent to orbit and reentry from orbit. These three coefficients are used to calculate the masses of the power systems required to move the aero surfaces, to gimbal the main engines and to power the avionics system. It may be possible for the on-orbit power systems to supply some of the ascent and reentry power demands.

The next three entries are coefficients used in calculating the power conversion and distribution system mass (Section 2.9). The first entry is used in calculating the electrical power distribution system mass. The next two coefficients are used to calculate the hydraulics power distribution systems mass used to move the aero surfaces and to gimbal the main engines. These last two entries are used only if hydraulic systems are used to move the aero surfaces and to gimbal the main engines.

The next two entries are coefficients used in calculating the vehicle's prime power system mass (Section 2.8) for the on-orbit power requirements. The assumption that is made here is that the fuel cells are used for the average on-orbit power requirement. The first coefficient is the power source (fuel cell power stack) unit mass and the second coefficient is the energy supply (fuel cell reactants and tankage) unit mass. It is assumed that the fuel cells will be able to supply the short term peak power requirements. It may be possible for these fuel cells to provide some of the ascent and reentry power requirements. Power and energy redundancy requirements may require a proportional increase in the value of these coefficients.

The next four entries are coefficients used in calculating the vehicle's environmental control and life support system (ECLSS) mass (Section 2.11). The first entry is the coefficient used in calculating the crew cabin mass. The second entry is the crew supplies unit mass. The third entry is the avionics system waste heat removal unit mass. The last entry is the unit mass of the vehicle on-orbit heat rejection system.

The next two entries are the coefficients used in calculating the personnel provisions mass (Section 2.16). The first entry is for the food waste and water management system mass. The second entry is the unit mass of the crew seats and other related items.

The next two entries are the coefficients used in calculating the personnel mass (Section 2.17). The first entry is the miscellaneous crew mass. The second entry is the crew personnel unit mass.

The next four entries are used to calculate the mass of the body flaps and that of the actuators used to move the body flaps. The control surfaces on this vehicle configuration are assumed to be four body flaps. In this model, the total body flap surface area is calculated by multiplying the first coefficient by the vehicle base cross sectional area. The second coefficient is used in calculating the mass of the body flaps (Section 2.1.3). The third and fourth entries are used in calculating mass of the control surface actuators (Section 2.1.4). Since the body flaps can supply part of the vehicle control during the tip over part of the landing maneuver, the body flaps used on this vehicle configuration will be the result of a tradeoff between the amount of propellant used in the landing maneuver and the mass of the body flaps, actuators and actuator power supply.

The next entry is the payload bay mass (Section 2.2.4). This entry includes the mass of the structure required to support the payload and to distribute the loads from the payload to the rest of the vehicle.

The next two entries are the vehicle base diameter and the vehicle cone angle. This is the angle between vertical and the side of the vehicle. These two parameters are the primary geometry sizing factors for this vehicle configuration (Section 1.2.1).

The next entry is a minimum gage factor. The larger of this minimum gage factor or the calculated body unpressurized structure unit mass based on the vehicle loads are used for the unit masses of the nose, the forward and aft intertanks and the aft skirt (Section 2.2.2).

The last two entries in this block of user input data are used in defining the bending moments imposed on this vehicle configuration in horizontal flight. These two entries are the vehicle angle of attack and the dynamic pressure on the vehicle during reentry and landing.

| OMS and RCS systems mass coefficients: | |
|---|---|
| RCS system mass coefficient= | 0.000151 lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 sec |
| RCS thruster specific impulse (reentry)= | 410.00 sec |
| RCS thruster specific impulse (ascent)= | 350.00 sec |
| OMS system thrust-to-weight= | 0.04 g |
| OMS engine mass coefficient= | 0.035 lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 lbm/lbm |
| OMS thruster specific impulse= | 462.00 sec |

The next block of user input data is used to define the orbital maneuvering system (OMS) and the reaction control system (RCS) dry mass (Sections 2.6 and 2.7 respectively).

The first entry is the RCS mass coefficient used in calculating the RCS mass. The next three entries are the specific impulses used for any on-orbit, entry and ascent RCS engine burns.

The next three entries are the coefficients used in calculating the OMS engine and propellant tankage mass. This program assumes that the OMS propellant tanks are sized to hold all of the OMS, RCS, engine restart and landing propellant.

The last entry in this data block is the OMS engine on-orbit specific impulse.

| The main propulsion system additional delta v may be used for additional on-orbit maneuvers and to adjust the mission velocity requirements based on trajectory analysis results. The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential throttling is selected for vehicle thrust vector control. | |
|---|---|
| RCS ascent mission velocity (applied to GLOW)= | 0 ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 80 ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1,140 ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -71 ft/sec |
| Landing maneuver specific impulse= | 333.50 sec |
| Landing maneuver hover time= | 16.00 sec |
| Landing maneuver drag coefficient= | 0.90 |

The next block of user input data is used to calculate the amount of OMS, RCS and landing propellant required to fly the design mission. This propellant requirement is based on the assumptions that the vehicle lands with the design payload and that the residual ascent propellant is vented prior to the on-orbit OMS and RCS burns. The program also assumes the residual OMS and RCS propellants are accounted for by an increase in the OMS and RCS on-orbit velocity budgets.

The first entry is the ascent RCS velocity requirement. The roll requirements on current launch vehicles include rolling the vehicle into the proper heading shortly after launch and controlling the vehicle's roll during ascent. Launch vehicles with multiple bell engines that use engine gimbaling for thrust vector control will not need to use the RCS thrusters for roll control. Launch vehicles with a single bell engine will need to use the RCS thrusters for roll control. Launch vehicles with multiple bell engines or a single plug nozzle that use differential throttling for thrust vector control may or may not need to use the RCS thrusters for roll control. This program assumes that ascent RCS velocity requirement is applied to the vehicle gross lift off weight (GLOW).

The next entry is the RCS on-orbit velocity requirement.

The next entry is the RCS reentry velocity requirement. This program assumes that the RCS reentry burn occurs after the on-orbit OMS and RCS burns. This vehicle configuration has a larger RCS velocity requirement than the other launch vehicle configurations because it is necessary to hold this vehicle configuration at a slide slip angle to increase the configuration's cross range capability.

The next entry is the OMS on-orbit velocity requirement. This velocity requirement includes orbit transfer from the main engine cutoff (MECO) conditions to the target orbit, any on-orbit maneuvers, and the deorbit burn.

After a new launch vehicle configuration has been developed, a trajectory analysis should be performed to find the actual mission velocity requirement. This next entry is where a delta to the mission velocity estimate calculated in the performance spreadsheet is entered into the program. This entry can also be used to enter into the program any main propulsion system velocity requirements beyond those of the MECO conditions.

The last three entries in this block of user input data are used to calculate the amount of propellant required for the landing maneuver (Section 2.14.4). The first entry is the landing maneuver engine specific impulse. The next entry is the hover time in the landing maneuver. The last entry is the drag coefficient of the vehicle in the vertical orientation. The drag coefficient is used to find the vehicle's terminal velocity in the vertical orientation. The hover time is used to match the landing velocity budget used in the program to a landing velocity budget calculated off-line.

| Nose definition: | |
|---|---|
| Exterior angle 1= | 17.00 deg |
| Exterior angle 2= | 30.00 deg |
| Ratio r2/r1= | 0.80 |
| Ratio r3/r1= | 0.20 |

The next section of user input data is used to define the vehicle configuration's biconic nose. A biconic nose cone reduces the length and therefore the surface area and also the mass of the nose cone. The biconic nose cone is modeled as a lower cone, an upper cone and a hemispherical tip. The first entry is the angle between vertical and the lower cone's side. The second entry is the angle between the vertical and the upper cone's side. The third entry is the ratio in the upper and lower radii for the lower cone. The last entry is the ratio in the upper and lower radii for the upper cone. The lower cone's base diameter is the vehicle diameter at the base of the nose. This cone's upper diameter is found from the lower diameter and the ratio between the upper and lower radii. The exterior angle then defines the cone's geometry. The base diameter of the upper cone is the upper surface of the lower cone. This cone's geometry is likewise defined by the exterior angle and the ratio of radii.

| TPS unit Masses: | |
|---|---|
| Nose= | 2.20 lbm/ft^2 |
| Windward Fwd tank= | 0.65 lbm/ft^2 |
| Leeward Fwd tank= | 0.50 lbm/ft^2 |
| Windward Fwd Intertank= | 0.60 lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 lbm/ft^2 |
| Windward Mid tank= | 0.55 lbm/ft^2 |
| Leeward Mid tank= | 0.35 lbm/ft^2 |
| Windward Aft Intertank= | 0.50 lbm/ft^2 |
| Leeward Aft Intertank= | 0.35 lbm/ft^2 |
| Windward Aft tank= | 0.50 lbm/ft^2 |
| Leeward Aft tank= | 0.35 lbm/ft^2 |
| Windward Aft Skirt= | 0.65 lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 lbm/ft^2 |
| Body Flaps= | 1.50 lbm/ft^2 |

The next block of user input data contains the TPS unit masses (Section 2.3.1) for the vehicle body sectors. These sectors are the windward and leeward sides of the vehicle body, the nose, the forward tank barrel section, the forward intertank, the mid tank barrel section, the aft intertank, the aft tank barrel section, the aft skirt, and the aerodynamic control surfaces. An accurate choice of the TPS unit masses will require knowledge of the distribution of heat loads and the resulting temperatures on the vehicle body. This information can be generated by doing a thermal map of the vehicle body for the ascent and reentry trajectories at the moment of the peak heat loads.

| | CAP | FRUS1 | FRUS2 | FRUS3 | FRUS4 |
|---|---|---|---|---|---|
| CNALPHA | 0.01407 | 0.01148 | 0.00134 | 0.00052 | 0.0179 |
| NF | 1.661773 | 1.355874 | 0.158264 | 0.061416 | 2.114124 |
| CA | 0.39498 | 0.01971 | 0.00229 | 0.00089 | 0.03070 |
| AF | 2.332505 | 0.116395 | 0.013523 | 0.005256 | 0.181295 |

The last table is from the airloads spreadsheet. The input data parameters are the axial and normal force aerodynamic coefficients Ca and Cn-alpha as a function of the body section. The body sections are the nose, forward tank barrel section, forward intertank, mid tank barrel section, aft intertank, aft tank barrel section, and the aft skirt. The table shown above was truncated at the aft intertank entry so that it could be fit on this page.

### 1.5.2 Winged Body VTHL Launch Vehicle Configuration

The following 12 blocks of data are from the RD-701 input data file for the winged body VTHL launch vehicle configuration sizing tool model. This input data is also in the input data section of the sizing tool performance spreadsheet. After each block of information, an explanation of the inputs will be given with cross references back to the equation descriptions in Section 2 where appropriate.

| Input Data: | |
|---|---|
| Payload (Wpay) | 25,000 lbm |
| Number of crew | 0 |
| Crew cabin volume | 0 ft^3 |
| Number of days on-orbit | 7 |
| Average on-orbit power usage | 5 kw |
| Average on-orbit heat rejection requirement | 10 kw |
| Maximum acceleration (No) | 3.000 g |
| Maximum normal acceleration (Nz) | 2.500 g |
| Factor of safety | 1.40 |
| Orbit inclination | 51.60 deg |
| Orbit perigee | 50.00 NM |
| Orbit apogee | 100.00 NM |

The vehicle mission is defined in this first block of user input data.

The first entry is the mission payload (Section 2.18). The sizing tool model assumes the payload is both carried up to orbit and landed.

The next two entries are the number of crew on the vehicle (Sections 2.2.1 and 2.11) and the pressurized structural volume on the vehicle (Section 2.11).

The next three entries are the number of days on-orbit, the average on-orbit power requirement and the average on-orbit heat rejection requirement. (Sections 2.8, 2.11). These three entries size the fuel cells supplying the on-orbit power, the mass of the fuel cell reactants and reactant tanks required to supply this amount of energy and the heat rejection system mass required to dissipate the waste heat. The reason the average heat rejection requirement is larger than the average power usage is there will be waste heat from the fuel cells in generating the required amount of power, and the crew (for crewed missions only) and possibly the payload will also be generating waste heat that needs to be removed.

The next entry is the maximum axial acceleration of the vehicle during ascent. This is one of the vehicle ascent trajectory constraints used in the trajectory analysis to verify the estimate of the mission velocity required to reach orbit.

The next entry is the maximum normal acceleration the vehicle will see in horizontal flight during reentry and landing. It is used in sizing the wing mass (Section 2.1.1) and the body structure unit mass requirements (Section 2.2.2).

The next entry is the vehicle safety factor. The safety factor is an allowance for the vehicle seeing larger than the design loads during the vehicle's operational life. The safety factor increases the vehicles loads and therefore the vehicle mass. The safety factor is applied to the wing's mass (Section 2.1.1), the vehicle's body unit mass (Section 2.2.2) and the propellant tank wall thickness (Sections 2.2.3).

The last three entries are the main engine cut off (MECO) orbit conditions of inclination, perigee and apogee. This information is used by the mission velocity requirements section of the performance spreadsheet to estimate the mission velocity required to reach the mission orbit and is also used in the trajectory analysis to verify the mission velocity requirements. The orbital maneuvering system (OMS) velocity budget (Section 2.14.3) includes allowances for the on-orbit transfer from the MECO orbit to the mission orbit, the de-orbit burn and any other on-orbit maneuvering burns required.

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank |
|---|---|---|---|
| Position= | 3 | 1 | 2 |
| Ullage= | 0.0500 | 0.0500 | 0.0500 |
| Density= | 71.20 | 50.50 | 4.43 lbm/ft^3 |
| Residual A (propellant mass)= | 0.0038 | 0.0038 | 0.0016 lbm/lbm |
| Residual B (engine thrust)= | 0.0010 | 0.0010 | 0.0012 lbm/lbf |
| Ullage pressure= | 35.00 | 35.00 | 50.00 psi |
| TPS unit mass= | 0.250 | 0.000 | 0.250 lbm/ft^2 |
| | Fwd Tank | Mid Tank | Aft Tank |
| Forward endcap height coefficient= | 0.7071 | 0.7071 | 0.3300 |
| Aft endcap height coefficient= | 0.7071 | 0.7071 | 0.3300 |
| Tank design= | 3 | 3 | 3 |
| Forward endcap flag= | 1 | 1 | 1 |
| Aft endcap flag= | 1 | 1 | 1 |

Propellant tank data is defined in this block of user input data. The first section refers to the oxidizer tank and the fuel one and fuel two tanks. The last section refers to the forward, mid and aft propellant tanks. The fuel one tank and the mid tank do not apply to a bipropellant vehicle configuration. The oxidizer in this table is liquid oxygen. The fuel one in this example is kerosene. The fuel two in this example is liquid hydrogen.

The first entry is the propellant tank position flag (Section 2.2.3). The options are the propellant tank is in the forward position, the middle position or the aft position.

The next entry is the propellant tank ullage factor (Section 2.2.3). This factor increases the volume in the propellant tank to account for the ullage space required for the pressurization of the tank, the propellant burned by the vehicle's main engines prior to liftoff, and the volume of the residual ascent main propellant.

The next entry is the propellant density (Section 2.2.3).

The next two entries are ascent residual propellant coefficients (Section 2.14.2). The first coefficient is a factor for the ascent propellant mass. The second coefficient is a factor for the main engine vacuum thrust.

The next entry is the propellant tank ullage pressure (Section 2.2.3).

The next entry is the unit mass of the propellant tank's cryogenic insulation (Section 2.3.3). A room temperature fluid, such as kerosene, does not require cryogenic insulation.

The next two entries are the forward and aft endcap height coefficients (Section 2.2.3). Only elliptical endcaps require inputs in this section. Endcap heights for hemispherical endcaps and toroidal endcaps are hardwired into the program.

The next entry is the tank design flag (Section 2.2.3). The options are a common bulkhead propellant tank design, a nested bulkhead propellant tank design, or a separate propellant tank design. If a nested bulkhead design is selected for a pair of tanks, the aft tank will have a concave forward endcap. If a common bulkhead is selected for a pair of tanks, the aft tank will not have a forward endcap. For both these cases the forward tank of this pair will have a convex aft endcap.

The last two entries are the propellant tank forward and aft endcap flags (Section 2.2.3). The options are an ellipsoidal endcap, a hemispherical endcap or a toroidal endcap.

| Vehicle Materials: | Fwd Tank | Mid Tank | Aft Tank |
|---|---|---|---|
| Density= | 0.098 | 0.057 | 0.098 lbm/ft^3 |
| Ftu= | 65,600 | 90,400 | 65,600 psi |

This block of user input data contains the propellant tank material properties used in calculating the propellant tank masses (Section 2.2.3). The middle tank is not used in a bipropellant vehicle configuration. The first entry is the density of the material used. The last entry is the ultimate strength of the material used. In the Access to Space Option 3 final report, the materials strength was reduced by 20% to account for fatigue. A similar knockdown over the handbook material properties is recommended.

| Mode 2 burn: | |
|---|---|
| Mode 2 mission velocity | 20,130 ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 sec |
| Iv2 | 452.70 sec |
| Mixture ratio (% Oxidizer) | 85.70 % |
| Mixture ratio (% Fuel 1) | 0.00 % |
| Mixture ratio (% Fuel 2) | 14.30 % |
| Engine height | 11.66 ft |
| Number of engines | 6 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine vac thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 40.26 lbf/lbm (vac) |
| No 2 (if engine flag= 2) | 1.400 g |

This block of user input data defines the Mode 2 engines.

The first entry is Mode 2 mission velocity. This parameter is varied by the user to find the best velocity split between Modes 1 and 2.

The next entry is the Mode 2 sea level specific impulse. This parameter is used only if burn flag (Section 2.5.1) is set to two.

The next entry is the Mode 2 engine vacuum specific impulse.

The next three entries are the Mode 2 mixture ratios. The sum of these entries is 100%. For a bipropellant vehicle configuration, the fuel one mixture ratio is set to zero.

The next entry is the engine height. This parameter will need to be changed to reflect the changes in engine height as the engine thrust is changed in a vehicle configuration optimization. The winged body vehicle configuration is not sensitive to this parameter.

The next entry is the number of engines. This parameter is used in the thrust structure mass calculation (Section 2.5.4).

The next entry is the engine flag (Section 2.5.1) definition.

The next two entries are used if the engine flag is set to one. The first parameter is the mass of one engine. The second parameter is the vacuum thrust of one engine.

The last two entries are used if the engine flag is set to two. The first parameter is the engine vacuum thrust-to-weight ratio. The second parameter is the Mode 2 initial thrust-to-weight ratio.

| Mode 1 burn: | |
|---|---|
| Isl1 | 333.50 sec |
| Iv1 | 385.10 sec |
| Mixture ratio (% Oxidizer) | 76.80 % |
| Mixture ratio (% Fuel 1) | 20.20 % |
| Mixture ratio (% Fuel 2) | 3.00 % |
| Engine height | 11.66 ft |
| Number of engines | 6 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine sl thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 82.90 lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 g |

This block of user input data defines the Mode 1 engines.

The first entry is the Mode 1 sea level specific impulse.

The next entry is the Mode 2 engine vacuum specific impulse.

The next three entries are the Mode 1 mixture ratios. The sum of these entries is 100%. For a bipropellant vehicle configuration, the fuel one mixture ratio is set to zero.

The next entry is the engine height. This parameter will need to be changed to reflect the changes in engine height as the engine thrust is changed in a vehicle configuration optimization. The winged body vehicle configuration is not sensitive to this parameter.

The next entry is the number of engines. This parameter is used in the thrust structure mass calculation (Section 2.5.4).

The next entry is the engine flag (Section 2.5.1) definition.

The next two entries are used if the engine flag is set to one. The first parameter is the mass of one engine. The second parameter is the sea level thrust of one engine.

The last two entries are used if the engine flag is set to two. The first parameter is the engine sea level thrust-to-weight ratio. The second parameter is the Mode 1 initial thrust-to-weight ratio.

| | |
|---|---|
| Burn flag= | 3 |
| Number of fuels used= | 2 |
| Payload bay location flag= | 2 |

The next block of user input data contains several of the vehicle configuration flags. The first entry is the burn flag (Section 2.5.1). The next entry is the number of fuel tanks. For a bipropellant vehicle configuration, there will be one fuel tank. For a tripropellant vehicle configuration, there will be two fuel tanks. The last entry is the payload bay location flag (Section 2.2.3).

| Vehicle sizing coefficients: | |
|---|---|
| Main propellant feed line and press sys= | 55.00 lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 |
| Avionics= | 710 lbm/lbm^(1/8) |
| Range safety= | 0 lbm |
| Tip Fin Constant= | 1.00 lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 lbm/ft^2 (if hot structure is selected) |
| | |
| Base engine heat shield unit mass= | 1.64 lbm/ft^2 |
| | |
| Gimbal actuator unit mass= | 0 lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 lbm/lbf |
| | |
| Prime Power (PWc) (aero surface)= | 0.274 lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface)= | 0.000 lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 lbm/kw-day |
| | |
| ECLSS crew cabin constant (Ec)= | 0.00 lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200 lbm/kw |
| | |
| Personnel waste systems (PPf)= | 0.00 lbm |
| Personnel seats and crew related (PPs)= | 0.00 lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 lbm |
| Personnel mass (Pp)= | 0.00 lbm/crew member |
| | |
| Body flap constant (Bbf)= | 1.17 lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 lbm/ft^2 |
| Control surface miscellaneous hardware (Spc)= | 200 lbm |
| | |
| Payload bay mass= | 5,786 lbm (from Langley SSTO(R) RD-701 case) |
| Payload bay diameter= | 15.00 ft |
| Vehicle base diameter= | 29.80 ft |
| Vehicle cone angle= | 0.00 deg (vehicle is a cylinder) |

The next block of user input data are the coefficients used in most of the vehicle subsystem mass calculations.

The first entry is the coefficient used in the calculation of the main propellant feed line and pressurization system mass (Section 2.5.2).

The next entry is the coefficient used in calculating the vehicle dry mass contingency factor (Section 2.12). The contingency factor is an allowance for an increase in vehicle mass during the program.

The next entry is the coefficient used in calculating the avionics system mass (Section 2.10).

The next entry is the coefficient used in calculating the range safety system mass (Section 2.2.6). A range safety system is used to destroy a launch vehicle if it approaches the sides of the firing range. Reusable vehicles may or may not have range safety systems.

The next entry is the coefficient used in calculating the tip fin mass (Section 2.1.2). The tip fins provide vehicle control after reentry to landing. Tip fins are smaller and lighter than a conventional rudder. This program assumes there will be two tip fins mounted on the outboard edges of the wings.

The next entry is the coefficient used in calculating the body wetted area unit mass (Section 2.2.2)

The next entry is the coefficient used in calculating the crew cabin mass (Section 2.2.1).

The next entry is the coefficient used in calculating the landing gear mass (Section 2.4).

The next entry is the coefficient used in calculating the body insulation mass (Section 2.3.2). The need for body insulation is a function of the thermal protection system (TPS) used on the vehicle. If the body insulation is used, it is assumed to cover the fuselage wetted area. A hot structure design will probably require body insulation.

The next entry is the coefficient used in calculating the vehicle engine bay heat shield (Section 2.2.5). This engine bay heat shield keeps the hot engine plume gas out of the engine bay. If the vehicle configuration uses an engine configuration, such as a plug nozzle, that keeps engine recirculating gas out of the engine bay, this system will not be needed.

The next entry is the coefficient used in calculating the main engine gimbal actuator mass (Section 2.5.3). This calculation is only the gimbal actuator mass. The power system mass used to run the main engine gimbals is calculated elsewhere.

The next two entries are the coefficients used in calculating the main engine thrust structure mass (Section 2.5.4). The first coefficient factors in the main engine vacuum thrust. The second coefficient factors in the number of engines used.

The next three entries are coefficients used in calculating the vehicle's prime power system mass (Section 2.8) for the power requirements during ascent to orbit and reentry from orbit. These three coefficients are used to calculate the masses of the power systems required to move the aero surfaces, to gimbal the main engines and to power the avionics system. It may be possible for the on-orbit power systems to supply some of the ascent and reentry power demands.

The next three entries are coefficients used in calculating the power conversion and distribution system mass (Section 2.9). The first entry is used in calculating the electrical power distribution system mass. The next two coefficients are used to calculate the hydraulics power distribution systems mass used to move the aero surfaces and to gimbal the main engines. The last two entries are used only if hydraulic systems are used to move the aero surfaces and to gimbal the main engines.

The next two entries are coefficients used in calculating the vehicle's prime power system mass (Section 2.8) for the on-orbit power requirements. The assumption that is made here is that the fuel cells are used for the average on-orbit power requirement. The first coefficient is the power source (fuel cell power stack) unit mass and the second coefficient is the energy supply (fuel cell reactants and tankage) unit mass. It is assumed that the fuel cells will be able to supply the short term peak power requirements. It may be possible for these fuel cells to provide some of the ascent and reentry power requirements. Power and energy redundancy requirements may require a proportional increase in the value of these coefficients.

The next four entries are coefficients used in calculating the vehicle's environmental control and life support system (ECLSS) mass (Section 2.11). The first entry is the coefficient used in calculating the crew cabin mass. The second entry is the crew supplies unit mass. The third entry is the avionics system waste heat removal unit mass. The last entry is the unit mass of the vehicle on-orbit heat rejection system.

The next two entries are the coefficients used in calculating the personnel provisions mass (Section 2.16). The first entry is for the food waste and water management system mass. The second entry is the unit mass of the crew seats and other related items.

The next two entries are the coefficients used in calculating the personnel mass (Section 2.17). The first entry is the miscellaneous crew mass. The second entry is the crew personnel unit mass.

The next three entries are used to calculate the mass of the body flaps and of the actuators used to move the body flaps. The first coefficient is used in calculating the mass of the body flaps (Section 2.1.3). The second and third entries are used in calculating mass of the control surface actuators (Section 2.1.4). The body flaps provide vehicle control during reentry to landing. They also shield the vehicle's main engines from the reentry heat loads.

The next entry is the payload bay mass (Section 2.2.4). This entry includes the mass of the structure required to support the payload and to distribute the loads from the payload to the rest of the vehicle.

The next entry is the payload bay diameter.

The last two entries in this block of user input data are the vehicle base diameter and the vehicle cone angle. The vehicle diameter is the primary geometry sizing factor for this vehicle configuration (Section 1.2.1). The vehicle cone angle is the angle between vertical and the side of the vehicle. Although this program has the capability of a non-zero vehicle cone angle, winged vehicles are usually cylinders and therefore this parameter will usually be set to zero.

| OMS and RCS systems mass coefficients: | | |
|---|---|---|
| RCS system mass coefficient= | 0.000151 | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 | sec |
| RCS thruster specific impulse (reentry)= | 410.00 | sec |
| RCS thruster specific impulse (ascent)= | 350.00 | sec |
| OMS system thrust-to-weight= | 0.04 | g |
| OMS engine mass coefficient= | 0.035 | lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 | lbm/lbm |
| OMS thruster specific impulse= | 462 | sec |

The next block of user input data is used to define the orbital maneuvering system (OMS) and the reaction control system (RCS) dry mass (Sections 2.6 and 2.7 respectively).

The first entry is the RCS mass coefficient used in calculating the RCS mass. The next three entries are the specific impulses used for any on-orbit, entry and ascent RCS engine burns.

The next three entries are the coefficients used in calculating the OMS engine and propellant tankage mass. This program assumes that the OMS propellant tanks are sized to hold all of the OMS and RCS propellant.

The last entry in this data block is the OMS engine on-orbit specific impulse.

| The main propulsion system additional delta v may be used for additional on-orbit maneuvers and to adjust the mission velocity requirements based on trajectory. analysis results. The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential throttling is selected for vehicle thrust vector control. | |
|---|---|
| RCS ascent mission velocity (applied to GLOW)= | 0 ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1140 ft/sec |
| Main propulsion system additional dv (applied to MECO)= | 29 ft/sec |

The next block of user input data is used to calculate the amount of OMS and RCS propellant required to fly the design mission. This propellant requirement is based on the assumptions that the vehicle lands with the design payload and that the residual ascent propellant is vented prior to the on-orbit OMS and RCS burns. The program also assumes the residual OMS and RCS propellants are accounted for by an increase in the OMS and RCS on-orbit velocity budgets.

The first entry is the ascent RCS velocity requirement. The roll requirements on current launch vehicles including rolling the vehicle into the proper heading shortly after launch and controlling the vehicle's roll during ascent. Launch vehicles with multiple bell engines that use engine gimbaling for thrust vector control will not need to use the RCS thrusters for roll control. Launch vehicles with a single bell engine will need to use the RCS thrusters for roll control. Launch vehicles with multiple bell engines or a single plug nozzle that uses differential throttling for thrust vector control may or may not need to use the RCS thrusters for roll control. This program assumes that ascent RCS velocity requirement is applied to the vehicle gross lift off weight (GLOW).

The next entry is the RCS on-orbit velocity requirement.

The next entry is the RCS reentry velocity requirement. This program assumes that the RCS reentry burn occurs after the on-orbit OMS and RCS burns.

The next entry is the OMS on-orbit velocity requirement.

After a new launch vehicle configuration has been developed, a trajectory analysis should be performed to find the actual mission velocity requirement. The last entry in this block of user input data is where a delta to the mission velocity estimate calculated in the performance spreadsheet is entered into the program. This entry can also be used to enter into the program any main propulsion system velocity requirements beyond those of the main engine cutoff conditions (MECO).

| | |
|---|---|
| Vehicle layout: | |
| Theoretical Wing Loading= | 60 lbm/ft^2 |
| Wing Planform Ratio (Sexp/Sref)= | 0.54 |
| Ratio of Exposed Wing Wetted Area/Planform Area= | 2.064 |
| Cord Thickness Ratio (Height/Cord Length)= | 0.20 |
| Ratio Body Flap Width/Diameter= | 0.25 |
| Ratio Tip Fin/Wing Planform Area= | 0.17 |
| Wing Surface Area Distribution: | |
| Leading Edge= | 0.10 |
| Elevon= | 0.15 |
| Windward Side= | 0.35 |
| Leeward Side= | 0.40 |
| | |
| Wing/Body efficiency factor (f)= | 0.1500 |
| Wing Carry through Constant (Wc)= | 0.0267 |
| Exposed wing Material/Configuration Constant (Wm)= | 0.2140 |
| Exposed Wing Aspect Ratio= | 1.7800 |
| Exposed Wing Taper Ratio= | 0.2360 |
| Body Carry Through Ratio (Carry Through Width/Body Width)= | 0.8230 |

The next block of user input data is used to define the vehicle configuration wings (Section 2.1.1). Unless trade studies are being run on the vehicle wing design, these parameters are fixed once the configuration has been defined.

The first entry is the vehicle wing loading based on the theoretical wing planform area. The theoretical wing planform area (Sref) includes the exposed wing planform area and the wing planform area buried inside the vehicle body. The exposed wing planform area is the part of the wing that extends beyond the vehicle body. This parameter is a major factor on the vehicle reentry environment, landing speed and dry mass.

The next entry is used to calculate the exposed wing planform area. It is the ratio of the exposed wing planform area (Sexp) divided by the theoretical wing planform area (Sref).

The next entry is used to calculate the wetted wing surface area. It is the ratio of the exposed wing wetted area divided by the exposed wing planform area.

The next entry is the ratio of the airfoil cord height divided by the cord length.

The next entry is the body flap length to diameter ratio. The program assumes the body flap width is the same as the vehicle width. Therefore, this parameter is used to define the body flap surface area required to control the vehicle during the reentry and landing mission phases. Since the body flap is also used to shield the main engines from the reentry heat loads, the body flap length calculated here is compared to the engine length and the larger value is used in the calculation of the body flap area.

The next entry is the ratio of tip fin area divided by the exposed wing planform area. The program assumes there are two tip fins mounted on the wing tips. These tip fins are also assumed to be aerodynamic control surfaces and therefore feed into the aerodynamic control surface power and actuator mass calculations.

The next four entries split the wing wetted surface area into zones for the TPS mass calculations. The first entry is the leading edge fraction of the wing wetted surface area. The next entry is the elevon fraction of the wing wetted surface area. The next entry is the windward fraction of the wing wetted surface area. The next entry is the leeward fraction of the wing wetted surface area.

The next entry is the wing/body efficiency factor. This parameter is the fraction of the total vehicle lift that is provided by the vehicle body.

The next entry is the wing carry through coefficient. This parameter is used to find the wing carry through structure mass.

The next entry is a coefficient for the wing material and method of construction used.

The next two entries are wing geometry parameters. The first entry is the exposed wing aspect ratio. The second entry is the exposed wing taper ratio.

The last entry in this block of user input data is the body carry through ratio. This parameter is the body width where the wing enters the body divided by the body diameter.

| Nose definition: | |
|---|---|
| Exterior angle 1= | 17 deg |
| Exterior angle 2= | 30 deg |
| Ratio r2/r1= | 0.8 |
| Ratio r3/r1= | 0.2 |
| Nose unit weight= | 0.25 lbm/ft^2 |

The next section of user input data is used to define the vehicle configuration's biconic nose. A biconic nose cone reduces the length and therefore the surface area and mass of the nose cone. The biconic nose cone is modeled as a lower cone, an upper cone and a hemispherical tip. The first entry is the angle between vertical and the lower cone's side. The second entry is the angle between the vertical and the upper cone's side. The third entry is the ratio in the upper and lower radii for the lower cone. The last entry is the ratio in the upper and lower radii for the upper cone. The lower cone's base diameter is the vehicle diameter at the base of the nose. This cone's upper diameter is found from the lower diameter and the ratio between the upper and lower radii. The exterior angle then defines the cone's geometry. The base diameter of the upper cone is the upper surface of the lower cone. This cone's geometry is likewise defined by the exterior angle and the ratio of radii.

| TPS unit Masses: | | |
|---|---|---|
| Nose= | 2.20 | lbm/ft^2 |
| Windward Fwd tank= | 0.90 | lbm/ft^2 |
| Leeward Fwd tank= | 0.40 | lbm/ft^2 |
| Windward Fwd Intertank= | 0.90 | lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 | lbm/ft^2 |
| Windward Mid tank= | 0.90 | lbm/ft^2 |
| Leeward Mid tank= | 0.40 | lbm/ft^2 |
| Windward Aft Intertank= | 0.90 | lbm/ft^2 |
| Leeward Aft Intertank= | 0.40 | lbm/ft^2 |
| Windward Aft tank= | 0.90 | lbm/ft^2 |
| Leeward Aft tank= | 0.40 | lbm/ft^2 |
| Windward Aft Skirt= | 0.90 | lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 | lbm/ft^2 |
| Body Flaps= | 2.00 | lbm/ft^2 |
| Tip Fin= | 2.00 | lbm/ft^2 |
| Elevon= | 2.00 | lbm/ft^2 |
| Wing Leading Edge= | 2.00 | lbm/ft^2 |
| Wing Leeward Side= | 0.40 | lbm/ft^2 |
| Wing Windward Side= | 1.30 | lbm/ft^2 |

The last block of user input data contains the TPS unit masses (Section 2.3.1) for the vehicle body sectors. These sectors are the windward and leeward sides of the vehicle body and wing, the nose, the forward tank barrel section, the forward intertank, the mid tank barrel section, the aft intertank, the aft tank barrel section, the aft skirt, the aerodynamic control surface and the wing leading edge. An accurate choice of the TPS unit masses will require knowledge of the distribution of heat loads and the resulting temperatures on the vehicle body. This information can be generated by doing a thermal map of the vehicle body for the ascent and reentry trajectories at the moment of the peak heat loads.

### 1.5.3 Lifting Body VTHL Launch Vehicle Configuration

The following 13 blocks of user input data are from the RD-701 input data file for the lifting body VTHL launch vehicle configuration sizing tool model. This input data is also in the input data section of the sizing tool performance spreadsheet. After each block of information, an explanation of the inputs will be given with cross references back to the equation descriptions in section 2 where appropriate.

| Input Data: | |
|---|---|
| Payload (Wpay) | 25,000 lbm |
| Number of crew | 0 |
| Crew cabin volume | 0 ft^3 |
| Number of days on-orbit | 7 |
| Average on-orbit power usage | 5 kw |
| Average on-orbit heat rejection requirement | 10 kw |
| Maximum acceleration (No) | 3.000 g |
| Maximum normal acceleration (Nz) | 1.600 g |
| Factor of safety | 1.40 |
| Orbit inclination | 51.60 deg |
| Orbit perigee | 50.00 NM |
| Orbit apogee | 100.00 NM |

The vehicle mission is defined in this first block of user input data.

The first entry is the mission payload (Section 2.18). The sizing tool model assumes the payload is both carried up to orbit and landed.

The next two entries are the number of crew on the vehicle (Sections 2.2.1 and 2.11) and the pressurized structural volume on the vehicle (Section 2.11).

The next three entries are the number of days on-orbit, the average on-orbit power requirement and the average on-orbit heat rejection requirement (Sections 2.8, 2.11). These three entries size the fuel cells supplying the on-orbit power, the mass of the fuel cell reactants and reactant tanks required to supply this amount of energy, and the heat rejection system mass required to dissipate the waste heat. The reason the average heat rejection requirement is larger than the average power usage is there will be waste heat from the fuel cells in generating the required amount of power and the crew (for crewed missions only) and possibly the payload will also be generating waste heat that needs to be removed.

The next entry is the maximum axial acceleration of the vehicle during ascent. This is one of the vehicle ascent trajectory constraints used in the trajectory analysis to verify the estimate the mission velocity required to reach orbit.

The next entry is the maximum normal acceleration the vehicle will see in horizontal flight during reentry and landing. It is used in sizing the body structure unit mass requirements (Section 2.2.2).

The next entry is the vehicle safety factor. The safety factor is an allowance for the vehicle seeing larger than the design loads during the vehicle's operational life. The safety factor increases the vehicle loads and therefore the vehicle mass. The safety factor is applied to the vehicle body unit mass (Section 2.2.2) and the propellant tank wall thickness (Section 2.2.3).

The last three entries are the main engine cut off (MECO) orbit conditions of inclination, perigee and apogee. This information is used by the mission velocity requirements section of the performance spreadsheet to estimate the mission velocity required to reach the mission orbit and is also used in the trajectory analysis to verify the mission velocity requirements. The orbital maneuvering system (OMS) velocity budget (Section 2.14.3) includes allowances for the on-orbit transfer from the MECO orbit to the mission orbit, the de-orbit burn and any other on-orbit maneuvering burns required.

| Main Propulsion: | | | | | | |
|---|---|---|---|---|---|---|
| | | | Ox tank | Fuel 1 tank | Fuel 2 tank | |
| Ullage= | | | 0.05 | 0.05 | 0.05 | |
| Density= | | | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Residual A (propellant mass)= | | | 0.0038 | 0.0038 | 0.0016 | lbm/lbm |
| Residual B (engine thrust)= | | | 0.001 | 0.001 | 0.0012 | lbm/lbf |
| Ullage pressure= | | | 20.00 | 20.00 | 20.00 | psi |
| TPS unit mass= | | | 0.250 | 0.000 | 0.250 | lbm/ft^2 |

The oxidizer, fuel one and fuel two propellant tank data are defined in this block of user input data. The fuel one tank does not apply to a bipropellant vehicle configuration. The oxidizer in this table is liquid oxygen. The fuel one in this example is kerosene. The fuel two in this example is liquid hydrogen.

The first entry is the propellant tank ullage factor (Section 2.2.3). This factor increases the volume in the propellant tank to account for the ullage space required for the pressurization of the tank, the propellant burned by the vehicle main engines prior to liftoff and the volume of the residual ascent main propellant.

The next entry is the propellant density (Section 2.2.3).

The next two entries are ascent residual propellant coefficients (Section 2.14.2). The first coefficient is a factor for the ascent propellant mass. The second coefficient is a factor for the main engine vacuum thrust.

The last entry is the propellant tank ullage pressure (Section 2.2.3).

| Vehicle Materials: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Density= | 0.098 | 0.098 | 0.057 | lbm/ft^3 |
| Ftu= | 65,600 | 65,600 | 90,400 | psi |

This block of user input data contains the propellant tank material properties used in calculating the propellant tanks masses (Section 2.2.3). The middle tank is not used in a bipropellant vehicle configuration. The first entry is the density of the material used. The last entry is the ultimate strength of the material used. In the Access to Space Option 3 final report, the materials strength was reduced by 20% to account for fatigue. A similar knockdown over the handbook material properties is recommended.

| Mode 2 burn: | |
|---|---|
| Mode 2 mission velocity | 19,830 ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 sec |
| Iv2 | 452.70 sec |
| Mixture ratio (% Oxidizer) | 85.70 % |
| Mixture ratio (% Fuel 1) | 0.00 % |
| Mixture ratio (% Fuel 2) | 14.30 % |
| Engine height | 13.26 ft |
| Number of engines | 5 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine vac thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 40.26 lbf/lbm (vac) |
| No2 (if engine flag= 2) | 1.400 g |

This block of user input data defines the Mode 2 engines.

The first entry is the Mode 2 mission velocity. This parameter is varied by the user to find the best velocity split between Modes 1 and 2.

The next entry is the Mode 2 sea level specific impulse. This parameter is used only if burn flag (Section 2.5.1) is set to two.

The next entry is the Mode 2 engine vacuum specific impulse.

The next three entries are the Mode 2 mixture ratios. The sum of these entries is 100%. For a bipropellant vehicle configuration, the fuel one mixture ratio is set to zero.

The next entry is the engine height. This parameter will need to be changed to reflect the changes in engine height as the engine thrust is changed in a vehicle configuration optimization. The lifting body vehicle configuration is sensitive to this parameter.

The next entry is the number of engines. This parameter is used in the thrust structure mass calculation (Section 2.5.4).

The next entry is the engine flag (Section 2.5.1) definition.

The next two entries are used if the engine flag is set to one. The first parameter is the mass of one engine. The second parameter is the vacuum thrust of one engine.

The last two entries are used if the engine flag is set to two. The first parameter is the engine vacuum thrust-to-weight ratio. The second parameter is the Mode 2 initial thrust-to-weight ratio.

| Mode 1 burn: | |
|---|---|
| Isl1 | 333.50 sec |
| Iv1 | 385.10 sec |
| Mixture ratio (% Oxidizer) | 76.80 % |
| Mixture ratio (% Fuel 1) | 20.20 % |
| Mixture ratio (% Fuel 2) | 3.00 % |
| Engine height | 13.26 ft |
| Number of engines | 5 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine sl thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 82.9 lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 g |

This block of user input data defines the Mode 1 engines.

The first entry is the Mode 1 sea level specific impulse.

The next entry is the Mode 1 engine vacuum specific impulse.

The next three entries are the Mode 1 mixture ratios. The sum of these entries is 100%. For a bipropellant vehicle configuration, the fuel one mixture ratio is set to zero.

The next entry is the engine height. This parameter will need to be changed to reflect the changes in engine height as the engine thrust is changed in a vehicle configuration optimization. The lifting body vehicle configuration is sensitive to this parameter.

The next entry is the number of engines. This parameter is used in the thrust structure mass calculation (Section 2.5.4).

The next entry is the engine flag (Section 2.5.1) definition.

The next two entries are used if the engine flag is set to one. The first parameter is the mass of one engine. The second parameter is the sea level thrust of one engine.

The last two entries are used if the engine flag is set to two. The first parameter is the engine sea level thrust-to-weight ratio. The second parameter is the Mode 1 initial thrust-to-weight ratio.

| | |
|---|---|
| Burn flag= | 3 |
| Number of fuels used= | 2 |

The next block of user input data contains several of the vehicle configuration flags. The first entry is the burn flag (Section 2.5.1). The last entry is the number of fuel tanks. For a bipropellant vehicle configuration, there will be one fuel tank. For a tripropellant vehicle configuration, there will be two fuel tanks.

| | | |
|---|---|---|
| Vehicle sizing coefficients: | | |
| Main propellant feed line and press sys= | 55.00 | lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm/lbm^(1/8) |
| Range safety= | 0.00 | lbm |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 | lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 | lbm/ft^2 (if hot structure is selected) |
| Base engine heat shield unit mass= | 1.64 | lbm/ft^2 |
| Gimbal actuator unit mass= | 0 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface) | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| ECLSS crew cabin constant (Ec)= | 0.00 | lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 | lbm/kw |
| Personnel waste systems (PPf)= | 0.00 | lbm |
| Personnel seats and crew related (PPs)= | 0.00 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| Control surface constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Control surface miscellaneous hardware (Spc)= | 200 | lbm |
| Payload bay mass= | 3,925 | lbm(from Langley SSTO(R) RD-701 case) |

The next block of user input data are the coefficients used in most of the vehicle subsystem mass calculations.

The first entry is the coefficient used in the calculation of the main propellant feed line and pressurization system mass (Section 2.5.2).

The next entry is the coefficient used in calculating the vehicle dry mass contingency factor (Section 2.12). The contingency factor is an allowance for an increase in vehicle mass during the program.

The next entry is the coefficient used in calculating the avionics system mass (Section 2.10).

The next entry is the coefficient used in calculating the range safety system mass (Section 2.2.6). A range safety system is used to destroy a launch vehicle if it approaches the sides of the firing range. Reusable vehicles may or may not have range safety systems.

The next entry is the coefficient used in calculating the tip fin mass (Section 2.1.2). The tip fins provide vehicle control after reentry to landing. Tip fins are smaller and lighter than a conventional rudder. This program assumes there will be two tip fins mounted on the outboard edges of the wings.

The next entry is the coefficient used in calculating the body wetted area unit mass (Section 2.2.2)

The next entry is the coefficient used in calculating the crew cabin mass (Section 2.2.1).

The next entry is the coefficient used in calculating the landing gear mass (Section 2.4).

The next entry is the coefficient used in calculating the body insulation mass (Section 2.3.2). The need for body insulation is a function of the thermal protection system (TPS) used on the vehicle. If the body insulation is used, it is assumed to cover the fuselage wetted area. A hot structure design will probably require body insulation.

The next entry is the coefficient used in calculating the vehicle engine bay heat shield (Section 2.2.5). This engine bay heat shield keeps the hot engine plume gas out of the engine bay. If the vehicle configuration uses an engine configuration, such as a plug nozzle, that keeps engine recirculating gas out of the engine bay, this system will not be needed.

The next entry is the coefficient used in calculating the main engine gimbal actuator mass (Section 2.5.3). This calculation is only the gimbal actuator mass. The power system mass used to run the main engine gimbals is calculated elsewhere.

The next two entries are the coefficients used in calculating the main engine thrust structure mass (Section 2.5.4). The first coefficient factors in the main engine vacuum thrust. The second coefficient factors in the number of engines used.

The next three entries are coefficients used in calculating the vehicle's prime power system mass (Section 2.8) for the power requirements during ascent to orbit and reentry from orbit. These three coefficients are used to calculate the masses of the power systems required to move the aero surfaces, to gimbal the main engines, and to power the avionics system. It may be possible for the on-orbit power systems to supply some of the ascent and reentry power demands.

The next three entries are coefficients used in calculating the power conversion and distribution system mass (Section 2.9). The first entry is used in calculating the electrical power distribution system mass. The next two coefficients are used to calculate the hydraulics power distribution systems mass used to move the aero surfaces and to gimbal the main engines. These last two entries are used only if hydraulic systems are used to move the aero surfaces and to gimbal the main engines.

The next two entries are coefficients used in calculating the vehicle's prime power system mass (Section 2.8) for the on-orbit power requirements. The assumption that is made here is that the fuel cells are used for the average on-orbit power requirement. The first coefficient is the power source (fuel cell power stack) unit mass and the second coefficient is the energy supply (fuel cell reactants and tankage) unit mass. It is assumed that the fuel cells will be able to supply the short term peak power requirements. It may be possible for these fuel cells to provide some of the ascent and reentry power requirements. Power and energy redundancy requirements may require a proportional increase in the value of these coefficients.

The next four entries are coefficients used in calculating the vehicle's environmental control and life support system (ECLSS) mass (Section 2.11). The first entry is the coefficient used in calculating the crew cabin mass. The second entry is the crew supplies unit mass. The third entry is the avionics system waste heat removal unit mass. The last entry is the unit mass of the vehicle on-orbit heat rejection system.

The next two entries are the coefficients used in calculating the personnel provisions mass (Section 2.16). The first entry is for the food waste and water management system mass. The second entry is the unit mass of the crew seats and other related items.

The next two entries are the coefficients used in calculating the personnel mass (Section 2.17). The first entry is the miscellaneous crew mass. The second entry is the crew personnel unit mass.

The next three entries are used to calculate the mass of the control surfaces and of the actuators used to move these control surfaces. The first coefficient is used in calculating the mass of the control surfaces (Section 2.1.3). The second and third entries are used in calculating mass of the control surface actuators (Section 2.1.4). These control surfaces provide vehicle control after reentry to landing. The body flaps, which are one of these control surfaces, also shield the vehicle main engines from the reentry heat loads.

The last entry in this block of user input data is the payload bay mass (Section 2.2.4). This entry includes the mass of the structure required to support the payload and to distribute the loads from the payload to the rest of the vehicle.

| OMS and RCS systems mass coefficients: | |
|---|---|
| RCS system mass coefficient= | 0.000151 lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 sec |
| RCS thruster specific impulse (reentry)= | 410.00 sec |
| RCS thruster specific impulse (ascent)= | 350.00 sec |
| | |
| OMS system thrust-to-weight= | 0.04 g |
| OMS engine mass coefficient= | 0.035 lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 lbm/lbm |
| OMS thruster specific impulse= | 462 sec |

The next block of user input data is used to define the orbital maneuvering system (OMS) and the reaction control system (RCS) dry mass (Sections 2.6 and 2.7 respectively).

The first entry is the RCS mass coefficient used in calculating the RCS mass. The next three entries are the specific impulses used for any on-orbit, entry, and ascent RCS engine burns.

The next three entries are the coefficients used in calculating the OMS engine and propellant tankage mass. This program assumes that the OMS propellant tanks are sized to hold all of the OMS and RCS propellant.

The last entry in this data block is the OMS engine on-orbit specific impulse.

The main propulsion system additional delta v may be used for additional on-orbit maneuvers and to adjust the mission velocity requirements based on trajectory analysis results.
The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential throttling is selected for vehicle thrust vector control.

| | |
|---|---|
| RCS ascent mission velocity (applied to GLOW)= | 0 ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 ft/sec |
| OMS system on-orbit mission velocity (applied to MECO-residuals)= | 1140 ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -280 ft/sec |

This next block of user input data is used to calculate the amount of OMS and RCS propellant required to fly the design mission. This propellant requirement is based on the assumptions that the vehicle lands with the design payload and that the residual ascent propellant is vented prior to the on-orbit OMS and RCS burns. The program also assumes the residual OMS and RCS propellants are accounted for by an increase in the OMS and RCS on-orbit velocity budgets.

The first entry is the ascent RCS velocity requirement. The roll requirements on current launch vehicles including rolling the vehicle into the proper heading shortly after launch and controlling the vehicle roll during ascent. Launch vehicles with multiple bell engines that use engine gimbaling for thrust vector control will not need to use the RCS thrusters for roll control. Launch vehicles with a single bell engine will need to use the RCS thrusters for roll control. Launch vehicles with multiple bell engines or a single plug nozzle that use differential throttling for thrust vector control may or may not need to use the RCS thrusters for roll control. This program assumes that ascent RCS velocity requirement is applied to the vehicle gross lift off weight (GLOW).

The next entry is the RCS reentry velocity requirement. This program assumes that the RCS reentry burn occurs after the on-orbit OMS and RCS burns.

The next entry is the OMS on-orbit velocity requirement.

After a new launch vehicle configuration has been developed, a trajectory analysis should be performed to find the actual mission velocity requirement. The last entry in this block of user input data is where a delta to the mission velocity estimate calculated in the performance spreadsheet is entered into the program. This entry can also be used to enter into the program any main propulsion system velocity requirements beyond those of the main engine cutoff conditions (MECO).

| Vehicle layout: | | |
|---|---|---|
| Nose length= | 5.00 | ft |
| Oxidizer tank fwd radius= | 13.57 | ft |
| Oxidizer tank aft radius= | 13.57 | ft |
| Fuel tank half angle= | 3.95 | deg |
| Payload bay diameter= | 15.00 | ft |
| Payload bay length= | 30.00 | ft |
| Payload bay/Oxidizer tank standoff= | 5.00 | ft |
| Engine bay height= | 17.75 | ft |
| Oxidizer tank/engine standoff distance= | 10.00 | ft |
| | | |
| Crew cabin length= | 12.50 | ft |
| Crew cabin fwd width= | 10.00 | ft |
| Crew cabin aft width= | 17.00 | ft |
| Crew Cabin/payload bay standoff= | 10.00 | ft |
| | | |
| Oxidizer/Fuel 2 tank standoff= | 0.50 | ft |
| Aeroshell standoff= | 0.50 | ft |

This next block of user input data contains the parameters used to define the vehicle configuration geometry and therefore the vehicle planform area. The vehicle body wetted area is calculated from the body planform area and a set of coefficients as discussed below in a following user input data block that relates the planform area of a sector of the vehicle's body to the wetted surface area of that body sector. Some of these parameters are constant for a given vehicle configuration. Other parameters will be varied to optimize the vehicle configuration. See also Section 1.2.3 and Figure 1.2.3-2.

The first entry is the nose length. This is the distance from the tip of the vehicle to the crew cabin forward bulkhead. This is a parameter that is not likely to change after the configuration geometry has been specified.

The next two entries are oxidizer tank forward and aft radii. These two entries are major vehicle geometry optimization parameters. Changing these parameters varies the vehicle body diameter and the length of the vehicle aft section. This will then vary the fuel tank cone forward, middle, and aft radii. These two oxidizer tank radii can be independently varied. However, care must be exercised to keep the slope of the oxidizer tank barrel section and therefore the vehicle body upper and lower surfaces over the oxidizer tank reasonable.

The next entry is the fuel tank half angle. This is the half angle of the forward fuel tank cone. This is also one of the major vehicle geometry sizing parameters that will be varied to optimize the vehicle. Changing this parameter changes the fuel tank forward, mid, and aft radii.

The next two entries are the payload bay diameter and length. These parameters are not likely to change after the configuration geometry has been specified.

The next entry is the standoff distance between the aft edge of the payload bay and the forward edge of the oxidizer tank. This is a parameter that is not likely to change after the configuration geometry has been specified.

The next entry is the engine bay height. This program assumes the aft fuel tank cone half angle continues to the main engine nozzle exit plane. Specifying the engine bay height defines the fuel tank aft radius and therefore changes the fuel tank forward and mid radii. This parameter also defines the vehicle engine bay width. This is one of the vehicle geometry parameters that will be changed during vehicle optimization.

The next entry is the standoff distance from the oxidizer tank aft edge and the main engine forward edge. The vehicle thrust structure and main propellant feed system is in this space. This is a parameter that is not likely to change after the configuration geometry has been specified.

The next four entries are used to define the forward part of the vehicle between the payload bay and the vehicle nose. The crew cabin would be mounted in this section. The first three entries are the crew cabin length, the crew cabin forward width and the crew cabin aft width. The fourth entry is the standoff distance between the crew cabin aft edge and the payload bay forward edge. These parameters are not likely to change after a configuration geometry is specified.

The next entry is the standoff distance between the oxidizer tank and the aft fuel tank cones. This is a parameter that is not likely to change after the configuration geometry has been specified.

The last entry is the standoff distance between the aeroshell and the inner vehicle components. This is a parameter that is not likely to change after the configuration geometry has been specified.

| Body distribution: (fraction of horizontal tip fin planform area) | |
|---|---|
| Horizontal tip fin outboard angle= | 11.970 deg |
| Tip fins= | 1.534 |
| Elevon= | 0.480 |
| Rudder= | 0.431 |

The next block of user input data contains coefficients used to calculate the tip fin planform area and the surface areas of the elevons and rudders. The input data here is used to calculate the area of one tip fin and therefore one elevon and one rudder. The program assumes the vehicle has two tip fins and therefore two elevons and two rudders.

The program assumes the tip fins have a vertical component and a horizontal component. The horizontal component is defined by a triangle whose length is from the widest point on the body to the main engine nozzle exit plane. The first coefficient is the angle of the outer edge of the horizontal section of the tip fin to the longitudinal axis of the vehicle. The next entry is the ratio of the tip fin horizontal to total planform areas. The last two entries are the ratio of the elevon and rudder surface areas to the tip fin horizontal section planform surface area.

| Body Element: | Body Coefficient | | TPS Coefficient | |
|---|---|---|---|---|
| Nose | 2.587 | ft^2/ft^2 | 2.20 | lbm/ft^2 |
| | | | | |
| Forward Section | | | | |
| Glove | 0.710 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| Leeward Surface | 0.670 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | 0.631 | ft^2/ft^2 | 1.30 | lbm/ft^2 |
| | | | | |
| Forward Fuel Tank Cone Section | | | | |
| Glove | 0.902 | ft^2/ft^2 | 0.65 | lbm/ft^2 |
| Leeward Surface | 0.706 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | 0.655 | ft^2/ft^2 | 1.00 | lbm/ft^2 |
| | | | | |
| Aft Fuel Tank Cone Section | | | | |
| Glove | 0.883 | ft^2/ft^2 | 0.60 | lbm/ft^2 |
| Leeward Surface | 0.713 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | 0.655 | ft^2/ft^2 | 0.90 | lbm/ft^2 |
| | | | | |
| Thrust Structure Section | | | | |
| Glove | 0.879 | ft^2/ft^2 | 0.55 | lbm/ft^2 |
| Leeward Surface | 0.720 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | 0.687 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| | | | | |
| Body flaps= | | | 2.00 | lbm/ft^2 |
| Tip fin leading edge= | | | 2.00 | lbm/ft^2 |
| Tip fin windward= | | | 0.80 | lbm/ft^2 |
| Tip fin leeward= | | | 0.65 | lbm/ft^2 |
| Elevons= | | | 2.00 | lbm/ft^2 |
| Rudder= | | | 2.00 | lbm/ft^2 |

This next block of user input data contains the TPS unit masses (Section 2.3.1) and the body coefficients for the vehicle body sectors. These sectors are the vehicle nose, the forward section, the forward fuel tank cone section, the aft fuel tank cone section, and the thrust structure section. The body is further subdivided into the windward side, the leeward side and the body glove.

The TPS unit masses are also given for the tip fins and aerodynamic control surfaces. The tip fins are further subdivided into the leading edges, the windward side and the leeward side. An accurate choice the TPS unit masses will require knowledge the distribution of heat loads and the resulting temperatures on the vehicle body. This information can be generated by doing a thermal map of the vehicle body for the ascent and reentry trajectories at the moment of the peak heat loads.

The vehicle body sector wetted surface areas are calculated from the body sector planform area and the input body coefficients. These coefficients are the ratio of body sector wetted area to the body planform area. The body planform area and therefore the body wetted area is calculated in the weights spreadsheet from the vehicle propellant tank geometry and the vehicle body data in the input data block discussed above. The lifting body configuration model was setup this way to give the user flexibility in defining the vehicle configuration. The user will need to generate a new set of vehicle body coefficients for a new vehicle configuration.

| Tip fin surface area breakdown: | |
|---|---|
| Tip fin surface area/tip fin planform area= | 2.06 ft^2/ft^2 |
| Tip fin leading edge area/tip fin surface area= | 0.10 ft^2/ft^2 |
| Tip fin windward surface area/tip fin surface area= | 0.42 ft^2/ft^2 |
| Tip fin leeward surface area/tip fin surface area= | 0.48 ft^2/ft^2 |

This last block of user input data is a set of coefficients used in calculating the tip fin wetted surface area from the tip fin planform area. The first coefficient is used to calculate the total tip fin wetted area from the tip fin planform calculated in the weights spreadsheet. The next three coefficients are used to calculate the tip fin leading edge, windward side and leeward side wetted areas from the total tip fin wetted area. The tip fin planform area is calculated in the weights spreadsheet from user input data as discussed above.

# 2. SUBSYSTEM DESCRIPTION

This section of the user's guide describes the vehicle subsystem mass equations. Typical values of constants to be input into the program are also given.

## 2.1 Aerosurface Group

### 2.1.1 Wing

This section calculates the mass of the wings in the winged body configuration. The separation of the carry-through and exposed wing terms improves the accuracy of this equation when a large range of wing/body ratios are considered. The wing/body efficiency factor (f) is used to better reflect redistribution of total lift between wing and body as the relative size of wing and body change. The wing mass equation is of the form:

$$Mw = \{[Nz*FS*ml*(1/(1+f*Sb/Sw))]^{0.386}\}*\{[Sw/Tr]^{0.572}\}*\{[Wm*Lw^{0.572}] + [Wc*Lb^{0.572}]\},$$

where

Mw = wing mass (lbm)

Nz = ultimate normal load factor (g)

FS = factor of safety

ml = mass of vehicle at landing (lbm)

Sb = body planform area ($ft^2$)

Sw = exposed wing planform ($ft^2$)

Tr = exposed wing root chord max thickness (ft)

Lw = exposed total structural wing span (ft)

Lb = body width at wing body juncture (ft)

Wm = exposed wing material/configuration constant, where

- 0.286 = aluminum skin/stringer, dry wing
- 0.343 = aluminum skin/stringer, wet wing for storable propellant
- 0.229 = metallic composite (boron aluminum) honeycomb dry wing
- 0.263 = metallic composite (boron aluminum) wet wing for storable propellant
- 0.214 = organic composite honeycomb
- 0.453 = honeycomb dry wing, super alloy hot structure

Wc = wing carry through constant, where

- 0.0267 = dry carry-through (integral)
- 0.0347 = wet carry-through (integral)
- 0.1000 = dry carry-through (conventional)
- 0.1200 = wet carry-through (conventional)

f = wing/body efficiency factor, where

0.20 = for conventional vehicles

0.15 = for control configures vehicle

### 2.1.2 Tail Group

This section calculates the mass of the tip fins in both the winged body configuration and the lifting body configuration. The tail mass equation is of the form:

$$Mt = Vt*(St)^{1.24}$$

where

Mt = mass of one tail (lbm)

St = planform area of one tail ($ft^2$)

Vt = tail material/configuration coefficient ($lbm/((ft^2)^{1.24}$), where

1.872 = aluminum skin/stringer

1.108 = metallic composite structure

1.000 = graphite epoxy composite structure

1.500 = super alloy honeycomb hot structure

### 2.1.3 Body Flap

A body flap can survive as a vehicle control surface and as a method to shield the engines from re-entry heating for side entry vehicle configurations. The body flap mass equation is of the form:

$$Mbf = Bbf*(Sf)^{1.15}$$

where

Mbf = body flap mass (lbm)

Sf = body flap planform area ($ft^2$)

Bbf = body flap constant ($lbm/(ft^2)^{1.15}$), where

1.69 = hot structure

1.38 = aluminum structure

1.17 = composite structure

The above equation is also used to calculate the mass of the elevons and rudder for the lifting body configuration. In the winged body configuration, the rudder mass is included in the tip fin mass (section 2.1.2) and the elevon mass is included in the wing mass (Section 2.1.1).

### 2.1.4 Control Surface Actuation

The surface control actuators are used to move the aerodynamic control surfaces and to measure their positions. The choice of hydraulic actuators and electromechanical actuators (EMAs) must be matched between Sections 2.5.3, 2.8, 2.9 and 2.1.4. The control surface actuator system mass equation is of the form:

$$msc = Ssc*Sc + Spc$$

where

msc = control surface actuation system mass (lbm)

Sc = control surface area ($ft^2$)

Ssc = control surface actuator constant ($lbm/ft^2$)

Spc = miscellaneous system mass (lbm), where

| Ssc | Spc | |
|---|---|---|
| 3.75, | 200 | = 1973, rotary position transducers, hydraulic actuators |
| 2.61, | 200 | = 1987, 5000 psi system of advanced materials |
| 2.61, | 200 | = 1992, EMAs, Actuator Control Units (ACUs), Rotor Position sensor |
| 2.61, | 200 | = 1997, EMAs with light weight rare earth magnets |

## 2.2 Body Group

The body group includes the crew cabin, vehicle body wetted area, propellant tanks, body flap, payload bay, engine bay heat shield and range safety system.

### 2.2.1 Crew Cabin

This section calculates the mass of the crew cabin. The crew cabin mass equation is of the form:

$$Mcc = Bc*(Nc)^{0.5}$$

where

Mcc = main cabin mass (lbm)

Nc = number of crew

Bc = Cabin constant (lbm), where

- 2043 = full windshield aluminum construction
- 1293 = aluminum construction with out windshield
- 1740 = full windshield composite construction
- 1140 = composite construction without windshield

### 2.2.2 Vehicle Body

This section calculates the vehicle wetted body surface area mass. The mass of any vehicle wings and tail surface are calculated with the wing and tail equations. If integral propellant tanks are used, the mass of the tank wetted surface area is calculated by the propellant tank equations.

The following equation applies to the aeroshell of the lifting body configuration and the forward intertank, aft intertank and the aft skirt of the winged body configuration:

$$Mb = Bb*Swetb*(Nz*FS)^{1/3}$$

where

| | | |
|---|---|---|
| Mb | = | body surface area structure mass (lbm) |
| Swetb | = | wetted area of body structure less areas of main propellant tankage which double as body shell ($ft^2$) |
| Nz | = | normal load factor (g) |
| FS | = | factor of safety |
| Bb | = | body constant ($lbm/g^{1/3}$), where |

- 1.32 = composite honeycomb structure
- 2.72 = composite structure
- 3.20 = aluminum structure
- 3.40 = hot metallic Ti/Rene HC
- 4.43 = mold line tankage, tank, body structure, cryogenic insulation integrated

For the side entry conical configuration, unit masses for the unpressurized structures and the propellant tank barrel sections are calculated in the loads spreadsheet from the axial loads on the vehicle elements during the mode one and mode two burns and the bending moments imposed on the vehicle during reentry. The user inputs a minimum value for the vehicle side unit mass. The larger of the vehicle unpressurized structure unit mass and the minimum vehicle side unit mass is transferred to the weights spreadsheet for calculation of the unpressurized structures mass. The unpressurized structure elements are the vehicle nose, the forward intertank, the aft intertank and the aft skirt. Although the propellant tank barrel section's unit masses are calculated during the calculation of the unpressurized structure unit masses, this information is not currently used in the calculation of the propellant tank masses (Section 2.2.3).

The equation for the vehicle element linear load is:

$$LL = FS*L/(2*12{,}000*\pi*r),$$

where

| | | |
|---|---|---|
| LL | = | vehicle element linear load (klbf/in) |
| FS | = | vehicle factor of safety |
| L | = | vehicle axial load (lbf) |
| $\pi$ | = | $\pi$ |
| r | = | vehicle element radius (ft) |

The equation for the vehicle element unit mass (for a ten foot diameter cylinder made of 2024-T4 aluminum) is:

$$x = k1*LL^{k2},$$

where

| | | |
|---|---|---|
| x | = | vehicle element unit mass assuming the vehicle element is a ten ft diameter cylinder made of 2024-T4 aluminum (lbm/ft$^2$) |
| LL | = | vehicle element linear load (klbf/in) |
| k1 | = | first vehicle element structural constant |
| k2 | = | second vehicle element structural constant, where |

| k1 | k2 | |
|---|---|---|
| 4.3 | 0.47 | = monocoque structure |
| 2.3 | 0.44 | = integral 45 degree waffle or optimized stiffeners and ring |
| 0.52 | 0.83 | = truss core dual sandwich |

The equation for the vehicle element unit mass corrected for the actual vehicle element diameter is:

xx = x*sqrt(12*d/120),

where

| | | |
|---|---|---|
| xx | = | vehicle unit mass corrected for actual vehicle element base diameter (lbm/ft$^2$) |
| x | = | vehicle element unit mass assuming the vehicle element is a ten feet diameter cylinder made of 2024-T4 aluminum (lbm/ft$^2$) |
| d | = | vehicle element base diameter (ft) |

The equation for the vehicle element unit mass corrected for the vehicle sidewall half angle is:

xxx = xx*k,

where

| | | |
|---|---|---|
| xxx | = | vehicle unit mass corrected for vehicle sidewall half angle (lbm/ft$^2$) |
| xx | = | vehicle unit mass corrected for actual vehicle element base diameter (lbm/ft$^2$) |
| θ | = | vehicle sidewall half angle (degrees) |
| k | = | corrective factor for vehicle sidewall half angle, where |

| θ | k |
|---|---|
| 0 | 1.00 |
| 10 | 1.10 |
| 20 | 1.28 |
| 30 | 1.50 |
| 40 | 1.77 |
| 50 | 2.15 |

The equation for the vehicle element unit mass corrected for the vehicle element materials is:

xxxx = xxx*(ρ/0.10)*(10,700,000/E)

where

| | | |
|---|---|---|
| xxxx | = | vehicle unit mass corrected vehicle materials (lbm/ft$^2$) |
| xxx | = | vehicle unit mass corrected for vehicle sidewall half angle (lbm/ft$^2$) |
| $\rho$ | = | vehicle element material density (lbm/in$^2$) |
| E | = | vehicle element material young's modulus (psi) |

The nose of the winged body configuration and the side entry conical configuration is a biconic with a hemispherical tip. Since the nose is a lightly loaded structure, its mass is calculated from an input surface area unit mass. For the winged body configuration, the user inputs a value for the nose unit mass. For the side entry conical configuration, the user input value for the minimum vehicle side unit mass is used in the calculation of the nose mass. Since the winged body configuration is a cylinder, r1 is calculated from the vehicle body diameter. For the side entry conical configuration, r1 is calculated from the vehicle base diameter, vehicle length and vehicle half angle.

The equations describing the nose surface area are:

| | | |
|---|---|---|
| r2 | = | r1*(r2/r1) |
| r3 | = | r1*(r3/r1) |
| l1 | = | (r1- r2)/tan(exterior angle 1) |
| l2 | = | (r2- r3)/tan(exterior angle 2) |
| a1 | = | $\pi$*(r1+ r2)*sqrt(((r1-r2)$^2$)+ (l1$^2$)) |
| a2 | = | $\pi$*(r2+ r3)*sqrt(((r2-r3)$^2$)+ (l2$^2$)) |
| a3 | = | (2/3)*$\pi$*(r3)$^3$ |
| a | = | a1+ a2+ a3, |

where

| | | |
|---|---|---|
| $\pi$ | = | $\pi$ |
| r1 | = | nose cone 1 base radius (ft) |
| r2 | = | nose cone 2 base radius (ft) |
| r3 | = | nose tip radius (ft) |
| l1 | = | nose cone 1 height (ft) |
| l2 | = | nose cone 2 height (ft) |
| a1 | = | surface area of nose cone 1 (ft$^2$) |
| a2 | = | surface area of nose cone 2 (ft$^2$) |
| a3 | = | surface area of the nose tip (ft$^2$) |
| a | = | total nose surface area (ft$^2$) |
| r2/r1 | = | ratio of nose cone 1 and 2 base radius, where |
| | | 0.80 = typical biconic nose value |

r3/r1 = ratio of nose cone 1 base radius and nose tip radius, where
0.20 = typical biconic nose value
exterior angle 1 = angle of the cone 1 side from vertical (deg), where
17.0 = typical biconic nose value
exterior angle 2 = angle of the cone 2 side from vertical (deg), where
30.0 = typical biconic nose value

The equation for the nose mass is:

$$Mn = kn*a,$$

where

Mn = nose mass (lbm)
a = nose surface area ($ft^2$)
kn = nose unit mass ($lbm/ft^2$), where
0.15 = minimum gage for Gr-Ep structure
0.20 = minimum gage for aluminum lithium structure

For the winged body configuration, a user specified value for the nose unit mass (kn) is used. For the side entry conical configuration, the minimum value the user input for the unpressurized structure unit mass is also used as the nose unit mass (kn).

### 2.2.3 Propellant Tanks

The performance spreadsheet calculates the fuel and oxidizer mass required for the vehicle to perform its mission. The weights spreadsheet takes the propellant masses and calculates the propellant volume required. The ullage factor includes the ullage space required at the start of the burn, an allowance for the residual propellants and the engine ignition propellant (see Section 2.14.6).

The equations for the propellant tank volumes are:

$$Vp = UF*Wp/\rho,$$

where

Vp = propellant volume ($ft^3$)
UF = ullage factor
Wp = propellant mass (lbm)
$\rho$ = propellant density ($lbm/ft^3$)

The side entry conical configuration and the winged body configuration sizing tools use the tank position flag, the tank design flag, the upper and lower endcap flags, the number of fuel tanks flag, and the payload bay location flag in defining the vehicle layout. The lifting body configuration uses only the number of fuel tanks flag in defining the vehicle layout. The sizing program does not check for consistency between tank position, tank design and tank endcap. If the number of fuel tanks flag is set to 1, do not use tank position 2 or fuel tank 1.

Position specifies the relative position of a fuel or oxidizer tank where

1 = tank is in the aft position
2 = tank is in the middle position
3 = tank is in the forward position

Tank design specifies the tank endcap type, where

1 = common bulkhead, tank has only aft endcap, intertank height = 0.0 foot
2 = nested bulkhead, intertank height 1.0 foot
3 = separate tanks, intertank height 1.2*height of both endcaps

Upper endcap flag specifies type of endcap, where

1 = ellipsoidal endcap, ratio of ellipse height and width is an input
2 = hemispherical endcap
3 = toroidal endcap, radius = .7*tank radius, height = .3*tank radius

Lower endcap flag specifies the type of endcap, where

1 = ellipsoidal endcap, ratio of ellipse height and width is an input
2 = hemispherical endcap
3 = toroidal endcap, radius = .7*tank radius, height .3*tank radius

Number of fuel tanks is a bi/tripropellant flag, where

1 = bipropellant configuration with one oxidizer and one fuel
2 = tripropellant configuration with one oxidizer and two fuels

Payload location flag specifies which intertank the payload bay is in, where

1 = payload bay is in aft intertank
2 = payload bay is in forward intertank

The propellant tanks are sized by pressure loads. The propellant tanks are split into a forward endcap, a barrel section and an aft endcap. The forward endcap pressure is the tank ullage pressure. The aft endcap pressure is the sum of the tank ullage pressure and the barrel section dynamic head. The barrel section pressure is the average of the forward and aft endcap pressures.

The equations calculating the propellant tank pressures are:

$$Pu = Pullage$$
$$Pl = Pu + No*\rho*l/144$$
$$Pb = (Pu + Pl)/2,$$

where

Pu = tank upper endcap pressure (psi)
Pl = tank lower endcap pressure (psi)
Pb = tank barrel section pressure (psi)
Pullage = tank ullage pressure (psi)
No = vehicle liftoff thrust to weight
ρ = propellant fuel (lbm/ft$^3$)
l = tank barrel section length (ft)

An ellipsoidal endcap:

a = 0.50*π*{r$^2$+ [h$^2$/(2*e)]*ln[(1+ e)/(1- e)]}
t = 12*FS*P*r*{[(r/h)+ 1]/4}/Ftu
h = r*e

where
a = ellipsoidal endcap surface area (ft$^2$)
t = ellipsoidal endcap thickness (in)
π = π
r = endcap radius (ft$^2$)
h = endcap height (ft)
FS = factor of safety
P = pressure in endcap (psi)
Ftu = ultimate tensile strength of the endcap material (psi)
e = ellipsoidal endcap eccentricity, where
0.7071 = typical value (square root of 2) for an ellipsoidal endcap

A hemispherical endcap:

a = 2*π*r$^2$
t = 12*FS*P*r*/(2*Ftu),

where

a = ellipsoidal endcap surface area (ft$^2$)
t = ellipsoidal endcap thickness (in)
π = π
r = endcap radius (ft$^2$)
FS = factor of safety
P = pressure in endcap (psi)
Ftu = ultimate tensile strength of the endcap material (psi)

A toroidal endcap:

| | | |
|---|---|---|
| a | = | $2*\pi^2*r*h$ |
| t | = | $12*(FS*P*h/Ftu)*(2*r- h)/(2*r- 2*h)$, |

where

| | | |
|---|---|---|
| a | = | ellipsoidal endcap surface area ($ft^2$) |
| t | = | ellipsoidal endcap thickness (in) |
| $\pi$ | = | $\pi$ |
| r | = | endcap inner edge radius ($ft^2$) |
| h | = | endcap height ($ft^2$) |
| FS | = | factor of safety |
| P | = | pressure in endcap (psi) |
| Ftu | = | ultimate tensile strength of the endcap material (psi) |

A conical barrel section:

| | | |
|---|---|---|
| a | = | $\pi(r1+ r2)*sqrt([(r2- r1)^2+ l^2]$ |
| t | = | $12*FS*P*r2/[Ftu*cos(\theta)]$, |

where

| | | |
|---|---|---|
| a | = | barrel section surface area ($ft^2$) |
| t | = | barrel section thickness (in) |
| r1 | = | upper tank radius (ft) |
| r2 | = | lower tank radius (ft) |
| l | = | barrel section length (ft) |
| FS | = | factor of safety |
| P | = | average pressure in the barrel section (psi) |
| Ftu | = | ultimate tensile strength of the barrel section material (psi) |
| $\theta$ | = | barrel section tank wall half angle (deg) |

The strength and stiffness of carbon-epoxy, kevlar-epoxy and e glass-epoxy in the following table used an average of the axial and transverse values. Since the fiber orientation can be tailored to the expected loads, these are conservative values. It is not known if the graphite polyimide strength and stiffness values are equally conservative. Factors of safety used on composite parts are typically larger than those used on metallic parts.

Table of possible propellant tank materials:

| Material | Ultimate Strength (psi) | Density ($lbm/in^3$) | Young's Modulus (psi) |
|---|---|---|---|
| 2219 Aluminum | 60,000 | 0.102 | 10,800,000 |
| 2195 Al-Li | 82,000 | 0.098 | 11,000,000 |
| Carbon-Epoxy | 113,000 | 0.057 | 20,000,000 |
| Kevlar-Epoxy | 112,000 | 0.050 | 12,500,000 |
| "E" Glass-Epoxy | 93,000 | 0.072 | 6,000,000 |
| Graphite Polyimide | 151,000 | 0.056 | 20,600,000 |

In selecting the allowable ultimate strength of a material, allowance must be made to the material properties to account for the fatigue that the vehicle would see. The value used in the Access to Space, Option 3 final report was a knockdown of 20% on the stress and 10% on the stiffness.

The equation describing the mass of a propellant tank mass as a pressure vessel is:

$$mpv = 144*a*t*\rho,$$

where

mpv = mass of the tank element as a pressure vessel (lbm)

a = surface area of the tank element ($ft^2$)

t = thickness of the tank element (in)

$\rho$ = propellant tank material density ($lbm/in^3$)

A tank efficiency factor is added to the propellant tank mass to account for the difference between the tank as an ideal pressure vessel and the historical mass of actual propellant tanks. This equation was based on the use of aluminum. A density correction is needed if a different material is used.

The equation for the additional mass for non ideal factors in the vehicle propellant tank masses is:

$$dm = (\rho/0.100)*0.10*(a^{1.339})$$

where

dm = additional propellant tank mass to account for non ideal factors (lbm)

$\rho$ = propellant tank material density ($lbm/in^3$)

a = surface area of the propellant tank ($ft^2$)

The equation for the total propellant tank mass is:

$$m = mpv + dm$$

where

m = total propellant tank mass (lbm)

mpv = mass of all of the elements of the propellant tank as a pressure vessel (lbm)

dm = the non-ideal propellant tank mass (lbm)

### 2.2.4 Payload Bay

The payload bay structure is used to mount the payload in the vehicle. The payload bay mass equation is of the form:

Mplb = kplb

where

Mplb = mass of the payload bay structure (lbm)

kplb = payload bay mass constant (lbm), where

3700 = Option 3 Access to Space SSTO(R) payload bay doors, support structure and canister (payload bay is 15 ft in diameter and 30 ft long)

5919 = Space Shuttle Orbiter payload bay door, liner and provisions (payload bay is 15 ft in diameter and 60 ft long)

### 2.2.5 Engine Bay Heat Shield

The engine bay heat shield stretches across the aft end of the vehicle and protects the engine bay from the main engine plume thermal radiation and hot gas recirculation. The engine bay heat shield mass equation is of the form:

Mehs = beg*Seb

where

Mehs = engine bay heat shield mass (lbm)

Seb = engine bay surface area ($ft^2$)

beg = engine bay heat shield constant ($lbm/ft^2$), where

1.64 = graphite-PEEK honeycomb structure, to which TABI blanket TPS is bonded

### 2.2.6 Range Safety System

A range safety system is a demolition system that can be used to destroy a launch vehicle that strays beyond the launch site range safety limits. This system is installed on the current generation of expendable launch vehicles. A program decision that must be made early in a reusable launch vehicle project is will the vehicle have a range safety system. A typical range safety system incorporates a demolition charge to destroy the vehicle, a radio receiver to receive the destruct command and a power source to run the system. The range safety system mass equation is of the form:

Mrs = krs

where

Mrs = range safety system mass (lbm)

krs = range safety system constant (lbm), where

323 = Early Heavy Lift Launch Vehicle (EHLLV) core range safety system mass

235 = National Launch System (NLS) range safety system mass

0 = no range safety system on the vehicle

## 2.3 Environmental Protection

The environmental protection system includes the vehicle thermal protection system, the body insulation system and the cryogenic propellant tank system.

### 2.3.1 Thermal Protection System (TPS)

The thermal protection system is on the vehicle skin. It protects the vehicle from the heat loads imposed on the vehicle during ascent and reentry by reducing the amount of energy conducted through the TPS to the vehicle interior. This allows the vehicle structure operating temperature to be less than the reentry temperatures.

The blankets are the preferred TPS materials where temperatures and airloads permit, since they can be bonded directly to contoured surfaces using a silicon rubber adhesive (RTV). The installation of the rigid tiles is more involved. Since there is a significant difference in the thermal expansion coefficients between the tile materials and aluminum, it is necessary to use strain insulation pads (SIP) as an interface to prevent damage to the tiles as the tiles heat up. Since the tile material and Gr-Ep have similar thermal expansion coefficients, it will be possible to bond tiles to Gr-Ep structure. The TPS tiles are silica tiles (LI), fibrous refractory composite insulation (FRCI) and alumina enhanced thermal barrier (AETB). The TPS blankets are tailorable advanced blanket insulation (TABI), advanced flexible reusable surface insulation (AFRSI) and composite flexible blanket insulation (CFBI).

AFRSI currently has an operational temperature limit of 1200°F. TABI currently has an operational temperature limit of 1800°F.

The metallic tiles are a possible alternative to the ceramic tiles. They would be larger than the ceramic tiles, be less of a problem to attach to the vehicle, and give the vehicle a better all-weather capability.

The TPS mass equation is of the form:

Mtps = ktps*Stps,

where

Mtps = TPS mass (lbm)

Stps = TPS surface area ($ft^2$)

The terms in the TPS materials and concepts tables are:

Qdot = heat flux on TPS (btu/$ft^2$-sec)

T = temperature that the TPS is exposed to (°F)

t = TPS thickness (in)

ktps = TPS unit mass (lbm/$ft^2$)

TPS material is AFRSI

Vehicle structure is aluminum (peak temperature limit = 350°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 0.35 | 431 | 0.25 | 0.28 * |
| 1.07 | 831 | 0.73 | 0.65 * |
| 4.99 | 1637 | 1.53 | 1.27 |

TPS material is CFBI

Vehicle structure is aluminum (peak temperature limit = 350°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 0.35 | 438 | 0.25 | 0.30 |
| 1.07 | 830 | 0.68 | 0.65 * |
| 4.99 | 1636 | 1.48 | 1.32 |

TPS material is TABI

Vehicle structure is aluminum (peak temperature limit = 350°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 0.35 | 431 | 0.25 | 0.30 |
| 1.07 | 809 | 0.69 | 0.66 |
| 4.99 | 1497 | 1.32 | 1.19 |
| 6.45 | 1647 | 1.50 | 1.34 |
| 8.16 | 1794 | 1.65 | 1.46 |
| 11.22 | 2013 | 1.86 | 1.64 |
| 18.80 | 2423 | 2.29 | 2.00 |

TPS material is LI900

Vehicle structure is aluminum (peak temperature limit = 350°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 0.35 | 427 | 0.25 | 0.51 |
| 1.07 | 775 | 0.58 | 0.76 |
| 4.99 | 1391 | 1.13 | 1.17 * |
| 6.45 | 1518 | 1.30 | 1.30 * |
| 8.16 | 1640 | 1.44 | 1.40 * |
| 11.22 | 1817 | 1.63 | 1.54 * |
| 18.70 | 2131 | 2.03 | 1.84 * |

TPS material is FRCI12

Vehicle structure is aluminum (peak temperature limit = 350°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 4.99 | 1386 | 1.22 | 1.54 |
| 6.45 | 1513 | 1.37 | 1.69 |
| 8.16 | 1636 | 1.49 | 1.81 |
| 11.22 | 1813 | 1.66 | 1.98 |
| 18.70 | 2128 | 2.02 | 2.34 |
| 39.37 | 2664 | 2.61 | 2.93 |

TPS material is AETB8

Vehicle structure is aluminum (peak temperature limit = 350°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 1.07 | 775 | 0.76 | 0.83 |
| 4.99 | 1390 | 1.59 | 1.38 |
| 6.45 | 1517 | 1.78 | 1.51 |
| 8.16 | 1639 | 1.97 | 1.63 |
| 11.22 | 1816 | 2.23 | 1.81 |
| 18.70 | 2131 | 2.73 | 2.14 |
| 39.37 | 2668 | 3.52 | 2.67 * |

*minimum TPS unit mass for this vehicle structural material

TPS material is TABI

Vehicle structure is Gr-Ep (peak temperature limit = 550°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 6.45 | 1647 | 0.85 | 0.80 |
| 8.16 | 1794 | 0.99 | 0.91 |
| 11.22 | 2014 | 1.21 | 1.10 |
| 18.70 | 2423 | 1.72 | 1.52 |

TPS material is LI900

Vehicle structure is Gr-Ep (peak temperature limit = 550°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 6.45 | 1517 | 0.72 | 0.79 |
| 8.16 | 1640 | 0.83 | 0.87 * |
| 11.22 | 1817 | 1.01 | 1.01 * |
| 18.70 | 2132 | 1.43 | 1.32 * |

TPS material is FACI12

Vehicle structure is Gr-Ep (peak temperature limit = 550°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 6.45 | 1513 | 0.80 | 1.05 |
| 8.16 | 1636 | 0.91 | 1.16 |
| 11.22 | 1813 | 1.09 | 1.34 |
| 18.70 | 2128 | 1.50 | 1.75 |
| 39.37 | 2664 | 2.19 | 2.44 |

TPS material is AETB8

Vehicle structure is Gr-Ep (peak temperature limit = 550°F)

| Qdot | T | t | ktps |
|---|---|---|---|
| 6.45 | 1516 | 1.04 | 0.94 |
| 8.16 | 1639 | 1.19 | 1.04 |
| 11.22 | 1816 | 1.45 | 1.22 |
| 18.70 | 2131 | 2.02 | 1.60 |
| 39.37 | 2668 | 2.91 | 2.19 * |

*Minimum TPS unit mass for this vehicle structural material

Metallic TPS Panel Concepts

| Temperature | Type | Material/Structure | ktps |
|---|---|---|---|
| <1000 F | prepackaged | titanium/multiwall | 0.75 |
| 1000 to 1600 F | prepackaged | super alloy/honeycomb | 1.41 |
| 1600 to 2000 F | prepackaged | super alloy/honeycomb | 1.50 |
| 1600 to 2000 F | standoff | carbon - carbon/rib stiff | 1.84 |
| >2000 F | standoff | carbon - carbon/rib stiff | 2.31 |

### 2.3.2 Body Insulation

Vehicle body insulation stands between the vehicle outer skin and the interior of the vehicle. It is used only if the vehicle TPS does not incorporate enough insulation to bring the interior body temperature down to a low enough level. The choice of a hot vehicle structure will probably require the use of body insulation. The body insulation mass equation is of the form:

Mbi = kbi*Swetv

where

Mbi = body insulation mass (lbm)

Swetv = wetted body area ($ft^2$)

kbi = body insulation constant (lbm/ft$^2$), where

0.31 = 1973, fibrous bulk blankets & MLI

0.28 = 1992, light weight fibrous bulk blankets & MLI

1.00 = hot 1997, micro quartz in nickel alloy packages

0.28 = warm 1997, micro quartz in nickel alloy packages

### 2.3.3 Cryogenic Propellant Tank Insulation

The cryogenic propellant tank insulation reduces the amount of thermal energy that flows into a cryogenic propellant tank. This reduces the propellant boiloff and prevents the condensation of water and air on the sides of the propellant tanks. One of the ways to prevent the condensation on the surface of a cryogenic propellant tank is to have purge gas flowing over the surface of the propellant tank. This choice implies a space between the tank wall and the vehicle skin. The cryogenic propellant tank insulation mass equation is of the form:

$$Mci = kci*Aci,$$

where

Mci = cryogenic tank insulation mass (lbm)

Aci = cryogenic tank surface area (ft$^2$)

kci = cryogenic tank insulation constant (lbm/ft$^2$), where

0.334 = 1973, spray on foam insulation

0.2375 = 1987, closed cell PVC with nitrogen purge

0.48 = 1992, Q fiber over PMC foam with nitrogen purge

0.202 = 1997, closed cell foam with kapton-aluminum-kapton

0.250 = external Rhoacell foam insulation

## 2.4 Landing Gear and Auxiliary Systems

Landing gear weight is applied against the vehicle landing mass. The payload mass should be included in the vehicle landing mass. The landing gear and auxiliary systems mass equation is of the form:

$$Mg = kl*ml,$$

where

Mg = landing gear mass (lbm)

ml = landed mass (lbm)

kl = a constant percentage of landed mass for landing gear (lbm/lbm), where

0.0330 = shuttle gear (horizontal landing)

0.0300 = advanced composite gear (horizontal landing)

0.0265 = composite skid system or composite wheel system with no brakes (horizontal landing)

0.0310 = SERV vehicle design (1969) (vertical landing)
0.0335 = Gomersall SSTO vehicle design (1970) (vertical landing)
0.0396 = BETA vehicle design (1971) (vertical landing)
0.0279 = Phoenix vehicle design (1985) (vertical landing)
0.0108 = BETA II vehicle design (1986) (vertical landing)
0.0213 = X Rocket vehicle design (1987) (vertical landing)

## 2.5 Main Propulsion

The main propulsion system includes the main engines, the feed line manifold and pressurization system, the gimbal actuators and the main engine thrust structure.

### 2.5.1 Main Engines

Main engine weight is input as a fixed engine weight and number of engines or as an engine thrust-to-weight ratio and a vehicle start of burn thrust-to-weight ratio.

Burn flag is a flag describing the sequence of firing the Mode 1 and Mode 2 engines, where

1 = The engines operate in a serial burn fashion where the Mode 1 and Mode 2 engines are separate engines and the mode two engines are not started until the mode one engines are shutoff

2 = The engines operate in a parallel burn fashion where the Mode 1 and Mode 2 engines are separate engines and both the Mode 1 engines and the Mode 2 engines are started at liftoff

3 = There is only one engine that can switch from Mode 1 operation to Mode 2 operation

Engine flag defines how the engine masses will be calculated, where

1 = The number of engines and the engine mass are input

2 = The engine thrust-to-weight ratio and vehicle initial thrust-to-weight ratio are input

Engine mass equations are:

| | |
|---|---|
| If(engine flag=1) | Me1= Ne1*We1<br>Me2= Ne2*We2 |
| If(engine flag=2) | Me1= Fsl1*ke1<br>Me2= Fv2*ke2 |
| If(burn flag= 2) | Me= Me1+Me2 |
| If(burn flag= 2) | Me= Me1+Me2 |
| If(burn flag= 3) | Me= Me1, |

where

Me1 = mass of the Mode 1 engines (lbm)
Me2 = mass of the Mode 2 engines (lbm)

Me = mass of the vehicles main engines (lbm)

Ne1 = number of the Mode 1 engines

Ne2 = number of the Mode 2 engines

Fsl1 = sea level thrust of the Mode 1 engines (lbf) (calculated by the program)

Fv2 = vacuum thrust of the Mode 2 engines (lbf) (calculated by the program)

We1 = mass of a single Mode 1 engine (lbm)

We2 = mass of a single Mode 2 engine (lbm)

ke1 = sea level thrust-to-weight ratio for the Mode 1 engines (Fsl1/We1) (lbf/lbm)

ke2 = vacuum thrust-to-weight ratio for the Mode 2 engines (Fsl1/We1) (lbf/lbm)

### 2.5.2 Feed Line Manifold and Pressurization System

This system includes the main propellant feed system, the main propellant tank pressurization system and the engine purge system masses. The feed line and pressurization system mass equation is of the form:

$$Mfp = rpf*mdot/\rho,$$

where

Mfp = propellant feed system mass (lbm)

mdot = propellant mass flow rate (lbm/sec)

$\rho$ = propellant density (lbm/ft$^3$)

rpf = line manifold and pressurization system (lbm-sec/ft$^3$), where

64.0 = metallic feed line, vacuum jacket insulation

55.0 = composite/metallic feed line, foam insulation

### 2.5.3 Gimbal Actuators

This system is the main engine gimbal system actuator mass. Power to run the systems is calculated in the prime power (Section 2.8). Transmission of this power is handled in the power conversion and distribution (Section 2.9). If differential throttling is used for vehicle thrust vector control (TVC), these coefficients will be set to zero. The choice of hydraulic actuators and EMAs must be matched between Sections 2.5.3, 2.8, 2.9 and 2.1.4. The gimbal actuator mass equation is of the form:

$$Mga = rga*Tvac,$$

where

Mga = gimbal actuator mass (lbm)

Tvac = vehicle vacuum thrust (lbf)

rga = gimbal actuators coefficient (lbm/lbf), where

0.00129 = hydraulic actuators

0.00075 = EMAs

### 2.5.4 Thrust Structure

Thrust structure mass is a function of both the maximum engine thrust and the number of engines. The thrust structure mass equation is of the form of:

Mts = (ktsf+ktsn*(Ne- 1))*Fv

where

Mts = thrust structure mass (lbm)

Ne = number of engines

Fv = maximum vehicle thrust (lbf)

ktsf = thrust structure constant for the maximum thrust (lbm/lbf), where

ktsn = thrust structure constant for the number of engines (lbm/lbf)

| ktsf | ktsn | | |
|---|---|---|---|
| 0.00300, | 0.00057 | - | Ti struts with boron/epoxy end fittings |
| 0.00240, | 0.00043 | - | diffusion bonded titanium |
| 0.00207, | 0.00039 | - | graphite epoxy truss |

If burn flag is set to 3, Fv is the Mode 1 engine vacuum thrust. If burn flag is not set to 3, Fv is the sum of the Mode 1 and Mode 2 engine vacuum thrust (see Section 2.5.1 for burn flag definition).

## 2.6 Reaction Control System (RCS)

The RCS is used to control the vehicle attitude when the vehicle's areo surfaces or the vehicle's TVC cannot provide vehicle attitude control. RCS propellant is part of the auxiliary propellant (see Section 2.7). The RCS mass equation is of the form:

Mrcs = rrcs*me*Lr,

where

Mrcs = RCS system mass (lbm)

me = entry mass (lbm)

Lr = vehicle reference length (ft)

rrcs = RCS constant (lbm/lbm-ft) (includes tanks, pressurization and feed, gimbal actuators inc.), where

1.36e-4 = 1973, $N_2O_4$/MMH

1.51e-4 = 1987, $O_2/H_2$

## 2.7 Orbital Maneuvering System (OMS)

The OMS is used for on-orbit maneuvers including orbit transfers, rendezvous, and deorbit. The OMS propellant tankage is sized to hold all of the auxiliary propellant (see Sections 2.6, 2.14.3, 2.14.4, 2.14.5 and 2.14.6). The OMS mass equation is of the form:

$$Moms = No*kme*Mo + kt*Wfax$$
$$Wfax = Wfoms + Wfrcsa + Wfrese + Wfrcso + Wfl + Wfrs,$$

where

| | | |
|---|---|---|
| Moms | = | OMS mass (lbm) |
| No | = | on-orbit initial thrust to weight ratio (g) |
| Mo | = | on-orbit initial mass (lbm) |
| Wfax | = | total auxiliary propellant load (lbm) |
| Wfoms | = | OMS propellant load (lbm) |
| Wfrcsa | = | ascent RCS propellant load (lbm) |
| Wfrcse | = | entry RCS propellant load (lbm) |
| Wfrcso | = | on-orbit RCS propellant load (lbm) |
| Wfl | = | landing propellant load (lbm) |
| Wfrs | = | engine restart propellant load (lbm) |
| kme | = | OMS engine constant (lbm/lbm-g), |
| kt | = | OMS propellant tank constant (lbm/lbm) (includes pressurization and feed), where |

| kme | kt | | |
|---|---|---|---|
| 0.0863, | 0.119 | = | 1973, $N_2O_4$/MMH |
| 0.035, | 0.152 | = | 1987, $O_2/H_2$ |

## 2.8 Prime Power

The prime power system is the energy source that is used to supply power to the vehicle during the mission. The vehicle's aero surface controls and engine thrust vector controls may be moved by hydraulic actuators or by EMAs. Since fuel cells may operate at peak power level for short periods of time, they may be used to supply power during ascent and landing. The power conversion and distribution system is used to move power from the prime power source (Section 2.8) to the engine gimbal actuators (Section 2.5.3) and the control surface actuators (Section 2.1.4). The choice of hydraulic actuators and EMAs must be matched between Sections 2.5.3, 2.8, 2.9, and 2.1.4. Be careful that you do not double count the ascent power (PWc, PWe and PWa) and the on-orbit power (FCc). The prime power system mass equation is of the form:

$$Mpow = PWc*Sc + PWe*Tvac + PWa*ma + D*PWav*FCc + PWav*FCw,$$

where

| | | |
|---|---|---|
| Mpow | = | prime power source mass (lbm) |
| D | = | number of days the vehicle is on orbit |
| PWav | = | average vehicle on orbit electrical power demand (kw) |
| Sc | = | total surface control area ($ft^2$) |
| Tvac | = | total vacuum thrust of gimbaled engines (lbf) |

ma = avionics mass (lbm)

FCc = on orbit energy supply unit mass (lbm/kw-day), where

29.26 = Orbiter fuel cell reactants and tankage

FCw = on orbit power supply unit mass (lbm/kw), where

28.71 = Orbiter fuel cell unit mass

3.78 = 270 volt high power density fuel cells (from Access to Space, Option 3 Final Report)

PWc = surface control power demand (lbm/ft$^2$)

PWe = engine gimbal power demand (lbm/lbf)

PWa = avionics power demand constant (lbm/lbm), where

| PWc | PWe | PWa | |
|---|---|---|---|
| 0.712, | 0.97e-4, | 0.405 | = 1973, hydrazine APU, hydraulic actuators, H2-O fuel cells |
| 0.610, | 0.97e-4, | 0.405 | = 1973, hydrazine APU, hydraulic actuators, accumulators for peak power, H2-O2 fuel cells |
| 0.4854, | 0.661e-4, | 0.276 | = 1987, EMAs, advanced fuel cells, H2-O2 APUs |
| 0.274, | 0.373e-4, | 0.155 | = 1992, EMAs, full power range advanced fuel cells, advanced batteries |
| 0.0, | 0.0, | 0.0 | = High power density fuel cells sized for on orbit operations, operating at peak power demand levels to EMAs during ascent and landing |

## 2.9 Power Conversion and Distribution

The power conversion and distribution system is used to move power from the prime power source (Section 2.8) to the control surface actuators (Section 2.1.4). The choice of hydraulic actuators and EMAs must be matched between Sections 2.5.3, 2.8, 2.9 and 2.1.4. The power conversion and distribution system mass equation is of the form:

Mh = Ncs*Sc+ Ne*Tvac+E*ml,

where

Mh = power conversion and distribution system mass (lbm)

ml = landed vehicle mass (lbm)

Sc = total surface control area (ft$^2$)

Tvac = total vacuum thrust of gimbaled engines (lbf)

E = electrical power conversion and distribution system constant (lbm/lbm),

Ncs = surface control constant (lbm/ft$^2$),

Ne = engine related gimbal actuation (lbm/lbf)

| E | Ncs | Ne | |
|---|---|---|---|
| 0.038 | 2.10 | 3.00e-4 | = 3000 psi hydraulic actuator system |
| 0.038 | 1.23 | 1.68e-4 | = 5000 psi hydraulic actuator system |
| 0.020, | 0.0, | 0.0 | = EMAs |

## 2.10 Avionics

The avionics subsystems include navigation, communications and tracking, displays and controls, instrumentation and data processing. The avionics system mass equation is of the form:

$$Mav = kav*(Md)^{1/8}$$

where

Mav = avionics system mass (lbm)

Md = vehicle dry mass (lbm)

kav = avionics system constant (lbm/lbm$^{1/8}$), where

- 1350 = 1973, high speed serial data busses, voting architecture, GPCs, magnetic memory, S-band
- 810 = 1987, ring laser gyros, high speed processors, data compression, GPS navigation update
- 729 = 1992, fiber optic gyros and data buss, health monitoring, very high speed integrated circuits, distributed processing
- 710 = 1997, adaptive guidance, navigation, and control, fault tolerance, health monitoring, optical memories, smart sensors

## 2.11 Environmental Control

The environmental control system provides cooling for the avionics and a way to remove the heat from the vehicle. If the vehicle has a crew, this system includes a crew cabin and a method to remove heat produced in the crew cabin. Because of losses in power generation and the heat produced by the crew, the average on-orbit heat rejection requirement (PWr) is larger than the average on-orbit electrical power demand (PWav, Section 2.8). The environmental control system mass equation is of the form:

$$Menv = Ec*(Vp)^{0.75} + Eo*Nc*D+ Ea* mav+PWr*Ew$$

where

Menv = environmental control system mass (lbm)

Nc = number of crew (crew member)

D = number of days on orbit (day)

mav = avionics mass (lbm)

PWr = average on orbit heat rejection requirement (kw)

Vp = total pressurized volume (including wheel wells) (ft$^3$)

Ec = crew cabin volume constant ((lbm/(ft$^{1/3}$)$^{0.75}$), where

5.85 = typical manned vehicle value

Eo = crew supplies constant (lbm/crew member-day), where

10.9 = typical manned vehicle value

Ea = avionics waste heat removal constant (lbm/lbm), where

0.44 = 1973, active cooling by thermal management system

0.22 = 1992, passive conduction into vehicle structure

Ew = cabin heat rejection constant (lbm/kw), where

200 = Space Shuttle orbiter water coolant loop, freon coolant loop and radiators

### 2.12 Dry Mass Contingency

A contingency factor is added to the vehicle's dry mass (Sections 2.1.1 through 2.11) to account for the growth in vehicle mass that has historically occurred as the vehicle program matures. The vehicle dry mass equation is of the form:

$$Mc = cf*dw$$

where

Mc = vehicle dry mass contingency (lbm)

dw = dry mass of the vehicle without a contingency factor (lbm)

cf = contingency constant (lbm/lbm); typical values being

0.05 = use existing hardware

0.10 = conventional vehicle design

0.15 = a new vehicle design concept

### 2.13 Dry mass

The vehicle dry mass is the manufactured mass of the vehicle. It is the sum of Sections 2.1.1 through 2.12. No additional user input is required for this equation.

### 2.14 Propellant

#### 2.14.1 Total Usable Ascent Propellant

The total ascent propellant is the propellant burned in the vehicle main engines from vehicle liftoff to the vehicle reaching the main engine cutoff (MECO) condition. It does not include propellant used during main engine start or the residuals left in the propellant tanks after MECO. The sizing program assumes the extra volume in the main propellant tanks is book-kept by adjusting the propellant tank ullage factor (Section 2.2.3). The vehicle flight performance reserve propellant is included in the total usable ascent propellant. The sizing program assumes that the trajectory analysis calculation of the mission velocity required to reach orbit includes the flight performance reserve requirement. The mission velocity estimation section of the performance spreadsheet assumes the vehicle has a 1% flight performance reserve incorporated into the total ascent mission velocity requirement.

### 2.14.2 Residual Ascent Propellants

The residual ascent propellants include gaseous propellants and the pressurization gases in addition to the trapped ascent propellants. The residual ascent propellants are modeled as a function of the propellant used, the propellant mass and the engine thrust. The equation below is set up for either bipropellants or tripropellants. If the vehicle is using bipropellants, the number of fuels is set to one and one of the fuel constants (kf) and one of the thrust constants (kt) is set to zero. If the vehicle is using tripropellants, the number of fuels is set to two.

If burn flag (Section 2.5.1) is set to 3, Fv is the burn one engine vacuum thrust. If burn flag is set to 1 or 2, Fv is the sum of the burn one engine vacuum thrust and burn two engine vacuum thrust.

The sizing programs assumes the residual ascent propellants are vented overboard following the main engine cutoff (MECO) and prior to the use of the on-orbit OMS and RCS burns. This decision reduces the vehicle mass for the following mission phases. It also means that there will not be any ascent propellant in the vehicle after touchdown. This will simplify the vehicle operations. The residual ascent propellant mass equation is of the form:

$$mrf = kf1*Wox + kf2*Wfu1 + kf3*Wfu2 + Fv*(kt1 + kt2 + kt3)/(Nf+1)$$

where

| | | |
|---|---|---|
| mrf | = | mass of the residual and unusable fluids (lbm) |
| Wox | = | mass of the oxidizer (lbm) |
| Wfu1 | = | mass of fuel 1 (lbm) |
| Wfu2 | = | mass of fuel 2 (lbm) |
| Fv | = | vehicle vacuum thrust (lbf) |
| Nf | = | number of fuels (one if the vehicle is bipropellant, two if the vehicle is tripropellant) |
| kf1,2,3 | = | the residual and unusable fluid propellant mass constant (lbm/lbm); where |
| | = | 0.0038 if propellant is not hydrogen |
| | = | 0.0016 if propellant is hydrogen |
| kt1,2,3 | = | the residual and unusable fluid thrust constant (lbm/lbm), where |
| | = | 0.0010 if propellant is not hydrogen |
| | = | 0.0012 if propellant is hydrogen |

### 2.14.3 On-Orbit and Reentry OMS/RCS Propellant

The on-orbit and reentry OMS/RCS propellant requirements are calculated in this section. The sequence of burns is assumed to be the on-orbit OMS burn is first, the on-orbit RCS burn is second and the reentry RCS burn is third. The on-orbit OMS burn velocity budget includes the vehicle circularization burn, all on-orbit maneuvering burns and the deorbit burn. The on-orbit RCS burn velocity budget includes all vehicle attitude control burns. The reentry RCS burn velocity budget includes all vehicle attitude burns required to control the vehicle during reentry and landing. The RCS thrusters are required to maintain the vehicles attitude until the vehicle gets low enough into the atmosphere that aerodynamic forces can control the vehicle. The sizing program assumes the residual ascent propellants (Section 2.14.2) have been vented prior to the initial on-orbit OMS burn. The sizing program assumes the payload is on the vehicle during these burns.

The sizing program assumes that the residual and unusable OMS and RCS propellants are bookkept as part of the on-orbit and reentry burn velocity budget. The on-orbit and reentry OMS and RCS propellant loads are contained in the auxiliary propellant tank (Section 2.7). The side entry conical configuration has a larger RCS entry velocity budget because it must be held at a side slip angle to have a cross range capability. The on-orbit and reentry OMS and RCS propellant mass equations are of the form:

for the on-orbit OMS burn:

$$r = e^{(dvoms/(32.174*Ispoms))},$$

$$Wfoms = W*(1-(1/r)),$$

for the on-orbit RCS burn:

$$r = e^{(dvrcso/(32.174*Isprcso))},$$

$$Wfrcso = W*(1-(1/r)),$$

for the reentry RCS burn:

$$r = e^{(dvrcsr/(32.174*Isprcsr))},$$

$$Wfrcsr = W*(1-(1/r)),$$

where

r = mass ratio during the burn

W = vehicle mass at the start of the burn (lbm)

Wfoms = OMS on-orbit propellant requirement (lbm)

Wfrcso = RCS on-orbit propellant requirement (lbm)

Wfrcsr = RCS reentry propellant requirement (lbm)

dvoms = OMS on-orbit velocity budget (ft/sec); a typical value is
= 1140 (from Option 3 study)

dvrcso = RCS on-orbit velocity budget (ft/sec); a typical value is
= 155 (from Option 3 study)

dvrcsr = RCS reentry velocity budget (ft/sec); typical values are
= 40 (from Option 3 study, used on lifting body and winged body configurations)
= 80 (used on side entry conical configuration)

Ispoms = OMS on-orbit specific impulse (sec); a typical value is
462 (from Option 3 study, O2/H2 thruster)

Isprcso = RCS on-orbit specific impulse (sec); a typical value is
422 (from Option 3 study, O2/H2 thruster)

Isprcsr = RCS reentry specific impulse (sec); a typical value is
410 (from Option 3 study, O2/H2 thruster)

### 2.14.4 Landing Propellant

The landing propellant calculation applies only to the side entry conical VTOL vehicle configuration. In this program, the landing maneuver is modeled by assuming that the vehicle is oriented vertically and has reached terminal velocity at sea level. The vehicle decelerates at three g's. Two of these g's are used to slow down the vehicle and one of the g's is used to offset the effects of the earth's gravity. After the vehicle has decelerated to a stop, it hovers for a user input amount of time. The actual landing maneuver for side entry conical launch vehicle configuration is more complex and will need to be modeled off line. It will start with the vehicle in horizontal gliding flight. Some combination of engines and body flaps will be used to orient the vehicle vertically and slow the vehicle to a stop at some altitude. The vehicle will then land. When the actual velocity requirements for the landing maneuver have been found, adjust the hover time in this model so its resulting landing maneuver velocity requirement matches the actual landing maneuver velocity requirement.

The sizing program assumes the landing propellant is contained in the auxiliary propellant tank (see Section 2.7). The landing propellant mass equation is of the form:

$$tv = (2*wl/0.002378*Cd*A)^{0.5}$$
$$dvd = 3*tv/2$$
$$dvh = 32.174*th$$
$$dvl = dvd + dvh$$
$$r = e^{(dvl/(32.174*Ispl))}$$
$$Wfl = wl*(r-1),$$

where

| | | |
|---|---|---|
| r | = | mass ratio during the burn |
| wl | = | vehicle mass at landing (lbm) |
| Ispl | = | landing maneuver specific impulse (sec) |
| Wfl | = | landing maneuver propellant requirement (lbm) |
| wl | = | landed vehicle mass (lbm) |
| dvl | = | landing maneuver velocity budget (ft/sec) |
| dvd | = | landing maneuver deceleration velocity budget (ft/sec) |
| dvh | = | landing maneuver hover velocity budget (ft/sec) |
| tv | = | vehicle terminal velocity in the vertical orientation (ft/sec) |
| A | = | vehicle base area ($ft^2$) |
| Cd | = | vehicle drag coefficient, where<br>0.90 = estimate for a cone on its base |
| Ispl | = | landing maneuver specific impulse (sec), where<br>330 = typical sea level value for a tripropellant engine |
| th | = | hover time (sec), where<br>16.0 = time selected to bring landing maneuver velocity budget close to 1000 ft/sec |

### 2.14.5 Engine Restart Propellant

The engine restart propellant calculation applies only to the side entry conical VTOL vehicle configuration. This vehicle configuration will use some combination of body flaps and the mode one engines to rotate the vehicle flight path angle 90 degrees from gliding horizontal flight to standing on its tail for touchdown.

To prevent propellant boiling in cryogenic rocket engine turbopumps at engine start, the turbopump temperature must match the propellant temperature. Therefore, prior to engine start, the turbopumps are pre-chilled. The sizing program calculates how much propellant is required to perform this turbopump pre-chill assuming the Mode 1 engines are used for the landing maneuver. The sizing program also assumes that this engine restart propellant is lost overboard instead of being burned in the engines. Vehicle mass at engine restart is about 10% of the vehicle liftoff mass. If the Mode 1 engines have several turbopump sets, it may not be necessary to pre-chill all of them prior to the landing maneuver. An example would be if the Mode 1 engines had six turbopump sets and the required maximum thrust level is 10%, only one turbopump set would need to be pre-chilled. If you have a redundancy requirement of one turbopump set out, only two of the turbopump sets would need to be pre-chilled. If the full set of Mode 1 turbopumps are not used, reduce the restart propellant coefficients proportionally.

The sizing program assumes the landing propellant is contained in the auxiliary propellant tank (see section 2.7). The engine restart propellant mass equation is of the form:

Wfrs = Fv1*(kcdf+ Kcdo),

where

Wfrs = engine restart propellant (lbm)

Fv1 = Mode 1 engine vacuum thrust (lbf)

kcdf = fuel pre-chill coefficient (lbm/lbf), where

- 0.000000 = room temperature fuels such as kerosene or hydrazine do not need to pre-chill a fuel turbopump
- 0.000620 = mild cryogenic fuels such as methane
- 0.000920 = deep cryogenic fuel such as hydrogen

kcdo = oxidizer pre-chill coefficient (lbm/lbf), where

- 0.000000 = room temperature oxidizer such as nitrogen tetroxide, nitric acid or hydrogen peroxide does not need to pre-chill an oxidizer turbopump
- 0.000620 = mild cryogenic oxidizer such as oxygen

### 2.14.6 Ascent RCS Propellant

Vehicle roll control during ascent may be provided by the vehicle thrust vector control (TVC) system or by the vehicle RCS. The sizing program calculates the ascent RCS propellant load based on the assumption that the ascent RCS propellant is burned at liftoff. The ascent RCS propellant mass equation is of the form:

$$r = e^{(dvrcsa/(32.174*Isprcsa))}$$

$$Wfrcsa = GLOW*(r-1)$$

where

| | | |
|---|---|---|
| r | = | mass ratio during the burn |
| GLOW | = | vehicle mass at liftoff (lbm) |
| dvrcsa | = | the burn velocity budget (ft/sec) |
| Isprcsa | = | burn specific impulse (sec) |
| Wfrcsa | = | RCS ascent propellant requirement (lbm) |
| dvrcsa | = | RCS ascent velocity budget (ft/sec); typical values are |
| | = | 0 (vehicle TVC supplies roll control) |
| | = | 40 (estimated value for RCS thrusters providing vehicle roll control during ascent) |
| Isprcsa | = | RCS ascent specific impulse (sec); a typical value is |
| | = | 350 (estimated value for O2/H2 thruster at sea level) |

## 2.15 Burnout Mass (w/o payload)

This is the sum of Sections 2.13 through 2.14.5. The ascent RCS propellant (Section 7.14.6) is assumed to have been burned during ascent.

## 2.16 Personnel Provisions

Personal provisions include the fixed life support system, food, waste, and water management systems, fire detection, pilot and crew stations. The personnel provisions mass equation is of the form:

$$Mpp = PPf + PPs * Nc,$$

where

| | | |
|---|---|---|
| Mpp | = | Personnel provisions mass (lbm) |
| Nc | = | number of crew (crew member) |
| PPf | = | food waste and water management system (1 to 4 crew) (lbm), |
| | = | 0 (for missions of less than 24 hours) |
| | = | 353 (for missions of greater than 24 hours) |
| PPs | = | seats and other pilot and crew related items (lbm/crew member); a typical manned vehicle value is |
| | = | 167 |

## 2.17 Personnel

Personnel includes the mass of crew, mission specialists, etc., and personnel-related GFE equipment and accessories. The personnel mass equation is of the form:

$$Mper = Pm + Pp * Nc,$$

where

| | | |
|---|---|---|
| Mper | = | Personnel mass (lbm) |

| | | |
|---|---|---|
| Nc | = | number of crew (crew member) |
| Pm | = | miscellaneous (lbm); a typical manned value is |
| | = | 400 |
| Pp | = | personnel (lbm/crew member); a typical manned value is |
| | = | 540 |

### 2.18 Payload

This entry is the vehicle design payload. The payload support structure is under payload bay (Section 2.2.4). This sizing program assumes the vehicle lands with the design payload on board.

### 2.19 Gross Liftoff Weight (GLOW)

This entry is the vehicle mass at liftoff. It is the sum of Sections 2.15 through 2.18.

### 2.20 Vehicle Mass at MECO

This entry is the vehicle mass at main engine cutoff. It is the sum of Sections 2.15 and 2.18. This is prior to the venting of the residual ascent propellants (Section 2.14.2).

### 2.21 Landed Vehicle Mass

This entry is the mass of the vehicle at touchdown. The design payload is on the vehicle at touchdown. It is Section 2.20 minus Sections 2.14.3 through 2.14.5. The ascent RCS propellant (Section 2.14.6) is assumed to have been burned during ascent.

# 3. TEST CASE FILES

This part of the user's guide contains example input data files and printouts of the sizing tool spreadsheets. Use these spreadsheets as a test case to verify the correct operation of the sizing tools.

## 3.1 Side Entry Conical SSTO Vehicle Configuration Sizing Tool Test Case Files

This section contains the example input data files and printouts of the sizing tool spreadsheets for the side entry conical SSTO vehicle configuration sizing tool. The first input data file uses the Option 3 Access to Space Report version of an evolved SSME for the vehicle's main engines. This input data file was selected as an example of a bipropellant oxygen/hydrogen vehicle configuration. The second input data file uses the Option 3 Access to Space Report version of the Russian RD-701 engine. This input data file was selected as an example of a tripropellant oxygen/hydrogen/kerosene vehicle configuration. The following vehicle configuration spreadsheets used the RD-701 input data file for the vehicle configuration definition.

### 3.1.1 Option 3 Evolved SSME Input Data File

This is the input data file for the conical VTOL configuration with side entry.

Engine is a rubber option 3 evolved SSME engine.

Tank position, 1 = aft, 2= middle, 3= forward
Tank design, 1= common (concave bkld), 2= nested (concave bkld), 3= separate
Common bulkhead means the aft propellant tank has only one endcap.
Nested bulkhead means a one foot distance between fwd and aft propellant tank endcaps.
Endcap flag, 1= ellipsoidal, 2= hemispherical, 3= toroidal
endcaps
If endcap flag =1 height=dia/2*sqrt(2), =2 height=dia/2, =3 radius=.7*dia/2, height=.3*dia/2
If number of fuel tanks is set to 1, do not use the mid propellant tank or the Fuel 1 fuel tank.

Engine flag, 1= engines with a known thrust and weight are used, 2= engines with a known thrust to weight are used (engine thrust is allowed to vary).
Burn flag, 1= series burn, 2= parallel burn, 3= dual mode parallel burn

Payload bay location flag, 1= aft intertank, 2= forward intertank

| Input Data: | |
|---|---|
| Payload (Wpay) | 25,000 lbm |
| Payload bay diameter | 15.00 ft |
| Number of crew | 0.00 |
| Crew cabin volume | 0.00 ft^3 |
| Number of days on-orbit | 7.00 |
| Average on-orbit power usage | 5.00 kw |
| Average on-orbit heat rejection requirement | 10.00 kw |
| Maximum acceleration (No) | 3.000 g |
| Factor of safety | 1.40 |
| Orbit inclination | 51.60 deg |
| Orbit perigee | 50.00 NM |
| Orbit apogee | 100.00 NM |

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Position= | 3 | | 1 | |
| Ullage= | 0.05 | | 0.05 | |
| Density= | 71.20 | | 4.43 | lbm/ft^3 |
| Residual A= | 0.0038 | | 0.0016 | lbm/lbm |
| Residual B= | 0.0010 | | 0.0012 | lbm/lbf |
| Ullage pressure= | 35.00 | | 50.00 | psi |
| TPS unit mass= | 0.250 | | 0.250 | lbm/ft^2 |
| | Fwd Tank | Mid Tank | Aft Tank | |
| Forward endcap height coefficient= | 0.7071 | | 0.7071 | |
| Aft endcap height coefficient= | 0.7071 | | 0.7071 | |
| Tank design= | 3 | | 3 | |
| Upper endcap Flag= | 1 | | 1 | |
| Lower endcap flag= | 1 | | 1 | |

| Engine Restart Propellant Coefficients: | |
|---|---|
| Engine conditioning coefficient, fuel= | 0.000620 |
| Engine conditioning coefficient, oxidizer= | 0.000920 |

| Vehicle Materials: | Fwd Tank | Mid Tank | Aft Tank | Unpressurized Structure | |
|---|---|---|---|---|---|
| Stiffener constant A= | 2.3 | | 2.3 | 0.52 | |
| Stiffener constant B= | 0.44 | | 0.44 | 0.83 | |
| Density= | 0.098 | | 0.057 | 0.057 | lbm/in^3 |
| Ftu= | 65,600 | | 90,400 | | psi |
| Modulus of elasticity= | 9,900,000 | | 9,450,000 | 9,450,000 | psi |

| Mode 2 burn: | |
|---|---|
| Mode two mission velocity | 18,500 ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 sec |
| Iv2 | 447.30 sec |
| Mixture ratio (% Oxidizer) | 85.73 % |
| Mixture ratio (% Fuel 1) | 0.00 % |
| Mixture ratio (% Fuel 2) | 14.27 % |
| Engine height | 16.15 ft |
| Number of engines | 7 |
| Engine flag | 2 |
| Engine mass (if engine flag = 1) | 0 lbm |
| Engine vac thrust (if engine flag = 1) | 0 lbf |
| Engine unit mass (if engine flag = 2) | 71.06 lbf/lbm(vac) |
| No 2 (if engine flag = 2) | 1.400 g |

| Mode 1 burn: | |
|---|---|
| Isl1 | 390.40 sec |
| Iv1 | 447.30 sec |
| Mixture ratio (% Oxidizer) | 85.73 % |
| Mixture ratio (% Fuel 1) | % |
| Mixture ratio (% Fuel 2) | 14.27 % |
| Engine height | 16.15 ft |
| Number of engines | 7 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine sl thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 62.02 lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 g |

| | |
|---|---|
| Burn flag= | 3 |
| Number of fuel tanks= | 1 |
| Payload bay location flag= | 1 |

| Vehicle sizing coefficients: | |
|---|---|
| Main propellant feed line and press sys= | 55.00 lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 |
| Avionics= | 710 lbm/lbm^(1/8)) |
| Range safety= | 0 lbm |
| Crew Cabin Body Constant (Bo)= | 0 lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.035 lbm/lbm |
| Body insulation constant (Kbi)= | 0 lbm/ft^2 (if hot structure is used) |
| Base engine heat shield unit mass= | 1.64 lbm/ft^2 |
| | |
| Gimbal actuator unit mass= | 0.00129 lbm/lbf |
| Thrust structure (Kts1)= | 0.00207 lbm/lbf |
| Thrust structure (Kts2)= | 0.00039 lbm/lbf |
| | |
| Prime Power (PWc) (aero surface)= | 0.274 lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.0000373 lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface)= | 0.000 lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 lbm/kw |
| Fuel cell reactant unit mass (FCc)= | 29.26 lbm/kw-day |
| | |
| ECLSS crew cabin constant (Ec)= | 0.00 lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 lbm/kw |
| | |
| Personnel waste systems (PPf)= | 0.00 lbm |
| Personnel seats and crew related (PPs)= | 0.00 lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 lbm |
| Personnel weight (Pp)= | 0.00 lbm/crew member |
| | |
| Body Flap Unit Area= | 0.25 ft^2/ft^2 (Four flaps are assumed) |
| Body flap constant (Bbf)= | 1.17 lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 lbm/ft^2 |
| Control surface miscellaneous (Spc)= | 200 lbm |
| | |
| Payload bay mass= | 5,786 lbm (from the Langley SSTO(R) case) |
| Vehicle base diameter= | 61.20 ft |
| Vehicle cone angle= | 5.50 deg |
| Minimum gage factor= | 1.00 lbm/ft^2 |
| | |
| Maximum normal load case: Angle of attack= | 20.00 deg |
| Maximum normal load case: Qbar= | 95.10 psf |

| OMS and RCS systems mass coefficients: | | |
|---|---|---|
| RCS system mass coefficient= | 0.000151 | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 | sec |
| RCS thruster specific impulse (reentry)= | 410.00 | sec |
| RCS thruster specific impulse (ascent)= | 350.00 | sec |
| | | |
| OMS system thrust-to-weight= | 0.04 | g |
| OMS engine mass coefficient= | 0.035 | lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 | lbm/lbm |
| OMS thruster specific impulse= | 462.00 | sec |

The main propulsion system additional delta v may be used for additional on-orbit maneuvers and to adjust the mission velocity requirements based on trajectory analysis results.
The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential throttling is selected for vehicle thrust vector control.

| | |
|---|---|
| RCS ascent mission velocity (applied to GLOW)= | 0 ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 80 ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1,140 ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -71 ft/sec |
| Landing maneuver specific impulse= | 390.40 sec |
| Landing maneuver hover time= | 16.00 sec |
| Landing maneuver drag coefficient= | 0.90 |

| Nose definition: | | |
|---|---|---|
| Exterior angle 1= | 17.00 | deg |
| Exterior angle 2= | 30.00 | deg |
| Ratio r2/r1= | 0.80 | |
| Ratio r3/r1= | 0.20 | |

| TPS unit Masses: | | |
|---|---|---|
| Nose= | 2.20 | lbm/ft^2 |
| Windward Fwd tank= | 0.65 | lbm/ft^2 |
| Leeward Fwd tank= | 0.50 | lbm/ft^2 |
| Windward Fwd Intertank= | 0.60 | lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 | lbm/ft^2 |
| Windward Mid tank= | 0.55 | lbm/ft^2 |
| Leeward Mid tank= | 0.35 | lbm/ft^2 |
| Windward Aft Intertank= | 0.50 | lbm/ft^2 |
| Leeward Aft Intertank= | 0.35 | lbm/ft^2 |
| Windward Aft tank= | 0.50 | lbm/ft^2 |
| Leeward Aft tank= | 0.35 | lbm/ft^2 |
| Windward Aft Skirt= | 0.65 | lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 | lbm/ft^2 |
| Body Flaps= | 1.50 | lbm/ft^2 |

### 3.1.2 RD-701 Input Data File

This is the input data file for the conical VTOL configuration with side entry.

Engine is a rubber RD-701 engine.

Tank position, 1 = aft, 2= middle, 3= forward
Tank design, 1= common (concave bkld), 2= nested (concave bkld), 3= separate
Common bulkhead means the aft propellant tank has only one endcap.
Nested bulkhead means a one foot distance between fwd and aft propellant tank endcaps.
Endcap flag, 1= ellipsoidal, 2= hemispherical, 3= toroidal endcaps
If endcap flag=1 height=dia/2*sqrt(2), =2 height=dia/2, =3 radius=.7*dia/2, height=.3*dia/2
If number of fuel tanks is set to 1, do not use the mid propellant tank or the Fuel 1 fuel tank.

Engine flag, 1= engines with a known thrust and weight are used, 2= engines with a known thrust and weight are used (engine thrust is allowed to vary).
Burn flag, 1= series burn, 2= parallel burn, 3= dual mode parallel burn

Payload bay location flag, 1= aft intertank, 2= forward intertank

| Input Data: | |
|---|---|
| Payload (Wpay) | 25,000 lbm |
| Payload bay diameter | 15.00 ft |
| Number of crew | 0.00 |
| Crew cabin volume | 0.00 ft^3 |
| Number of days on-orbit | 7.00 |
| Average on-orbit power usage | 5.00 kw |
| Average on-orbit heat rejection requirement | 10.00 kw |
| Maximum acceleration (No) | 3.000 g |
| Factor of safety | 1.40 |
| Orbit inclination | 51.60 deg |
| Orbit perigee | 50.00 NM |
| Orbit apogee | 100.00 NM |

| Tank definition: | | | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|---|---|
| Position= | | | 3 | 2 | 1 | |
| Ullage= | | | 0.05 | 0.05 | 0.05 | |
| Density= | | | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Residual A= | | | 0.0038 | 0.0038 | 0.0016 | lbm/lbm |
| Residual B= | | | 0.0010 | 0.0010 | 0.0012 | lbm/lbf |
| Ullage pressure= | | | 35.00 | 35.00 | 50.00 | psi |
| TPS unit mass= | | | 0.250 | 0.000 | 0.250 | lbm/ft^2 |
| | | | | | | |
| | | | Fwd Tank | Mid Tank | Aft Tank | |
| Forward endcap height coefficient= | | | 0.7071 | 0.7071 | 0.7071 | |
| Aft endcap height coefficient= | | | 0.7071 | 0.7071 | 0.7071 | |
| Tank design= | | | 3 | 2 | 3 | |
| Upper endcap Flag= | | | 1 | 1 | 1 | |
| Lower endcap flag= | | | 1 | 1 | 1 | |

| Engine Restart Propellant Coefficients: | |
|---|---|
| Engine conditioning coefficient, fuel= | 0.000620 |
| Engine conditioning coefficient, oxidizer= | 0.000920 |

| | | | | | Unpressurized | |
|---|---|---|---|---|---|---|
| Vehicle Materials: | | Fwd Tank | Mid Tank | Aft Tank | Structure | |
| Stiffener constant A= | | 2.3 | 2.3 | 2.3 | 0.52 | |
| Stiffener constant B= | | 0.44 | 0.44 | 0.44 | 0.83 | |
| Density= | | 0.098 | 0.098 | 0.057 | 0.057 | lbm/in^3 |
| Ftu= | | 65,600 | 65,600 | 90,400 | | psi |
| Modulus of elasticity= | | 9,900,000 | 9,900,000 | 9,450,000 | 9,450,000 | psi |

| Mode 2 burn: | |
|---|---|
| Mode two mission velocity | 20,040 ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 sec |
| Iv2 | 452.70 sec |
| Mixture ratio (% Oxidizer) | 85.70 % |
| Mixture ratio (% Fuel 1) | 0.00 % |
| Mixture ratio (% Fuel 2) | 14.30 % |
| Engine height | 11.67 ft |
| Number of engines | 7 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine vac thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 40.26 lbf/lbm(vac) |
| No2 (if engine flag= 2) | 1.400 g |

| Mode 1 burn: | |
|---|---|
| Isl1 | 333.50 sec |
| Iv1 | 385.10 sec |
| Mixture ratio (% Oxidizer) | 76.80 % |
| Mixture ratio (% Fuel 1) | 20.20 % |
| Mixture ratio (% Fuel 2) | 3.00 % |
| Engine height | 11.87 ft |
| Number of engines | 7 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine sl thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 82.90 lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 g |

| | | |
|---|---|---|
| Burn flag= | | 3 |
| Number of fuel tanks= | | 2 |
| Payload bay location flag= | | 1 |

| Vehicle sizing coefficients: | |
|---|---|
| Main propellant feed line and press sys= | 55.00 lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 |
| Avionics= | 710 lbm/lbm^(1/8) |
| Range safety= | 0 lbm |
| Crew Cabin Body Constant (Bo)= | 0 lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.035 lbm/lbm |
| Body insulation constant (Kbi)= | 0 lbm/ft^2 (if hot structure is used) |
| Base engine heat shield unit mass= | 1.64 lbm/ft^2 |
| | |
| Gimbal actuator unit mass= | 0 lbm/lbf |
| Thrust structure (Kts1)= | 0.00207 lbm/lbf |
| Thrust structure (Kts2)= | 0.00039 lbm/lbf |
| | |
| Prime Power (PWc) (aero surface)= | 0.274 lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.0000000 lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface)= | 0.000 lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 lbm/kw |
| Fuel cell reactant unit mass (FCc)= | 29.26 lbm/kw-day |
| | |
| ECLSS crew cabin constant (Ec)= | 0.00 lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 lbm/kw |
| | |
| Personnel waste systems (PPf)= | 0.00 lbm |
| Personnel seats and crew related (PPs)= | 0.00 lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 lbm |
| Personnel weight (Pp)= | 0.00 lbm/crew member |
| | |
| Body Flap Unit Area= | 0.25 ft^2/ft^2 (Four flaps are assumed) |
| Body flap constant (Bbf)= | 1.17 lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 lbm/ft^2 |
| Control surface miscellaneous (Spc)= | 200 lbm |
| | |
| Payload bay mass= | 5,786 lbm (from the Langley SSTO(R) case) |
| Vehicle base diameter= | 48.40 ft |
| Vehicle cone angle= | 5.50 deg |
| Minimum gage factor= | 1.00 lbm/ft^2 |
| | |
| Maximum normal load case: Angle of attack= | 20.00 deg |
| Maximum normal load case: Qbar= | 95.10 psf |

| OMS and RCS systems mass coefficients: | |
|---|---|
| RCS system mass coefficient= | 0.000151 lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 sec |
| RCS thruster specific impulse (reentry)= | 410.00 sec |
| RCS thruster specific impulse (ascent)= | 350.00 sec |
| OMS system thrust-to-weight= | 0.04 g |
| OMS engine mass coefficient= | 0.035 lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 lbm/lbm |
| OMS thruster specific impulse= | 462.00 sec |

The main propulsion system additional delta v may be used for additional on-orbit maneuvers and to adjust the mission velocity requirements based on trajectory analysis results.
The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential throttling is selected for vehicle thrust vector control.

| | |
|---|---|
| RCS ascent mission velocity (applied to GLOW)= | 0 ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 80 ft/sec |
| OMS system on-orbit mission velocity (applied to MECO-residuals)= | 1,140 ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -71 ft/sec |
| Landing maneuver specific impulse= | 333.50 sec |
| Landing maneuver hover time= | 16.00 sec |
| Landing maneuver drag coefficient= | 0.90 |

| Nose definition: | | |
|---|---|---|
| Exterior angle 1= | 17.00 | deg |
| Exterior angle 2= | 30.00 | deg |
| Ratio r2/r1= | 0.80 | |
| Ratio r3/r1= | 0.20 | |

| TPS unit Masses: | | |
|---|---|---|
| Nose= | 2.20 | lbm/ft^2 |
| Windward Fwd tank= | 0.65 | lbm/ft^2 |
| Leeward Fwd tank= | 0.50 | lbm/ft^2 |
| Windward Fwd Intertank= | 0.60 | lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 | lbm/ft^2 |
| Windward Mid tank= | 0.55 | lbm/ft^2 |
| Leeward Mid tank= | 0.35 | lbm/ft^2 |
| Windward Aft Intertank= | 0.50 | lbm/ft^2 |
| Leeward Aft Intertank= | 0.35 | lbm/ft^2 |
| Windward Aft tank= | 0.50 | lbm/ft^2 |
| Leeward Aft tank= | 0.35 | lbm/ft^2 |
| Windward Aft Skirt= | 0.65 | lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 | lbm/ft^2 |
| Body Flaps= | 1.50 | lbm/ft^2 |

### 3.1.3 Airloads Spreadsheet

LOADS ANALYSIS
LOAD CASE 1: Max bending moment

| | |
|---|---|
| MASS T = | 107428.22 |
| Xcg = | 36.00 |
| Gn = | 0.01 |

| THETAdd |
|---|
| 0.000000 |
| -0.006230 |

| Xi (m) | Wi (kg) | norm for (kN) | Wi*Xi | DEL X | DELX2 | Wi*DX2 | gn | Wi*Gn | SUM N |
|---|---|---|---|---|---|---|---|---|---|
| 4.441 | 1,516 | 1.66 | 6,730 | -31.56 | 996.10 | 1,509,229 | 0.03 | 0.42 | 1.25 |
| 12.093 | 8,3177 | 0.00 | 101,300 | -23.91 | 571.68 | 4,788,703 | 0.02 | 1.90 | -1.90 |
| 13.903 | | 1.36 | 0 | -22.10 | 488.40 | 0 | 0.02 | 0.00 | 1.36 |
| 17.676 | 92 | | 1,626 | -18.33 | 335.88 | 30,904 | 0.02 | 0.02 | -0.02 |
| 17.727 | | 0.16 | 0 | -18.28 | 334.02 | 0 | 0.02 | 0.00 | 0.16 |
| 18.936 | 2,602 | | 49,279 | -17.07 | 291.29 | 758,075 | 0.02 | 0.48 | -0.48 |
| 19.305 | | 0.06 | 0 | -16.70 | 278.83 | 0 | 0.02 | 0.00 | 0.06 |
| 26.974 | 26,181 | | 706,204 | -9.03 | 81.52 | 2,134,107 | 0.01 | 3.52 | -3.52 |
| 29.285 | | 2.11 | 0 | -6.72 | 45.13 | 0 | 0.01 | 0.00 | 2.11 |
| 35.327 | 3,814 | 0.00 | 134,747 | -0.68 | 0.46 | 1,744 | 0.01 | 0.31 | -0.31 |
| 35.801 | 0 | 0.89 | 0 | -0.20 | 0.04 | 0 | 0.01 | 0.00 | 0.89 |
| 43.394 | 0 | 2.16 | 0 | 7.39 | 54.63 | 0 | 0 | 0.00 | 2.16 |
| 44.225 | 64,847 | 0.00 | 2,867,853 | 8.22 | 67.60 | 4,383,461 | 0 | 1.75 | 1.75 |
| | | | | | | | | | |
| 46.717 | | | 0 | 10.71 | 114.79 | 0 | 0 | 0 | 0 |
| | 107,428 | 8.40 | 3,867,740 | | | 13,606,224 | | | |

| | | |
|---|---|---|
| Q = | 95.1 | psf |
| Base Diameter = | 48.40 | ft |

| AREF | Q (kPa) | ALPHA |
|---|---|---|
| 1.2969 | 4.553388 | 2.0 |

| | CAP | FRUS1 | FRUS2 | FRUS3 | FRUS4 | FRUS5 | FRUS6 | TOTAL | BASE |
|---|---|---|---|---|---|---|---|---|---|
| CNALPHA | 0.01407 | 0.01148 | 0.00134 | 0.00052 | 0.0179 | 0.0075 | 0.01827 | 0.07108 | |
| NF | 1.661773 | 1.355874 | 0.158264 | 0.061416 | 2.114124 | 0.885806 | 2.157824 | 8.395082 | |
| CA | 0.39498 | 0.01971 | 0.00229 | 0.00089 | 0.03070 | 0.01288 | 0.03135 | 0 | 0.0211 |

| SHER | Xi + 1 - Xi | SHE*P | MOM |
|---|---|---|---|
| (kN) | | | (kN * M) |
| 1.25 | | | 0.00 |
| -0.66 | 7.65 | 9.53 | 9.53 |
| 0.70 | 1.81 | -1.19 | 8.34 |
| 0.68 | 3.77 | 2.64 | 10.98 |
| 0.84 | 0.05 | 0.03 | 11.01 |
| 0.36 | 1.21 | 1.02 | 12.03 |
| 0.42 | 0.37 | 0.13 | 12.16 |
| -3.10 | 7.67 | 3.23 | 15.39 |
| -0.98 | 2.31 | -7.16 | 8.23 |
| -1.30 | 6.04 | -5.94 | 2.29 |
| -0.41 | 0.47 | -0.61 | 1.68 |
| 1.75 | 7.59 | -3.13 | -1.45 |
| 0.00 | 0.83 | 1.45 | 0 |
| | | | |
| 0.00 | 2.49 | 0 | 0 |

| Station | Moment | Station | Moment | |
|---|---|---|---|---|
| (M) | (kN-m) | (ft) | (ft-lb) | |
| 4.44 | 0 | 14.57 | 0 | Nose |
| 12.09 | 10 | 39.68 | 7,028 | |
| 13.90 | 8 | 45.62 | 6,151 | Fwd Tank |
| 17.68 | 11 | 57.99 | 8,096 | |
| 17.73 | 11 | 58.16 | 8,121 | Fwd Intertank |
| 18.94 | 12 | 62.13 | 8,870 | |
| 19.30 | 12 | 63.34 | 8,968 | Mid Tank |
| 26.97 | 15 | 88.50 | 11,349 | |
| 29.28 | 8 | 96.08 | 6,071 | Aft Intertank |
| 35.33 | 2 | 115.91 | 1,689 | |
| 35.80 | 2 | 117.46 | 1,236 | Aft Tank |
| 43.39 | -1 | 142.38 | -1,07 | |
| 44.22 | 0 | 145.10 | 0 | |
| 46.72 | 0 | 153.28 | 0 | Aft Skirt |

| Component Moments | | |
|---|---|---|
| Component | Moment | |
| Nose = | 0 | ft-lb |
| Fwd Tank = | 7,028 | ft-lb |
| Fwd Intertank = | 8,121 | ft-lb |
| Mid Tank = | 8,968 | ft-lb |
| Aft Intertank = | 11,349 | ft-lb |
| Aft Tank = | 1,689 | ft-lb |
| Aft Skirt = | 1,070 | ft-lb |

| Vehicle Station | |
|---|---|
| Component | Station (ft) |
| Nose = | 0.00 |
| Fwd tank fwd edge = | 21.86 |
| Fwd tank aft edge = | 57.49 |
| Mid tank fwd edge = | 58.49 |
| Mid tank aft edge = | 65.76 |
| Aft tank fwd edge = | 111.25 |
| Aft tank aft edge = | 120.57 |
| Base = | 153.28 |

| Vehicle Mass | | |
|---|---|---|
| Nose = | 3,341 | lbm |
| Fwd tank = | 18,470 | lbm |
| Fwd intertank = | 203 | lbm |
| Mid tank = | 5,738 | lbm |
| Aft intertank = | 10,825 | lbm |
| Aft tank = | 8,411 | lbm |
| Aft skirt = | 142,988 | lbm |
| Dry weight = | 189,976 | lbm |
| Ascent RCS = | 0 | lbm |
| Descent RCS/OMS prop = | 23,934 | lbm |
| Residuals = | 12,158 | lbm |
| Engine restart propellant = | 5,487 | lbm |
| Landing propellant = | 21,903 | lbm |
| Burnout weight = | 253,459 | lbm |
| Ascent prop = | 2,293,077 | lbm |
| Payload = | 25,000 | lbm |
| Total = | 2,571,536 | lbm |

Engine restart prop., landing prop and payload added to payload bay location

Payload bay location flag = 1

Mass at max normal conditions (post reentry)

| Component: | Mass (lbm) | Mass (kg) | cg (ft) | cg (m) | cp (ft) | cp (m) |
|---|---|---|---|---|---|---|
| Nose = | 3,341 | 1,515 | 14.57 | 4.44 | 14.57 | 4.44 |
| Fwd tank = | 18,470 | 8,377 | 39.68 | 12.09 | 45.62 | 13.90 |
| Fwd intertank = | 203 | 92 | 57.99 | 17.68 | 58.16 | 17.73 |
| Mid tank = | 5,738 | 2,602 | 62.13 | 18.94 | 63.34 | 19.30 |
| Aft intertank = | 57,728 | 26,181 | 88.50 | 26.97 | 96.08 | 29.28 |
| Aft tank = | 8,411 | 3,814 | 115.91 | 35.33 | 117.46 | 35.80 |
| Aft skirt = | 142,988 | 64,847 | 145.10 | 44.22 | 142.38 | 43.39 |

NOTE: cg is in center of element (in aft skirt, cg is three fourths of the way back)
cp is two thirds of the way back

### 3.1.4 Loads Spreadsheet

| |
|---|
| This spreadsheet is used to calculate the load intensity on the vehicles structure. |
| From this, the structure unit weights are calculated. |
| |
| Side entry conical VTOL vehicle configuration. |
| |
| Tank position is 1= aft, 2= mid, 3= fwd |
| |
| Linear loading is converted into unit surface loading by the use of a graph (Fig 2A-8:14) |
| in "Aerospace Vehicle Design, Volume II, Spacecraft Design", by K. D. Wood |
| The input is kilo pounds per inch of load on the circumference on the booster. |

| | | |
|---|---|---|
| Configuration definition: | | |
| Maximum vehicle acceleration (No)= | [illegible] | g |
| Factor of safety (FS)= | [illegible] | |
| Burn one vacuum thrust= | [illegible] | lbf |
| Burn two vacuum thrust= | [illegible] | lbf |
| | | |
| Burn 1: | | |
| Initial throttle setting= | [illegible] | % |
| End of burn thrust to weight= | [illegible] | g |
| Initial propellant: | | |
| Oxidizer= | [illegible] | lbm |
| Fuel 1= | [illegible] | lbm |
| Fuel 2= | [illegible] | lbm |
| | | |
| Burn 2: | | |
| Initial throttle setting= | [illegible] | % |
| End of burn thrust to weight= | [illegible] | g |
| Initial propellant: | | |
| Oxidizer= | [illegible] | lbm |
| Fuel 1= | [illegible] | lbm |
| Fuel 2= | [illegible] | lbm |

| | | | | | |
|---|---|---|---|---|---|
| Stage masses: | | | | | |
| Nose= | | | [illegible] | lbm | |
| Fwd tank= | | | [illegible] | lbm | |
| Fwd intertank= | | | [illegible] | lbm | |
| Mid tank= | | | [illegible] | lbm | |
| Aft intertank= | | | [illegible] | lbm | |
| Aft tank= | | | [illegible] | lbm | |
| Aft skirt= | | | [illegible] | lbm | |
| Dry Weight= | | | [illegible] | lbm | |
| | | | | | |
| | | | | | |
| Ascent RCS= | | | [illegible] | lbm | (burned during ascent) |
| On orbit & entry RCS/OMS prop= | | | [illegible] | lbm | |
| Residuals= | | | [illegible] | lbm | |
| Landing Propellant= | | | [illegible] | lbm | |
| Engine Restart Propellant= | | | [illegible] | lbm | |
| Burn out mass= | | | [illegible] | lbm | |
| | | | | | |
| Ascent prop= | | | [illegible] | lbm | |
| Payload= | | | 25,000 | lbm | |
| Total= | | | [illegible] | lbm | |
| | | | | | |
| Burn 1 propellant= | | | [illegible] | lbm | |
| Burn 2 Propellant= | | | [illegible] | lbm | |

| | | | |
|---|---|---|---|
| Tank sequence | | | |
| Oxidizer tank= | | | [illegible] |
| Fuel tank 1= | | | [illegible] |
| Fuel tank 2= | | | [illegible] |

| | | |
|---|---|---|
| Number of fuel tanks= | | [redacted] |
| Payload bay location flag= | | [redacted] |

| | Propellant tank side wall structures | | | | Unpressurized | |
|---|---|---|---|---|---|---|
| | | Fwd Tank | Mid Tank | Aft Tank | Structure | |
| Stiffener constant A= | [redacted] | [redacted] | [redacted] | [redacted] | [redacted] | |
| Stiffener constant B= | [redacted] | [redacted] | [redacted] | [redacted] | [redacted] | |
| Density (lbm/in^3)= | [redacted] | [redacted] | [redacted] | [redacted] | [redacted] | |
| Modulus of elasticity (psi)= | [redacted] | [redacted] | [redacted] | [redacted] | [redacted] | |

| Correction factor for unit weights on a conic surface | | | | |
|---|---|---|---|---|
| (K. D. Wood, fig 2A-8:13) | (40 & 50 deg points were extrapolated) | | | |
| Angle (deg) | K | | | |
| 0 | 1.00 | 1 | | |
| 10 | 1.10 | 2 | | |
| 20 | 1.28 | 3 | | |
| 30 | 1.50 | 4 | | |
| 40 | 1.77 | 5 | | |
| 50 | 2.15 | 6 | | |

| | | radius (ft) | length (ft) | angle (deg) | conic correction | | | Station (ft) |
|---|---|---|---|---|---|---|---|---|
| Nose | | 0 | | | | | | [redacted] |
| Fwd tank fwd radius= | | [redacted] | [redacted] | [redacted] | [redacted] | (nose height) | | [redacted] |
| Fwd tank aft radius= | | [redacted] | [redacted] | [redacted] | [redacted] | (fwd tank barrel section) | | [redacted] |
| Mid tank fwd radius= | | [redacted] | [redacted] | [redacted] | [redacted] | (fwd intertank) | | [redacted] |
| Mid tank aft radius= | | [redacted] | [redacted] | [redacted] | [redacted] | (mid tank barrel section) | | [redacted] |
| Aft tank fwd radius= | | [redacted] | [redacted] | [redacted] | [redacted] | (mid intertank) | | [redacted] |
| Aft tank aft radius= | | [redacted] | [redacted] | [redacted] | [redacted] | (aft tank barrel section) | | [redacted] |
| Base radius= | | [redacted] | [redacted] | [redacted] | [redacted] | (aft skirt) | | [redacted] |
| Total= | | | [redacted] | | | | | |

ascent RCS, descent OMS/RCS, engine restart prop, landing prop and payload added to payload bay location
residuals added to aft skirt

Load= mass of all components associated with this element
Load^= mass of this element and the mass of all elements above it
Linear Load= load^/element circumference
Linear Load^= linear load converted to units of klbf/in
Unit Mass= vehicle element unit mass at vehicle diameter (referenced to a ten foot diameter cylinder)
Unit Mass^= vehicle element unit mass at vehicle element base diameter
Conic Cor= vehicle element unit mass corrected to being a cone instead of being a cylinder
Material Cor= vehicle unit mass corrected to actual material instead of aluminum

The amount of propellant the vehicle sees at max acceleration depends on when the maximum acceleration point is.
If maximum vehicle acceleration is reached during burn one, the stage will see only the remaining burn one and all the burn two propellan
If the first stage reaches maximum acceleration at the end of burn one, there will be no burn one propellant remaining.
The first stage aft skirt sees only the gross lift off weight of the vehicle.



Vehicle at maximum acceleration during burn 1
The maximum burn one acceleration will be at the end of burn one or the maximum vehicle acceleration.

| End of burn | Max vehicle | Max | |
|---|---|---|---|
| [redacted] | [redacted] | [redacted] | g |

| | | |
|---|---|---|
| First stage mass at max vehicle acceleration= | [redacted] | lbm |
| First stage propellant remaing at max accele[redacted] | | lbm |
| Oxidizer= | [redacted] | lbm |
| Fuel 1= | [redacted] | lbm |
| Fuel 2= | [redacted] | lbm |
| Maximum burn one acceleration= | [redacted] | g |

| | Load | load^ | Linear Load | Linear Load^ | Unit Weight | Unit Weight^ | conic cor. | material cor. |
|---|---|---|---|---|---|---|---|---|
| | | | (lbf/ft | (k | (lbm/ft^2 | (lbm/ft^2 | (lbm/ft^2 | (lbm/ft^2 |
| Nose= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Fwd tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Fwd intertank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Mid tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft intertank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft skirt= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| Vehicle at maximum acceleration during burn 2 | | | | | | | | |
| The maximum burn two acceleration will be at the end of burn two or the maximum vehicle acceleration. | | | | | | | | |
| End of burn | Max vehicle | Max | | | | | | |
| [illegible] | [illegible] | [illegible] | g | | | | | |
| | | | | | | | | |
| ~~First stage mass at max vehicle acceleration=~~ | | | [illegible] | lbm | | | | |
| ~~First stage propellant remaing at max acceleration=~~ | | | [illegible] | lbm | | | | |
| Oxidizer= | | | [illegible] | lbm | | | | |
| Fuel 1= | | | [illegible] | lbm | | | | |
| Fuel 2= | | | [illegible] | lbm | | | | |
| Maximum burn two acceleration= | | | [illegible] | g | | | | |
| | | | | | | | | |
| | Load | load^ | Linear Load | Linear Load^ | Unit Weight | Unit Weight^ | conic cor. | material cor. |
| | | | (lbf/ft | (k | (lbm/ft^2 | (lbm/ft^2 | (lbm/ft^2 | (lbm/ft^2 |
| Nose= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Fwd tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Fwd intertank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Mid tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft intertank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft skirt= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |

| Maximum bending moment during reentry (moments are from spreadsheet AIRLOADS) | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| | Moment (ft-lb) | Radius (ft) | Equivalent load (lbf/ft) | Linear Load^ (lbf/in) | Unit Weight (lbm/ft^2) | Unit Weight^ (lbm/ft^2) | conic cor. (lbm/ft^2) | material cor. (lbm/ft^2) |
| Nose= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Fwd tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Fwd intertank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Mid tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft intertank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft tank= | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft skirt= | [illegible] | 24.20 | [illegible] | [illegible] | [illegible] | 0.026 | [illegible] | [illegible] |

| A logical test is preformed to find the larger of the axial or moment loads. | | | |
|---|---|---|---|
| Use a minimum gage factor= | [illegible] | lbm/ft^2 | |
| 1 | Axial | Bending Moments | Maximum | Maximum' |
| Nose= | [illegible] | 0.000 | [illegible] | [illegible] |
| Fwd tank= | [illegible] | [illegible] | [illegible] | [illegible] |
| Fwd intertank= | [illegible] | [illegible] | [illegible] | [illegible] |
| Mid tank= | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft intertank= | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft tank= | [illegible] | [illegible] | [illegible] | [illegible] |
| Aft skirt= | [illegible] | [illegible] | [illegible] | [illegible] |

### 3.1.5 Performance Spreadsheet

This spread sheet is used to calculate the size of a dual burn single stage to orbit vehicle.

This model is a conical VTOL configuration with side entry.

| Input Data: | |
|---|---|
| Payload (Wpay) | 25,000 lbm |
| Payload bay diameter | 15.00 ft |
| Number of crew | 0.00 |
| Crew cabin volume | 0.00 ft^3 |
| Number of days on-orbit | 7.00 |
| Average on-orbit power usage | 5.00 kw |
| Average on-orbit heat rejection requirement | 10.00 kw |
| Maximum acceleration (No) | 3.000 g |
| Factor of safety | 1.40 |
| Orbit inclination | 51.60 deg |
| Orbit perigee | 50.00 NM |
| Orbit apogee | 100.00 NM |

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank |
|---|---|---|---|
| Position= | 3 | 2 | 1 |
| Ullage= | 0.05 | 0.05 | 0.05 |
| Density= | 71.20 | 50.50 | 4.43 lbm/ft^3 |
| Residual A= | 0.0038 | 0.0038 | 0.0016 lbm/lbm |
| Residual B= | 0.0010 | 0.0010 | 0.0012 lbm/lbf |
| Ullage pressure= | 35.00 | 35.00 | 50.00 psi |
| TPS unit mass= | 0.250 | 0.000 | 0.250 lbm/ft^2 |
| | Fwd Tank | Mid Tank | Aft Tank |
| Forward endcap height coefficient= | 0.7071 | 0.7071 | 0.7071 |
| Aft endcap height coefficient= | 0.7071 | 0.7071 | 0.7071 |
| Tank design= | 3 | 2 | 3 |
| Upper endcap Flag= | 1 | 1 | 1 |
| Lower endcap flag= | 1 | 1 | 1 |

| Engine Restart Propellant Coefficients: | |
|---|---|
| Engine conditioning coefficient, fuel= | 0.000620 |
| Engine conditioning coefficient, oxidizer= | 0.000920 |

| Vehicle Materials: | Fwd Tank | Mid Tank | Aft Tank | Unpressurized Structure | |
|---|---|---|---|---|---|
| Stiffener constant A= | 2.3 | 2.3 | 2.3 | 0.52 | |
| Stiffener constant B= | 0.44 | 0.44 | 0.44 | 0.83 | |
| Density= | 0.098 | 0.098 | 0.057 | 0.057 | lbm/in^3 |
| Ftu= | 65,600 | 65,600 | 90,400 | | psi |
| Modulus of elasticity= | 9,900,000 | 9,900,000 | 9,450,000 | 9,450,000 | psi |

| Mode 2 burn: | | |
|---|---|---|
| Mode two mission velocity | 20,040 | ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 | sec |
| Iv2 | 452.70 | sec |
| Mixture ratio (% Oxidizer) | 85.70 | % |
| Mixture ratio (% Fuel 1) | 0.00 | % |
| Mixture ratio (% Fuel 2) | 14.30 | % |
| Engine height | 11.67 | ft |
| Number of engines | 7 | |
| Engine flag | 2 | |
| Engine mass (if engine flag= 1) | 0 | lbm |
| Engine vac thrust (if engine flag= 1) | 0 | lbf |
| Engine unit mass (if engine flag= 2) | 40.26 | lbf/lbm(vac) |
| No2 (if engine flag= 2) | 1.400 | g |

| Mode 1 burn: | | |
|---|---|---|
| Isl1 | 333.50 | sec |
| Iv1 | 385.10 | sec |
| Mixture ratio (% Oxidizer) | 76.80 | % |
| Mixture ratio (% Fuel 1) | 20.20 | % |
| Mixture ratio (% Fuel 2) | 3.00 | % |
| Engine height | 11.87 | ft |
| Number of engines | 7 | |
| Engine flag | 2 | |
| Engine mass (if engine flag= 1) | 0 | lbm |
| Engine sl thrust (if engine flag= 1) | 0 | lbf |
| Engine unit mass (if engine flag= 2) | 82.90 | lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 | g |

| | |
|---|---|
| Burn flag= | 3 |
| Number of fuel tanks= | 2 |
| Payload bay location flag= | 1 |

| Vehicle sizing coefficients: | | |
|---|---|---|
| Main propellant feed line and press sys= | 55.00 lbm-sec/ft^3 | |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 lbm/lbm^(1/8) | |
| Range safety= | 0 lbm | |
| Crew Cabin Body Constant (Bo)= | 0 lbm/number of crew^(1/2) | |
| Landing gear constant (Kl)= | 0.035 lbm/lbm | |
| Body insulation constant (Kbi)= | 0 lbm/ft^2 | (if hot structure is used) |
| Base engine heat shield unit mass= | 1.64 lbm/ft^2 | |
| | | |
| Gimbal actuator unit mass= | 0 lbm/lbf | |
| Thrust structure (Kts1)= | 0.00207 lbm/lbf | |
| Thrust structure (Kts2)= | 0.00039 lbm/lbf | |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 lbm/ft^2 | |
| Prime Power (PWe) (engine gimbaling)= | 0.0000000 lbm/lbf | |
| Prime Power (PWa) (avionics)= | 0.155 lbm/lbm | |
| Electrical Power Conv & Dist= | 0.020 lbm/lbm | |
| Hydraulic Power Conv & Dist (aero surface) = | 0.000 lbm/ft^2 | |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 lbm/lbf | |
| Fuel cell unit mass (FCw)= | 28.71 lbm/kw | |
| Fuel cell reactant unit mass (FCc)= | 29.26 lbm/kw-day | |
| | | |
| ECLSS crew cabin constant (Ec)= | 0.00 lbm/(ft^(1/3))^0.75 | |
| ECLSS crew supplies constant (Eo)= | 0.00 lbm/crew member-day | |
| ECLSS avionics waste heat (Ea)= | 0.22 lbm/lbm | |
| Active thermal control loop unit mass (Ew)= | 200.00 lbm/kw | |
| | | |
| Personal waste systems (PPf)= | 0.00 lbm | |
| Personal seats and crew related (PPs)= | 0.00 lbm/crew member | |
| Personal miscellaneous (Pm)= | 0.00 lbm | |
| Personal weight (Pp)= | 0.00 lbm/crew member | |
| | | |
| Body Flap Unit Area= | 0.25 ft^2/ft^2 | Four flaps are assumed |
| Body flap constant (Bbf)= | 1.17 lbm/(ft^2)^1.15 | |
| Control surface actuator constant (Ssc)= | 2.61 lbm/ft^2 | |
| Control surface miscellaneous (Spc)= | 200 lbm | |
| | | |
| Payload bay mass= | 5,786 lbm | (from the Langley SSTO(R) case) |
| Vehicle base diameter= | 48.40 ft | |
| Vehicle cone angle= | 5.50 deg | |
| Minimum gage factor= | 1.00 lbm/ft^2 | |
| | | |
| Maximum normal load case: Angle of attack= | 20.00 deg | |
| Maximum normal load case: Qbar= | 95.10 psf | |

| OMS and RCS systems mass coefficients: | |
|---|---|
| RCS system mass coefficient= | 0.000151 lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 sec |
| RCS thruster specific impulse (reentry)= | 410.00 sec |
| RCS thruster specific impulse (ascent)= | 350.00 sec |
| OMS system thrust-to-weight= | 0.04 g |
| OMS engine mass coefficient= | 0.035 lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 lbm/lbm |
| OMS thruster specific impulse= | 462.00 sec |

The main propulsion system additional delta v may be used for additional on-orbit maneuvers and to adjust the mission velocity requirements based on trajectory analysis results.
The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential throttling is selected for vehicle thrust vector control.

| | |
|---|---|
| RCS ascent mission velocity (applied to GLOW)= | 0 ft/sec |
| RCS on-orbit mission velocity (applied to MECO - residuals)= | 155 ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 80 ft/sec |
| OMS system on-orbit mission velocity (applied to MECO - residuals)= | 1,140 ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -71 ft/sec |
| Landing maneuver specific impulse= | 333.50 sec |
| Landing maneuver hover time= | 16.00 sec |
| Landing maneuver drag coefficient= | 0.90 |

| Nose definition: | |
|---|---|
| Exterior angle 1= | 17.00 deg |
| Exterior angle 2= | 30.00 deg |
| Ratio r2/r1= | 0.80 |
| Ratio r3/r1= | 0.20 |

| TPS unit Masses: | |
|---|---|
| Nose= | 2.20 lbm/ft^2 |
| Windward Fwd tank= | 0.65 lbm/ft^2 |
| Leeward Fwd tank= | 0.50 lbm/ft^2 |
| Windward Fwd Intertank= | 0.60 lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 lbm/ft^2 |
| Windward Mid tank= | 0.55 lbm/ft^2 |
| Leeward Mid tank= | 0.35 lbm/ft^2 |
| Windward Aft Intertank= | 0.50 lbm/ft^2 |
| Leeward Aft Intertank= | 0.35 lbm/ft^2 |
| Windward Aft tank= | 0.50 lbm/ft^2 |
| Leeward Aft tank= | 0.35 lbm/ft^2 |
| Windward Aft Skirt= | 0.65 lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 lbm/ft^2 |
| Body Flaps= | 1.50 lbm/ft^2 |

| Calculated Data: | | | | |
|---|---|---|---|---|
| Mission velocity (dV) (required) | 30,536 ft/sec | | | |

| Engine data: | Mode 1 | Mode 2 | Total | |
|---|---|---|---|---|
| Engine Thrust (single engine) (sl) | 440,835 | 0 | | lbf |
| Engine Thrust (single engine) (vac) | 509,042 | 220,460 | | lbf |
| Engine masses (single engine) | 5,318 | 5,476 | | lbm |
| Number of engines | 7 | 7 | | |
| Total sea level thrust | 3,085,844 | 0 | 3,085,844 | lbf |
| Total vacuum thrust | 3,563,293 | 1,543,223 | 3,563,293 | lbf |

| Parallel burn data (Burn flag= 2) | | |
|---|---|---|
| Thrust split (Fv2/Fvtotal) | 0 | |
| Iv1 | 385.10 | sec |

| Burn Two: | | |
|---|---|---|
| Stage velocity | 20,040 | ft/sec |
| Iv2 | 452.70 | sec |
| Mass ratio (r2) | 3.9586 | |
| Burn out weight (W2i) | 278,459 | lbm |
| Initial weight (W2o) | 1,102,302 | lbm |
| Propellant (Wf2) | 823,843 | lbm |
| Initial acceleration (No2) | 1.400 | g |

| Burn One: | | |
|---|---|---|
| Structure (Wst1) | 253,459 | lbm |
| Stage velocity (dV1) | 10,496 | ft/sec |
| Iv1 | 385.10 | sec |
| Mass ratio (r1) | 2.3329 | |
| Burn out weight (W1i) | 1,102,302 | lbm |
| Gross Loft Off Weight | 2,571,536 | lbm |
| Propellant (Wf1) | 1,469,234 | lbm |
| Initial accelerating (No1) | 1.200 | g |
| Mode 1 propellant | 1,469,234 | lbm |
| Mode 2 propellant | 0 | lbm |
| Mass flow rate | 9,253 | lbm/sec |
| Propellant density | 46.39 | lbm/ft^3 |

| Results: | | |
|---|---|---|
| Gross Lift Off Weight (GLOW) | 2,571,536 | lbf |
| Total Structural Mass (includes residuals) | 253,459 | lbf |
| Total Propellant Mass | 2,293,077 | lbf |
| Total mode 1 propellant | 1,469,234 | lbm |
| Total mode 2 propellant | 823,843 | lbm |
| Total oxidizer | 1,834,405 | lbm |
| Total fuel 1 | 296,785 | lbm |
| Total fuel 2 | 161,887 | lbm |
| Burn 1: | | |
| Initial throttle setting= | 100.00 | % |
| End of burn thrust to weight= | 3.233 | g |
| Initial propellant: | | |
| Oxidizer= | 1,128,372 | lbm |
| Fuel 1= | 296,785 | lbm |
| Fuel 2= | 44,077 | lbm |
| Burn 2: | | |
| Initial throttle setting= | 100.00 | % |
| End of burn thrust to weight= | 5.542 | g |
| Initial propellant: | | |
| Oxidizer= | 706,033 | lbm |
| Fuel 1= | 0 | lbm |
| Fuel 2= | 2,866 | lbm |

| If the vehicle has ascent RCS propellant, the vehicle GLOW and liftoff thrust must be modified to reflect the additional vehicle mass. | | |
|---|---|---|
| Vehicle GLOW= | 2,571,536 | lbm |
| Ascent RCS Propellant= | 0 | lbm |
| Revised Vehicle GLOW= | 2,571,536 | lbm |

This section calculates the velocity required to reach orbit.
O2/RP-1 on first stage.
O2/H2 on second stage.
Vacuum specific impulse used on both stages.
15x220 NM orbit, i= 28.5 deg, launch due east from KSC.
Mission delta v includes 1% FPR
Nominal data, No1= 1.565g, No2= 1.423g, delta v= 29261 fps.

For purposes of mission velocity requirement, stage 1 acceleration is constrained between 1.150 and 1.475 g.
For purposes of mission velocity requirement, stage 2 acceleration is constrained between 0.662 and 1.422 g.

| Stage 1 | | |
|---|---|---|
| No1 | delta delta v | N |
| 1.150 | 630 | 1 |
| 1.216 | 380 | 2 |
| 1.302 | 107 | 3 |
| 1.388 | -93 | 4 |
| 1.475 | -185 | 5 |
| 1.519 | -172 | 6 |
| 1.565 | 0 | 7 |

| Stage 2 | | |
|---|---|---|
| No2 | delta delta v | N |
| 0.662 | 1,609 | 1 |
| 0.696 | 1,394 | 2 |
| 0.730 | 1,198 | 3 |
| 0.798 | 918 | 4 |
| 0.867 | 688 | 5 |
| 1.005 | 398 | 6 |
| 1.143 | 201 | 7 |
| 1.238 | 80 | 8 |
| 1.423 | 0 | 9 |

Allowable stage initial accelerations:

| | Sizing value | Max Filter | Min Filter | Input Values | |
|---|---|---|---|---|---|
| No1= | 1.200 | 1.200 | 1.200 | 1.200 | g |
| No2= | 1.400 | 1.400 | 1.400 | 1.400 | g |

| | | |
|---|---|---|
| First stage throttle setting= | 1.000 | % |
| Second stage throttle setting= | 1.000 | % |
| First stage end of burn acceleration= | 3.233 | g |
| Second stage end of burn acceleration= | 5.542 | g |

Based on the results of 17 vehicles, an additional 200 ft/sec is added to the nominal delta v.
A single stage to orbit calibration run increased the mission velocity by another 450 ft/sec.

| | | |
|---|---|---|
| Nominal mission velocity= | 29,911 | ft/sec |
| ddv1= | 441 | ft/sec |
| ddv2= | 10 | ft/sec |
| Mission velocity= | 30,362 | ft/sec |

Experience has shown that a correction must be made for low first stage velocity.
Use additional velocity correction= 2000- .1905*dV1, do not allow this term to go below zero.

| | | |
|---|---|---|
| Nominal mission velocity= | 30,362 | ft/sec |
| First stage velocity, dV1= | 10,496 | ft/sec |
| Additional velocity correction= | 1 | ft/sec |
| Mission velocity= | 30,362 | ft/sec |

| The nominal orbit is inclination= 28.5 deg, perigee= 15 NM and apogee= 220 NM. Adjust for a different inclination | | |
|---|---|---|
| Mission inclination= | 51.60 | deg |
| Nominal mission velocity= | 30,362 | ft/sec |
| Velocity of the Earth's equatorial spin= | 1,526 | ft/sec |
| Velocity of the Earth's spin at i= 28.5= | 1,341 | ft/sec |
| Velocity of the Earth's spin at mission inc.= | 948 | ft/sec |
| Mission Velocity= | 30,755 | ft/sec |

| To adjust for a different perigee and apogee, at the nominal perigee (15 NM) do a burn to reach the mission apogee. At the nominal apogee, do a burn to bring the perigee up to the mission perigee. | | |
|---|---|---|
| Earth's radius= | 20,925,673 | ft |
| Earth's gravitational parameter= | 1.408E+16 | ft^3/sec^2 |
| Mission perigee= | 50.00 | NM |
| Mission apogee= | 100.00 | NM |
| | | |
| Nominal mission perigee= | 15.00 | NM |
| Nominal mission apogee= | 220.00 | NM |
| Nominal mission perigee= | 21,016,815 | ft |
| Nominal mission apogee= | 22,262,418 | ft |
| Nominal mission semi major axis= | 21,639,616 | ft |
| Nominal mission eccentricity= | 0.0287806 | |
| Nominal mission apogee velocity= | 24,781 | ft/sec |
| Nominal mission perigee velocity= | 26,250 | ft/sec |
| | | |
| Transfer orbit 1 perigee= | 15.00 | NM |
| Transfer orbit 1 apogee= | 100.00 | NM |
| Transfer orbit 1 perigee= | 21,016,815 | ft |
| Transfer orbit 1 apogee= | 21,533,284 | ft |
| Transfer orbit 1 semi major axis= | 21,275,050 | ft |
| Transfer orbit 1 eccentricity= | 0.0121379 | |
| Transfer orbit 1 apogee velocity= | 25,412 | ft/sec |
| Transfer orbit 1 perigee velocity= | 26,037 | ft/sec |
| | | |
| Transfer orbit 2 perigee= | 50.00 | NM |
| Transfer orbit 2 apogee= | 100.00 | NM |
| Transfer orbit 2 perigee= | 21,229,479 | ft |
| Transfer orbit 2 apogee= | 21,533,284 | ft |
| Transfer orbit 2 semi major axis= | 21,381,382 | ft |
| Transfer orbit 2 eccentricity= | 0.0071044 | |
| Transfer orbit 2 apogee velocity= | 25,477 | ft/sec |
| Transfer orbit 2 perigee velocity= | 25,841 | ft/sec |

| Results from change to a new orbit | |
|---|---|
| Nominal mission velocity= | 30,755 ft/sec |
| Perigee burn delta v= | -213 ft/sec |
| Apogee burn delta v= | 65 ft/sec |
| Mission velocity= | 30,607 ft/sec |

| Results from additional mission velocity | |
|---|---|
| Nominal mission velocity= | 30,607 ft/sec |
| Additional MPS mission velocity= | -71 ft/sec |
| Mission velocity= | 30,536 ft/sec |

### 3.1.6 Weights Spreadsheet

This spreadsheet is for sizing the vehicle (side entry cone).

| Input data: | | |
|---|---|---|
| Number of crew= | [redacted] | |
| Crew cabin volume= | [redacted] | ft^3 |
| Number of days on orbit= | [redacted] | |
| Average on orbit power usage= | [redacted] | kw |
| Average on orbit heat rejection requirement= | [redacted] | kw |
| Maximum acceleration (No) | [redacted] | g |
| Lift-off thrust to weight | [redacted] | g |
| Factor of safety | [redacted] | |
| Main propellant feed line and press sys= | [redacted] | lbm-sec/lbf |
| Vehicle mass prop contingency factor= | [redacted] | |
| Avionics= | [redacted] | lbm |
| Range safety= | [redacted] | lbm |
| Fwd intertank unit mass= | [redacted] | lbm/ft^2 |
| Aft intertank unit mass= | [redacted] | lbm/ft^2 |
| Aft skirt unit mass= | [redacted] | lbm/ft^2 |
| Crew Cabin Body Constant (Bo)= | [redacted] | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | [redacted] | lbm/lbm |
| Body insulation constant (Kbi)= | [redacted] | lbm/ft^2 (if hot structure selected) |
| Gimbal Actuator unit mass= | [redacted] | lbm/lbf |
| Thrust structure (Ktsf)= | [redacted] | lbm/lbf |
| Thrust structure (Ktsn)= | [redacted] | lbm/lbf |
| Prime Power (PWc) (aero surface)= | [redacted] | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | [redacted] | lbm/lbf |
| Prime Power (PWa) (avionics)= | [redacted] | lbm/lbm |
| Electrical Power Conv & Dist= | [redacted] | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface)= | [redacted] | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | [redacted] | lbm/lbf |
| Fuel cell unit mass (FCw)= | [redacted] | lbm/kw |
| Fuel cell reactant unit mass (FCc)= | [redacted] | lbm/kw-day |
| ECLSS crew cabin constant (Ec)= | [redacted] | lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | [redacted] | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | [redacted] | lbm/lbm |
| Active thermal control loop unit mass (Ew)= | [redacted] | lbm/kw |
| Personnel waste systems (PPf)= | [redacted] | lbm |
| Personnel seats and crew related (PPs)= | [redacted] | lbm/crew member |
| Personnel miscellaneous (Pm)= | [redacted] | lbm |
| Personnel weight (Pp)= | [redacted] | lbm/crew member |
| Body flap constant (Bbf)= | [redacted] | lbm/(ft^2)^1.15 |
| Surface control actuator constant (Ssc)= | [redacted] | lbm/ft^2 |
| Surface control miscellaneous (Spc)= | [redacted] | lbm |
| Payload bay diameter= | [redacted] | ft |
| Vehicle base diameter= | [redacted] | ft |
| Vehicle cone angle= | [redacted] | deg |

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Position= | [redacted] | [redacted] | [redacted] | |
| Ullage= | [redacted] | [redacted] | [redacted] | |
| Density= | [redacted] | [redacted] | [redacted] | lbm/ft^3 |
| Propellant mass= | [redacted] | [redacted] | [redacted] | lbm |
| Residual A (prop load)= | [redacted] | [redacted] | [redacted] | lbm/lbm |
| Residual B (max thrust)= | [redacted] | [redacted] | [redacted] | lbm/lbf |
| Ullage pressure= | [redacted] | [redacted] | [redacted] | psi |
| TPS unit mass= | [redacted] | [redacted] | [redacted] | lbm/ft^2 |
| Burn flag= | [redacted] | | | |
| Number of fuel tanks= | [redacted] | | | |
| Payload bay location flag= | [redacted] | | | |
| | **Fwd Tank** | **Mid Tank** | **Aft Tank** | |
| Forward endcap height coefficient= | [redacted] | [redacted] | [redacted] | |
| Aft endcap height coefficient= | [redacted] | [redacted] | [redacted] | |
| Tank design= | [redacted] | [redacted] | [redacted] | |
| Forward endcap flag= | [redacted] | [redacted] | [redacted] | |
| Aft endcap flag= | [redacted] | [redacted] | [redacted] | |
| **Propellant tank materials:** | **Fwd Tank** | **Mid Tank** | **Aft Tank** | |
| Density= | [redacted] | [redacted] | [redacted] | lbm/in^3 |
| Ftu= | [redacted] | [redacted] | [redacted] | psi |

| Calculated geometry: (tanks are conical) | Tank Radius (ft) | Endcap Radius (ft) | Height (ft) | (for elliptical endcaps) k | e |
|---|---|---|---|---|---|
| Base radius= | 24.20 | | | | |
| Aft skirt height= | 32.71 | | | | |
| Aft tank aft radius= | 21.05 | 21.05 | 14.88 | 1.414214 | 0.707107 |
| Aft tank barrel section height= | 9.32 | | | | |
| Aft tank fwd radius= | 20.15 | 20.15 | 14.25 | 1.414214 | 0.707107 |
| Aft intertank height= | 45.48 | | | | |
| Mid tank aft radius= | 15.77 | 15.77 | 11.15 | 1.414214 | 0.707107 |
| Mid tank barrel section height= | 7.27 | | | | |
| Mid tank fwd radius= | 15.07 | 15.07 | 10.66 | 1.414214 | 0.707107 |
| Fwd intertank height= | 1.00 | | | | |
| Fwd tank aft radius= | 14.98 | 14.98 | 10.59 | 1.414214 | 0.707107 |
| Fwd tank barrel section height= | 35.63 | | | | |
| Fwd tank fwd radius= | 11.55 | 11.55 | 8.16 | 1.414214 | 0.707107 |

| Tank | Forward | Middle | Aft | |
|---|---|---|---|---|
| Propellant mass= | 1,834,405 | 296,785 | 161,887 | lbm |
| Density= | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Ullage= | 0.05 | 0.05 | 0.05 | % |
| Volume= | 27,052 | 6,171 | 38,370 | ft^3 |
| Tank upper radius= | 11.55 | 15.07 | 20.15 | ft |
| Tank upper endcap radius= | 11.55 | 15.07 | 20.15 | ft |
| Tank upper endcap height= | 8.16 | 10.66 | 14.25 | ft |
| Upper Endcap volume= | 2,279 | 5,072 | 12,121 | ft^3 |
| Upper Endcap area= | 340 | 579 | 1,036 | ft^2 |
| Tank lower radius= | 14.98 | 15.77 | 21.05 | ft |
| Tank lower endcap radius= | 14.98 | 15.77 | 21.05 | ft |
| Tank lower endcap height= | 10.59 | 11.15 | 14.88 | ft |
| Lower Endcap volume= | 4,975 | 5,812 | 13,814 | ft^3 |
| Lower Endcap area= | 572 | 634 | 1,130 | ft^2 |
| Barrel section volume= | 19,798 | 5,431 | 12,435 | ft^3 |
| Barrel section length= | 35.63 | 7.27 | 9.32 | ft |
| Barrel section area 1= | 2,983 | 707 | 1,213 | ft^2 |
| Barrel section area 2= | 0 | 0 | 0 | ft^2 |
| Total area= | 3,895 | 1,921 | 3,378 | ft^2 |
| Cone half angle= | 5.50 | 5.50 | 5.50 | deg |
| Tank insulation= | 0.25 | 0 | 0.250 | lbm/ft^2 |

| Propellant tank mass data: | Forward | Middle | Aft | |
|---|---|---|---|---|
| Ullage (upper endcap) pressure= | 35.00 | 35.00 | 50.00 | psi |
| Lower endcap pressure= | 56.14 | 38.06 | 50.34 | psi |
| Average barrel section pressure= | 45.57 | 36.53 | 50.17 | psi |
| Upper endcap thickness= | 0.062 | 0.082 | 0.113 | in |
| Lower endcap thickness= | 0.130 | 0.093 | 0.119 | in |
| Barrel thickness 1= | 0.176 | 0.148 | 0.197 | in |
| Barrel thickness 2= | 0.000 | 0.000 | 0.000 | in |
| Endcap (pressurized structure) mass= | 1,349 | 1,497 | 2,063 | lbm |
| Barrel section (pressurized structure) mass= | 7,392 | 1,480 | 1,963 | lbm |
| Total mass (based on pressurized structure)= | 8,740 | 2,977 | 4,026 | lbm |
| Delta mass (semi empirical correction)= | 6,422 | 2,493 | 5,307 | lbm |
| Delta mass (density correction)= | 6,293 | 2,443 | 3,025 | lbm |
| Total mass = | 15,034 | 5,420 | 7,051 | lbm |
| Unit mass= | 0.5557 | 0.8783 | 0.1838 | lbm/ft^3 |

OMS/RCS system

| | | |
|---|---|---|
| RCS system= | [redacted] | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | [redacted] | sec |
| RCS thruster specific impulse (entry)= | [redacted] | sec |
| RCS thruster specific impulse (ascent)= | [redacted] | sec |
| RCS ascent mission velocity= | [redacted] | ft/sec |
| RCS on orbit mission velocity= | [redacted] | ft/sec |
| RCS entry mission velocity= | [redacted] | ft/sec |
| OMS system thrust to weight= | [redacted] | g |
| OMS engine constant= | [redacted] | lbm/lbf |
| OMS prop system weights= | [redacted] | lbm/lbm |
| OMS thruster specific impulse= | [redacted] | sec |
| OMS system on orbit mission velocity= | [redacted] | ft/sec |

| RCS Propellant: | RCS Ascent | OMS | RCS on orbit | RCS Entry | Total | |
|---|---|---|---|---|---|---|
| Mass Ratio= | 1.0000 | 1.0797 | 1.0115 | 1.0061 | | |
| Start Burn Mass= | 2,571,536 | 266,301 | 246,641 | 243,841 | | lbm |
| End Burn Mass= | 2,571,536 | 246,641 | 243,841 | 242,367 | | lbm |
| Propellant load= | 0 | 19,660 | 2,800 | 1,474 | 23,934 | lbm |

delta V for the landing maneuver:

| | | |
|---|---|---|
| Hover burn time= | [redacted] | sec |
| Drag Coefficient= | [redacted] | |
| Landing Weight= | 242,367 | lbm |
| Drag Area= | 1,840 | ft^2 |
| Terminal velocity= | 351 | ft/sec |

Assume the vehicle is deaccelerating at 3 g's, therefore 2 g's are used to come to a stop.

| | | |
|---|---|---|
| delta V to come to a sto | 526 | ft/sec |
| hover delta V= | 515 | ft/sec |
| Total delta V= | 1,041 | ft/sec |

Landing propellant is stored in the OMS/RCS tanks

| | | |
|---|---|---|
| Landing Isp= | [redacted] | sec |
| Landing delta V= | 1,041 | ft/sec |
| Landing Propellant= | 21,903 | lbm |

Engine Restart Propellant:

| | | |
|---|---|---|
| Maximum vacuum thrust= | 3,563,293 | lbf |
| Engine conditioning coefficient, fuel= | [redacted] | lbm/lbf |
| Engine conditioning coefficient, oxidizer= | [redacted] | lbm/lbf |
| Engine restart propellant= | 5,487 | lbm |

| | | |
|---|---|---|
| RCS System mass= | 5,709 | lbm |
| OMS Engine mass= | 373 | lbm |
| OMS System Mass= | 7,801 | lbm |
| Total= | 13,883 | lbm |

| Body flap planform area: | | |
|---|---|---|
| Vehicle base radius= | 24.20 | ft |
| Vehicle base area= | 1,840 | ft^2 |
| Body Flap Unit Area= | [redacted] | ft^2/ft^2 |
| Body Flap Planform Area= | 460 | ft^2 |

| Nose: (assume a bi conic nose cone with hemispherical tip)<br>(nose radius= cone 1 base radius= r1, cone 2 base radius= r2, tip radius= r3) | | |
|---|---|---|
| Exterior angle 1= | [redacted] | deg |
| Exterior angle 2= | [redacted] | deg |
| Ratio r2/r1= | [redacted] | |
| Ratio r3/r1= | [redacted] | |
| Aft radius (r1)= | 11.55 | ft |
| Cone 2 base radius (r2)= | 9.24 | ft |
| Nose tip radius (r3)= | 2.31 | ft |
| Cone 1 length (l1)= | 7.55 | ft |
| Cone 2 length (l2)= | 12.00 | ft |
| Total nose length= | 21.86 | ft |
| Lower cone (1) surface area= | 516 | ft^2 |
| Upper cone (2) surface area= | 503 | ft^2 |
| Tip (3) surface area= | 26 | ft^2 |
| Total nose surface area= | 1,044 | ft^2 |
| Unit weight= | [redacted] | lbm/ft |
| Nose weight= | 1,044 | lbm |

| TPS unit Masses: | | |
|---|---|---|
| Nose= | [redacted] | lbm/ft^2 |
| Windward Fwd tank= | [redacted] | lbm/ft^2 |
| Leeward Fwd tank= | [redacted] | lbm/ft^2 |
| Windward Fwd Intertank= | [redacted] | lbm/ft^2 |
| Leeward Fwd Intertank= | [redacted] | lbm/ft^2 |
| Windward Mid tank= | [redacted] | lbm/ft^2 |
| Leeward Mid tank= | [redacted] | lbm/ft^2 |
| Windward Aft Intertank= | [redacted] | lbm/ft^2 |
| Leeward Aft Intertank= | [redacted] | lbm/ft^2 |
| Windward Aft tank= | [redacted] | lbm/ft^2 |
| Leeward Aft tank= | [redacted] | lbm/ft^2 |
| Windward Aft Skirt= | [redacted] | lbm/ft^2 |
| Leeward Aft Skirt= | [redacted] | lbm/ft^2 |
| Body Flaps= | [redacted] | lbm/ft^2 |

| TPS Mass: | | |
|---|---|---|
| Nose= | 2,297 | lbm |
| Fwd tank= | 2,463 | lbm |
| Fwd intertank= | 47 | lbm |
| Mid tank= | 318 | lbm |
| Aft intertank= | 2,192 | lbm |
| Aft tank= | 515 | lbm |
| Aft skirt= | 3,142 | lbm |
| Total TPS= | 10,975 | lbm |

| Engine heat shield mass: | | |
|---|---|---|
| Base engine heat shield unit mass= | 1.64 | lbm/ft^2 |
| Vehicle base diameter= | 48.40 | ft |
| Vehicle base surface area= | 1,840 | ft^2 |
| Engine heat shield mass= | 3,017 | lbm |

| | | |
|---|---|---|
| Fwd intertank area= | 95 | ft^2 |
| Aft intertank area= | 5,157 | ft^2 |
| Aft skirt area= | 4,671 | ft^2 |

| Vehicle height: | | |
|---|---|---|
| Nose= | 21.86 | ft |
| Fwd tank= | 35.63 | ft |
| Fwd intertank= | 1.00 | ft |
| Mid tank= | 7.27 | ft |
| Aft intertank= | 45.48 | ft |
| Aft tank= | 9.32 | ft |
| Aft skirt= | 32.71 | ft (Aft skirt extends to cover the engine exit) |
| Total= | 153.28 | ft |

Propellant feed and pressurization system:
Included are the main propellant feed system, main tank pressurization system, and the purge and vent system.

| | | |
|---|---|---|
| Rpf= | [redacted] | lbm-sec/ft^3 |
| Mass flow rate= | [redacted] | lbm/sec |
| Density= | [redacted] | lbm/ft^3 |
| Feed/press sys= | 10,971 | lbm |

| Engine data: | Number | Total Fv (lbf) | Length (ft) | Unit mass (lbm) | Iv (sec) |
|---|---|---|---|---|---|
| Mode 1 engine= | [redacted] | | | | |
| Mode 2 engines= | [redacted] | | | | |
| Maximum vacuum thrust= | | [redacted] lbf | | | |
| Engine length= | [redacted] ft | | | | |

| Vehicle mass: | | | |
|---|---|---|---|
| Nose= | 1,044 | lbm | |
| Oxidizer Tank= | [illegible] | lbm | |
| Oxidizer Tank Insulation= | [illegible] | lbm | |
| Fwd Intertank= | [illegible] | lbm | |
| Aft Intertank= | [illegible] | lbm | |
| Fuel Tank 1= | 5,420 | lbm | |
| Fuel Tank 1 Insulation= | 0 | lbm | |
| Fuel Tank 2= | [illegible] | lbm | |
| Fuel Tank 2 Insulation= | [illegible] | lbm | |
| Aft skirt= | 6,396 | lbm | |
| Range Safety= | 0 | lbm | |
| Gimbal Actuators= | 0 | lbm | |
| Engines= | 37,224 | lbm | |
| Thrust Structure= | 15,714 | lbm | |
| Avionics= | 3,244 | lbm | |
| Prime Power= | 1,797 | lbm | |
| Power Conv & Dist= | 4,300 | lbm | |
| Feed/Press System= | 10,971 | lbm | |
| OMS/RCS System= | 13,883 | lbm | |
| Crew Cabin= | 0 | lbm | |
| Environmental Control= | 2,714 | lbm | |
| Landing Gear= | 7,524 | lbm | |
| Body Insulation= | 0 | lbm | |
| Body Flap= | 1,096 | lbm | (4 body flaps assumed) |
| Control Surface Actuators= | 1,400 | lbm | |
| Payload Bay= | 5,786 | lbm | |
| Engine Bay Heat Shield= | 3,017 | lbm | |
| TPS Tile Mass= | 10,975 | lbm | |
| Dry Mass Contingence= | 24,779 | lbm | |
| Dry Mass= | 189,976 | lbm | |
| On-orbit & Entry OMS/RCS Propellant= | 23,934 | lbm | |
| Residual Ascent Propellant= | 12,158 | lbm | |
| Landing Propellant= | 21,903 | lbm | |
| Engine Restart Propellant= | 5,487 | lbm | |
| Personnel Provisions= | 0 | lbm | |
| Personnel= | 0 | lbm | |
| Burnout mass (w/o payload)= | 253,459 | lbm | |
| Ascent RCS Propellant= | 0 | lbm | (burned during ascent) |
| Total Usable Ascent Propellant= | 2,293,077 | lbm | |
| Payload= | 25,000 | lbm | |
| Gross Lift Off Weight (GLOW)= | 2,571,536 | lbm | |
| Vehicle Mass at MECO= | 278,459 | lbm | |
| Landed vehicle Mass= | 214,976 | lbm | |

Summary vehicle dry weight
TPS tile and body insulation mass are spread over the vehicle's skin.
Gimbal actuators, engines, thrust structure, engine bay heat shield,
Feed/press, landing gear, body flap, surface controls are in the aft skirt.

Range safety, avionics, contingence, prime power,
Power conv & dist, OMS/RCS sys, crew cabin, environmental control, personnel provisions and personnel are in the same location as the payload bay.

Propellant tank and its insulation are listed as tank weight.

| | | |
|---|---|---|
| Nose= | 3,341 | lbm |
| Fwd tank= | 18,470 | lbm |
| Fwd intertank= | 203 | lbm |
| Mid tank= | 5,738 | lbm |
| Aft intertank= | 10,825 | lbm |
| Aft tank= | 8,411 | lbm |
| Aft skirt= | 142,988 | lbm |
| Total= | 189,976 | lbm |

### 3.2 Winged Body SSTO Vehicle Configuration Sizing Tool Test Case Files

This section contains the example input data files and printouts of the sizing tool spreadsheets for the Winged SSTO vehicle configuration sizing tool. The first input data file uses the Option 3 Access to Space Report version of an evolved SSME for the vehicle's main engines. This input data file was selected as an example of a bipropellant oxygen/hydrogen vehicle configuration. The second input data file uses the Option 3 Access to Space Report version of the Russian RD-701 engine. This input data file was selected as an example of a tripropellant oxygen/hydrogen/ kerosene vehicle configuration. The following vehicle configuration spreadsheets used the RD-701 input data file for the vehicle configuration definition.

#### 3.2.1 Option 3 Evolved SSME Input Data File

This is the input data file for the VTHL configuration with a round body and wings.
The engines used are rubber Option 3 Evolved SSMEs.
The payload bay is sideways in the mid intertank.
Tank position, 1 = aft, 2= middle, 3= forward
Tank design, 1= common (concave bkld), 2= nested (concave bkld), 3= separate
Common bulkhead means the aft propellant tank has only one endcap.
Nested bulkhead means a one foot distance between fwd and aft propellant tank endcaps.
Endcap flag, 1= ellipsoidal, 2= hemispherical, 3= toroidal endcaps
If endcap flag=1 height=dia/2*sqrt(2), =2 height=dia/2, =3 radius=.7*dia/2, height=.3*dia/2
If number of fuel tanks is set to 1, do not use the mid propellant tank or the Fuel 1 fuel tank.
Engine flag, 1= engines with a known thrust and weight are used, 2= engines with a known thrust to weight are used (engine thrust is allowed to vary).
Burn flag, 1= series burn, 2= parallel burn, 3= dual mode parallel burn
Payload bay location flag, 1= aft intertank, 2= forward intertank

| Input Data: | |
|---|---|
| Payload (Wpay) | 25,000 lbm |
| Number of crew | 0 |
| Crew cabin volume | 0 ft^3 |
| Number of days on-orbit | 7 |
| Average on-orbit power usage | 5 kw |
| Average on-orbit heat rejection requirement | 10 kw |
| Maximum acceleration (No) | 3.000 g |
| Maximum normal acceleration (Nz) | 2.500 g |
| Factor of safety | 1.40 |
| Orbit inclination | 51.60 deg |
| Orbit perigee | 50.00 NM |
| Orbit apogee | 100.00 NM |

| Tank Definition: | Ox tank | Fuel 1 tank | Fuel 2 tank |
|---|---|---|---|
| Position= | 3 | | 1 |
| Ullage= | 0.0500 | | 0.0500 |
| Density= | 71.20 | | 4.43 lbm/ft^3 |
| Residual A (propellant mass)= | 0.0038 | | 0.0016 lbm/lbm |
| Residual B (engine thrust)= | 0.0010 | | 0.0012 lbm/lbf |
| Ullage pressure= | 35.00 | | 50.00 psi |
| TPS unit mass= | 0.250 | | 0.250 lbm/ft^2 |
| | **Fwd Tank** | **Mid Tank** | **Aft Tank** |
| Forward endcap height coefficient= | 0.7071 | | 0.7071 |
| Aft endcap height coefficient= | 0.7071 | | 0.7071 |
| Tank design= | 3 | | 3 |
| Forward endcap flag= | 1 | | 1 |
| Aft endcap flag= | 1 | | 1 |

| Vehicle Materials: | Fwd Tank | Mid Tank | Aft Tank |
|---|---|---|---|
| Density= | 0.098 | | 0.057 lbm/ft^3 |
| Ftu= | 65,600 | | 90,400 psi |

| Mode 2 burn: | |
|---|---|
| Mode 2 mission velocity | 20,000 ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 sec |
| Iv2 | 447.30 sec |
| Mixture ratio (% Oxidizer) | 85.73 % |
| Mixture ratio (% Fuel 1) | 0.00 % |
| Mixture ratio (% Fuel 2) | 14.27 % |
| Engine height | 15.70 ft |
| Number of engines | 6 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine vac thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 71.06 lbf/lbm (vac) |
| No 2 (if engine flag= 2) | 1.400 g |

| Mode 1 burn: | |
|---|---|
| Isl1 | 390.40 sec |
| Iv1 | 447.30 sec |
| Mixture ratio (% Oxidizer) | 85.73 % |
| Mixture ratio (% Fuel 1) | % |
| Mixture ratio (% Fuel 2) | 14.27 % |
| Engine height | 15.70 ft |
| Number of engines | 6 |
| Engine flag | 2 |
| Engine mass (if engine flag= 1) | 0 lbm |
| Engine sl thrust (if engine flag= 1) | 0 lbf |
| Engine unit mass (if engine flag= 2) | 62.02 lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 g |

| | |
|---|---|
| Burn flag= | 3 |
| Number of fuels used= | 1 |
| Payload bay location flag= | 2 |

| Vehicle sizing coefficients: | |
|---|---|
| Main propellant feed line and press sys= | 55.00 lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 |
| Avionics= | 710 lbm/lbm^(1/8) |
| Range safety= | 0 lbm |
| Tip Fin Constant= | 1.00 lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 lbm/ft^2 (if hot structure is selected) |
| Base engine heat shield unit mass= | 1.64 lbm/ft^2 |
| Gimbal actuator unit mass= | 0.00075 lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 lbm/lbf |
| Prime Power (PWc) (aero surface)= | 0.274 lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 3.73E-05 lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface) | 0.000 lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 lbm/kw-day |
| | |
| ECLSS crew cabin constant (Ec)= | 0.00 lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200 lbm/kw |
| | |
| Personnel waste systems (PPf)= | 0.00 lbm |
| Personnel seats and crew related (PPs)= | 0.00 lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 lbm |
| Personnel mass (Pp)= | 0.00 lbm/crew member |
| | |
| Body flap constant (Bbf)= | 1.17 lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 lbm/ft^2 |
| Control surface miscellaneous hardware (Spc)= | 200 lbm |
| | |
| Payload bay mass= | 5,786 lbm (from Langley SSTO(R) RD-701 case) |
| Payload bay diameter= | 15.00 ft |
| Vehicle base diameter= | 35.50 ft |
| Vehicle cone angle= | 0.00 deg (vehicle is a cylinder) |

| OMS and RCS systems mass coefficients: | |
|---|---|
| RCS system mass coefficient= | 0.000151 lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 sec |
| RCS thruster specific impulse (reentry)= | 410.00 sec |
| RCS thruster specific impulse (ascent)= | 350.00 sec |
| | |
| OMS system thrust-to-weight= | 0.04 g |
| OMS engine mass coefficient= | 0.035 lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 lbm/lbm |
| OMS thruster specific impulse= | 462 sec |

The main propulsion system additional delta v may be used for additional on-orbit maneuvers and to adjust the mission velocity requirements based on trajectory analysis results.
The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential throttling is selected for vehicle thrust vector control.

| | |
|---|---|
| RCS ascent mission velocity (applied to GLOW)= | 0 ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1140 ft/sec |
| Main propulsion system additional dv (applied to MECO)= | 29 ft/sec |

| Vehicle layout: | |
|---|---|
| Theoretical Wing Loading= | 60 lbm/ft^2 |
| Wing Planform Ratio (Sexp/Sref)= | 0.54 |
| Ratio of Exposed Wing Wetted Area/Planform Area= | 2.064 |
| Cord Thickness Ratio (Height/Cord Length)= | 0.20 |
| Ratio Body Flap Width/Diameter= | 0.25 |
| Ratio Tip Fin/Wing Planform Area= | 0.17 |
| Wing Surface Area Distribution: | |
| Leading Edge= | 0.10 |
| Elevon= | 0.15 |
| Windward Side= | 0.35 |
| Leeward Side= | 0.40 |
| Wing/Body efficiency factor (f)= | 0.1500 |
| Wing Carry through Constant (Wc)= | 0.0267 |
| Exposed wing Material/Configuration Constant (Wm)= | 0.2140 |
| Exposed Wing Aspect Ratio= | 1.7800 |
| Exposed Wing Taper Ratio= | 0.2360 |
| Body Carry Through Ratio (Carry Through Width/Body Width)= | 0.8230 |

| Nose definition: | |
|---|---|
| Exterior angle 1= | 17 deg |
| Exterior angle 2= | 30 deg |
| Ratio r2/r1= | 0.8 |
| Ratio r3/r1= | 0.2 |
| Nose unit weight= | 0.25 lbm/ft^2 |

| TPS unit Masses: | |
|---|---|
| Nose= | 2.20 lbm/ft^2 |
| Windward Fwd tank= | 0.90 lbm/ft^2 |
| Leeward Fwd tank= | 0.40 lbm/ft^2 |
| Windward Fwd Intertank= | 0.90 lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 lbm/ft^2 |
| Windward Mid tank= | 0.90 lbm/ft^2 |
| Leeward Mid tank= | 0.40 lbm/ft^2 |
| Windward Aft Intertank= | 0.90 lbm/ft^2 |
| Leeward Aft Intertank= | 0.40 lbm/ft^2 |
| Windward Aft tank= | 0.90 lbm/ft^2 |
| Leeward Aft tank= | 0.40 lbm/ft^2 |
| Windward Aft Skirt= | 0.90 lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 lbm/ft^2 |
| Body Flaps= | 2.00 lbm/ft^2 |
| Tip Fin= | 2.00 lbm/ft^2 |
| Elevon= | 2.00 lbm/ft^2 |
| Wing Leading Edge= | 2.00 lbm/ft^2 |
| Wing Leeward Side= | 0.40 lbm/ft^2 |
| Wing Windward Side= | 1.30 lbm/ft^2 |

### 3.2.2 RD-701 Input Data File

| | | | | | | |
|---|---|---|---|---|---|---|
| This is the input data file for the VTHL configuration with a round body and wings. | | | | | | |
| | | | | | | |
| The engines used are rubber RD-701 engines. | | | | | | |
| | | | | | | |
| The payload bay is sideways in the forward intertank. | | | | | | |
| | | | | | | |
| Tank position, 1 = aft, 2= middle, 3= forward | | | | | | |
| Tank design, 1= common (concave bkld), 2= nested (concave bkld), 3= separate | | | | | | |
| Common bulkhead means the aft propellant tank has only one endcap. | | | | | | |
| Nested bulkhead means a one foot distance between fwd and aft propellant tank endcaps. | | | | | | |
| Endcap flag, 1= ellipsoidal, 2= hemispherical, 3= toroidal endcaps | | | | | | |
| If endcap flag=1 heigth=dia/2*sqrt(2), =2 heigth=dia/2, =3 radius=.7*dia/2, heigth=.3*dia/2 | | | | | | |
| If number of fuel tanks is set to 1, do not use the mid propellant tank or the Fuel 1 fuel tank. | | | | | | |
| | | | | | | |
| | | | | | | |
| Engine flag, 1= engines with an known thrust and weight are used, 2= engines with a known | | | | | | |
| thrust to weight are used (engine thrust is allowed to vary). | | | | | | |
| Burn flag, 1= series burn, 2= parallel burn, 3= dual mode parallel burn | | | | | | |
| | | | | | | |
| Payload bay location flag, 1= aft intertank, 2= forward intertank | | | | | | |

| Input Data: | | |
|---|---|---|
| Payload (Wpay) | 25,000 | lbm |
| Number of crew | 0 | |
| Crew cabin volume | 0 | ft^3 |
| Number of days on-orbit | 7 | |
| Average on-orbit power usage | 5 | kw |
| Average on-orbit heat rejection requirement | 10 | kw |
| Maximum acceleration (No) | 3.000 | g |
| Maximum normal acceleration (Nz) | 2.500 | g |
| Factor of safety | 1.40 | |
| Orbit inclination | 51.60 | deg |
| Orbit perigee | 50.00 | NM |
| Orbit apogee | 100.00 | NM |

| Tank definition: | | | | | | |
|---|---|---|---|---|---|---|
| | | | Ox tank | Fuel 1 tank | Fuel 2 tank | |
| Position= | | | 3 | 1 | 2 | |
| Ullage= | | | 0.0500 | 0.0500 | 0.0500 | |
| Density= | | | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Residual A (propellant mass)= | | | 0.0038 | 0.0038 | 0.0016 | lbm/lbm |
| Residual B (engine thrust)= | | | 0.0010 | 0.0010 | 0.0012 | lbm/lbf |
| Ullage pressure= | | | 35.00 | 35.00 | 50.00 | psi |
| TPS unit mass= | | | 0.250 | 0.000 | 0.250 | lbm/ft^2 |
| | | | | | | |
| | | | Fwd Tank | Mid Tank | Aft Tank | |
| Forward endcap height coefficient= | | | 0.7071 | 0.7071 | 0.3300 | |
| Aft endcap height coefficient= | | | 0.7071 | 0.7071 | 0.3300 | |
| Tank design= | | | 3 | 3 | 3 | |
| Forward endcap flag= | | | 1 | 1 | 1 | |
| Aft endcap flag= | | | 1 | 1 | 1 | |

| Vehicle Materials: | | | Fwd Tank | Mid Tank | Aft Tank | |
|---|---|---|---|---|---|---|
| Density= | | | 0.098 | 0.057 | 0.098 | lbm/ft^3 |
| Ftu= | | | 65,600 | 90,400 | 65,600 | psi |

| Mode 2 burn: | | | | |
|---|---|---|---|---|
| Mode 2 mission velocity | | | 20,130 | ft/sec |
| Isl2 (if Burn flag= 2) | | | 0.00 | sec |
| Iv2 | | | 452.70 | sec |
| Mixture ratio (% Oxidizer) | | | 85.70 | % |
| Mixture ratio (% Fuel 1) | | | 0.00 | % |
| Mixture ratio (% Fuel 2) | | | 14.30 | % |
| Engine height | | | 11.66 | ft |
| Number of engines | | | 6 | |
| Engine flag | | | 2 | |
| Engine mass (if engine flag= 1) | | | 0 | lbm |
| Engine vac thrust (if engine flag= 1) | | | 0 | lbf |
| Engine unit mass (if engine flag= 2) | | | 40.26 | lbf/lbm (vac) |
| No2 (if engine flag= 2) | | | 1.400 | g |

| Mode 1 burn: | | | | |
|---|---|---|---|---|
| Isl1 | | | 333.50 | sec |
| Iv1 | | | 385.10 | sec |
| Mixture ratio (% Oxidizer) | | | 76.80 | % |
| Mixture ratio (% Fuel 1) | | | 20.20 | % |
| Mixture ratio (% Fuel 2) | | | 3.00 | % |
| Engine height | | | 11.66 | ft |
| Number of engines | | | 6 | |
| Engine flag | | | 2 | |
| Engine mass (if engine flag= 1) | | | 0 | lbm |
| Engine sl thrust (if engine flag= 1) | | | 0 | lbf |
| Engine unit mass (if engine flag= 2) | | | 82.90 | lbf/lbm (sl) |
| No1 (if engine flag= 2) | | | 1.200 | g |

| | | | |
|---|---|---|---|
| Burn flag= | | | 3 |
| Number of fuels used= | | | 2 |
| Payload bay location flag= | | | 2 |

| | | |
|---|---|---|
| Vehicle sizing coefficients: | | |
| Main propellant feed line and press sys= | 55.00 | lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm/lbm^(1/8) |
| Range safety= | 0 | lbm |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 | lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 | lbm/ft^2 (if hot structure is selected) |
| | | |
| Base engine heat shield unit mass= | 1.64 | lbm/ft^2 |
| | | |
| Gimbal actuator unit mass= | 0 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimb | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| | | |
| ECLSS crew cabin constant (Ec)= | 0.00 | lbm/(ft^(1/3)^0.75) |
| ECLSS crew supplies constant (Eo)= | 0.00 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew) | 200 | lbm/kw |
| | | |
| Personnel waste systems (PPf)= | 0.00 | lbm |
| Personnel seats and crew related (PPs)= | 0.00 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| | | |
| Body flap constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Control surface miscellaneous hardware ( | 200 | lbm |
| | | |
| Payload bay mass= | 5,786 | lbm (from Langly SSTO(R) RD-701 case) |
| Payload bay diameter= | 15.00 | ft |
| Vehicle base diameter= | 29.80 | ft |
| Vehicle cone angle= | 0.00 | deg (vehicle is a cylinder) |

| | | |
|---|---|---|
| OMS and RCS systems mass coefficients: | | |
| RCS system mass coefficient= | 0.000151 | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 | sec |
| RCS thruster specific impulse (reentry)= | 410.00 | sec |
| RCS thruster specific impulse (ascent)= | 350.00 | sec |
| | | |
| OMS system thrust-to-weight= | 0.04 | g |
| OMS engine mass coefficient= | 0.035 | lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 | lbm/lbm |
| OMS thruster specific impulse= | 462 | sec |

| | | |
|---|---|---|
| The main propulsion system additional delta v may be used for additional on-orbit maneuvers | | |
| and to adjust the mission velocity requirements based on trajectory analysis results. | | |
| The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential | | |
| throttling is selected for vehicle thrust vector control. | | |
| RCS ascent mission velocity (applied to GLOW)= | 0 | ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 | ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 | ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1140 | ft/sec |
| Main propulsion system additional dv (applied to MECO)= | 29 | ft/sec |

| | | |
|---|---|---|
| Vehicle layout: | | |
| Theoretical Wing Loading= | 60 | lbm/ft^2 |
| Wing Planform Ratio (Sexp/Sref)= | 0.54 | |
| Ratio of Exposed Wing Wetted Area/Planform Area= | 2.064 | |
| Cord Thickness Ratio (Height/Cord Length)= | 0.20 | |
| Ratio Body Flap Width/Diameter= | 0.25 | |
| Ratio Tip Fin/Wing Planform Area= | 0.17 | |
| Wing Surface Area Distribution: | | |
| Leading Edge= | 0.10 | |
| Elevon= | 0.15 | |
| Windward Side= | 0.35 | |
| Leeward Side= | 0.40 | |
| | | |
| Wing/Body efficiency factor (f)= | 0.1500 | |
| Wing Carry through Constant (Wc)= | 0.0267 | |
| Exposed wing Material/Configuration Constant (Wm)= | 0.2140 | |
| Exposed Wing Aspect Ratio= | 1.7800 | |
| Exposed Wing Taper Ratio= | 0.2360 | |
| Body Carry Through Ratio (Carry Through Width/Body Width)= | 0.8230 | |

| | | |
|---|---|---|
| Nose definition: | | |
| Exterior angle 1= | 17 | deg |
| Exterior angle 2= | 30 | deg |
| Ratio r2/r1= | 0.8 | |
| Ratio r3/r1= | 0.2 | |
| Nose unit weight= | 0.25 | lbm/ft^2 |

| | | |
|---|---|---|
| TPS unit Masses: | | |
| Nose= | 2.20 | lbm/ft^2 |
| Windward Fwd tank= | 0.90 | lbm/ft^2 |
| Leeward Fwd tank= | 0.40 | lbm/ft^2 |
| Windward Fwd Intertank= | 0.90 | lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 | lbm/ft^2 |
| Windward Mid tank= | 0.90 | lbm/ft^2 |
| Leeward Mid tank= | 0.40 | lbm/ft^2 |
| Windward Aft Intertank= | 0.90 | lbm/ft^2 |
| Leeward Aft Intertank= | 0.40 | lbm/ft^2 |
| Windward Aft tank= | 0.90 | lbm/ft^2 |
| Leeward Aft tank= | 0.40 | lbm/ft^2 |
| Windward Aft Skirt= | 0.90 | lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 | lbm/ft^2 |
| Body Flaps= | 2.00 | lbm/ft^2 |
| Tip Fin= | 2.00 | lbm/ft^2 |
| Elevon= | 2.00 | lbm/ft^2 |
| Wing Leading Edge= | 2.00 | lbm/ft^2 |
| Wing Leeward Side= | 0.40 | lbm/ft^2 |
| Wing Windward Side= | 1.30 | lbm/ft^2 |

### 3.2.3 Performance Spreadsheet

| |
|---|
| This spread sheet is used to calculate the size of a dual burn single stage to orbit vehicle. |
| |
| This model is a VTHL configuration with a round body and wings. |
| The payload bay is sideways in the fwd intertank. |

| | | |
|---|---|---|
| Input Data: | | |
| Payload (Wpay) | 25,000 | lbm |
| Number of crew | 0 | |
| Crew cabin volume | 0 | ft^3 |
| Number of days on-orbit | 7 | |
| Average on-orbit power usage | 5 | kw |
| Average on-orbit heat rejection requirement | 10 | kw |
| Maximum acceleration (No) | 3.000 | g |
| Maximum normal acceleration (Nz) | 2.500 | g |
| Factor of safety | 1.40 | |
| Orbit inclination | 51.60 | deg |
| Orbit perigee | 50.00 | NM |
| Orbit apogee | 100.00 | NM |

| Tank definition: | | | | | | |
|---|---|---|---|---|---|---|
| | | | Ox tank | Fuel 1 tank | Fuel 2 tank | |
| Position= | | | 3 | 1 | 2 | |
| Ullage= | | | 0.0500 | 0.0500 | 0.0500 | |
| Density= | | | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Residual A (propellant mass)= | | | 0.0038 | 0.0038 | 0.0016 | lbm/lbm |
| Residual B (engine thrust)= | | | 0.0010 | 0.0010 | 0.0012 | lbm/lbf |
| Ullage pressure= | | | 35.00 | 35.00 | 50.00 | psi |
| TPS unit mass= | | | 0.250 | 0.000 | 0.250 | lbm/ft^2 |
| | | | | | | |
| | | | Fwd Tank | Mid Tank | Aft Tank | |
| Forward endcap height coefficient= | | | 0.7071 | 0.7071 | 0.3300 | |
| Aft endcap height coefficient= | | | 0.7071 | 0.7071 | 0.3300 | |
| Tank design= | | | 3 | 3 | 3 | |
| Forward endcap flag= | | | 1 | 1 | 1 | |
| Aft endcap flag= | | | 1 | 1 | 1 | |

| Vehicle Materials: | | | Fwd Tank | Mid Tank | Aft Tank | |
|---|---|---|---|---|---|---|
| Density= | | | 0.098 | 0.057 | 0.098 | lbm/ft^3 |
| Ftu= | | | 65,600 | 90,400 | 65,600 | psi |

| Mode 2 burn: | | | | |
|---|---|---|---|---|
| Mode 2 mission velocity | | | 20,130 | ft/sec |
| Isl2 (if Burn flag= 2) | | | 0.00 | sec |
| Iv2 | | | 452.70 | sec |
| Mixture ratio (% Oxidizer) | | | 85.7 | % |
| Mixture ratio (% Fuel 1) | | | 0 | % |
| Mixture ratio (% Fuel 2) | | | 14.3 | % |
| Engine height | | | 11.66 | ft |
| Number of engines | | | 6 | |
| Engine flag | | | 2 | |
| Engine mass (if engine flag= 1) | | | 0 | lbm |
| Engine vac thrust (if engine flag= 1) | | | 0 | lbf |
| Engine unit mass (if engine flag= 2) | | | 40.26 | lbf/lbm (vac) |
| No2 (if engine flag= 2) | | | 1.400 | g |

| Mode 1 burn: | | | | |
|---|---|---|---|---|
| Isl1 | | | 333.50 | sec |
| Iv1 | | | 385.10 | sec |
| Mixture ratio (% Oxidizer) | | | 76.80 | % |
| Mixture ratio (% Fuel 1) | | | 20.20 | % |
| Mixture ratio (% Fuel 2) | | | 3.00 | % |
| Engine height | | | 11.66 | ft |
| Number of engines | | | 6 | |
| Engine flag | | | 2 | |
| Engine mass (if engine flag= 1) | | | 0 | lbm |
| Engine sl thrust (if engine flag= 1) | | | 0 | lbf |
| Engine unit mass (if engine flag= 2) | | | 82.90 | lbf/lbm (sl) |
| No1 (if engine flag= 2) | | | 1.200 | g |

| | | | |
|---|---|---|---|
| Burn flag= | | | 3 |
| Number of fuels used= | | | 2 |
| Payload bay location flag= | | | 2 |

| | | |
|---|---|---|
| Vehicle sizing coefficients: | | |
| Main propellant feed line and press sys= | 55.00 | lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm/lbm^(1/8) |
| Range safety= | 0 | lbm |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 | lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 | lbm/ft^2 (if hot structure is selected) |
| | | |
| Base engine heat shield unit mass= | 1.64 | lbm/ft^2 |
| | | |
| Gimbal actuator unit mass= | 0 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimb | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| | | |
| ECLSS crew cabin constant (Ec)= | 0.00 | lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew) | 200 | lbm/kw |
| | | |
| Personnel waste systems (PPf)= | 0.00 | lbm |
| Personnel seats and crew related (PPs)= | 0.00 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| | | |
| Body flap constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Control surface miscellaneous hardware ( | 200 | lbm |
| | | |
| Payload bay mass= | 5,786 | lbm (from Langley SSTO(R) RD-701 case) |
| Payload bay diameter= | 15.00 | ft |
| Vehicle base diameter= | 29.80 | ft |
| Vehicle cone angle= | 0.00 | deg (vehicle is a cylinder) |

| | | |
|---|---|---|
| OMS and RCS systems mass coefficients: | | |
| RCS system mass coefficient= | 0.000151 | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | 422.00 | sec |
| RCS thruster specific impulse (reentry)= | 410.00 | sec |
| RCS thruster specific impulse (ascent)= | 350.00 | sec |
| | | |
| OMS system thrust-to-weight= | 0.04 | g |
| OMS engine mass coefficient= | 0.035 | lbm/lbf |
| OMS propellant system mass coefficient= | 0.152 | lbm/lbm |
| OMS thruster specific impulse= | 462 | sec |

| | | |
|---|---|---|
| The main propulsion system additional delta v may be used for additional on-orbit maneuvers | | |
| and to adjust the mission velocity requirements based on trajectory analysis results. | | |
| The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential | | |
| throttling is selected for vehicle thrust vector control. | | |
| RCS ascent mission velocity (applied to GLOW)= | 0 | ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 | ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 | ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1140 | ft/sec |
| Main propulsion system additional dv (applied to MECO)= | 29 | ft/sec |

| | | |
|---|---|---|
| Vehicle layout: | | |
| Theoretical Wing Loading= | 60 | lbm/ft^2 |
| Wing Planform Ratio (Sexp/Sref)= | 0.54 | |
| Ratio of Exposed Wing Wetted Area/Planform Area= | 2.064 | |
| Cord Thickness Ratio (Height/Cord Length)= | 0.20 | |
| Ratio Body Flap Width/Diameter= | 0.25 | |
| Ratio Tip Fin/Wing Planform Area= | 0.17 | |
| Wing Surface Area Distribution: | | |
| Leading Edge= | 0.10 | |
| Elevon= | 0.15 | |
| Windward Side= | 0.35 | |
| Leeward Side= | 0.40 | |
| | | |
| Wing/Body efficiency factor (f)= | 0.1500 | |
| Wing Carry through Constant (Wc)= | 0.0267 | |
| Exposed wing Material/Configuration Constant (Wm)= | 0.2140 | |
| Exposed Wing Aspect Ratio= | 1.7800 | |
| Exposed Wing Taper Ratio= | 0.2360 | |
| Body Carry Through Ratio (Carry Through Width/Body Width)= | 0.8230 | |

| | | |
|---|---|---|
| Nose definition: | | |
| Exterior angle 1= | 17 | deg |
| Exterior angle 2= | 30 | deg |
| Ratio r2/r1= | 0.8 | |
| Ratio r3/r1= | 0.2 | |
| Nose unit weight= | 0.25 | lbm/ft^2 |

| TPS unit Masses: | | |
|---|---|---|
| Nose= | 2.20 | lbm/ft^2 |
| Windward Fwd tank= | 0.90 | lbm/ft^2 |
| Leeward Fwd tank= | 0.40 | lbm/ft^2 |
| Windward Fwd Intertank= | 0.90 | lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 | lbm/ft^2 |
| Windward Mid tank= | 0.90 | lbm/ft^2 |
| Leeward Mid tank= | 0.40 | lbm/ft^2 |
| Windward Aft Intertank= | 0.90 | lbm/ft^2 |
| Leeward Aft Intertank= | 0.40 | lbm/ft^2 |
| Windward Aft tank= | 0.90 | lbm/ft^2 |
| Leeward Aft tank= | 0.40 | lbm/ft^2 |
| Windward Aft Skirt= | 0.90 | lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 | lbm/ft^2 |
| Body Flaps= | 2.00 | lbm/ft^2 |
| Tip Fin= | 2.00 | lbm/ft^2 |
| Elevon= | 2.00 | lbm/ft^2 |
| Wing Leading Edge= | 2.00 | lbm/ft^2 |
| Wing Leeward Side= | 0.40 | lbm/ft^2 |
| Wing Windward Side= | 1.30 | lbm/ft^2 |

| Calculated Data: | | |
|---|---|---|
| Mission velocity (dV) (required) | 30,635 | ft/sec |

| Engine data: | Mode 1 | Mode 2 | Total | |
|---|---|---|---|---|
| Engine Thrust (single engine) (sl) | 404,649 | 0 | | lbf |
| Engine Thrust (single engine) (vac) | 407,258 | 202,210 | | lbf |
| Engine masses (single engine) | 4,881 | 5,023 | | lbm |
| Number of engines | 6 | 6 | | |
| Total sea level thrust | 2,427,896 | 0 | 2,427,896 | lbf |
| Total vacuum thrust | 2,503,546 | 1,213,259 | 2,503,546 | lbf |

| Burn one vacuum Isp (Iv1) correction based on burn flag) | | |
|---|---|---|
| Thrust split (Fv2/Fvtotal) | 0 | |
| Iv1 | 385.10 | sec |

| Burn Two: | | |
|---|---|---|
| Stage velocity | 20,130 | ft/sec |
| Iv2 | 452.70 | sec |
| Mass ratio (r2) | 3.9831 | |
| Burn out weight (W2i) | 217,572 | lbm |
| Initial weight (W2o) | 866,614 | lbm |
| Propellant (Wf2) | 649,042 | lbm |
| Initial acceleration (No2) | 1.400 | g |

| Burn One: | | |
|---|---|---|
| Structure (Wst1) | 192,572 | lbm |
| Stage velocity (dV1) (required) | 10,505 | ft/sec |
| Iv1 | 385.10 | sec |
| Mass ratio (r1) (required) | 2.3347 | |
| Burn out weight (W1i) | 866,614 | lbm |
| Gross Loft Off Weight | 2,023,246 | lbm |
| Propellant (Wf1) | 1,156,633 | lbm |
| Initial accelerating (No1) | 1.200 | g |
| Mode 1 propellant | 1,156,633 | lbm |
| Mode 2 propellant | 0 | lbm |
| Mass flow rate | 7,250 | lbm/sec |
| Propellant density | 46.32 | lbm/ft^3 |

| Results: | | |
|---|---|---|
| Gross Lift Off Weight (GLOW) | 2,023,246 | lbf |
| Total Structural Mass (includes residuals) | 192,572 | lbf |
| Total Propellant Mass | 1,805,674 | lbf |
| Total mode 1 propellant | 1,156,633 | lbm |
| Total mode 2 propellant | 649,042 | lbm |
| Total oxidizer | 1,444,523 | lbm |
| Total fuel 1 | 233,640 | lbm |
| Total fuel 2 | 127,512 | lbm |
| | | |
| Burn 1: | | |
| Initial throttle setting= | 100.00 | % |
| End of burn thrust to weight= | 3.235 | g |
| Initial propellant: | | |
| Oxidizer= | 888,294 | lbm |
| Fuel 1= | 233,640 | lbm |
| Fuel 2= | 34,699 | lbm |
| | | |
| Burn 2: | | |
| Initial throttle setting= | 100.00 | % |
| End of burn thrust to weight= | 5.976 | g |
| Initial propellant: | | |
| Oxidizer= | 556,229 | lbm |
| Fuel 1= | 0 | lbm |
| Fuel 2= | 2,879 | lbm |

| If the vehicle has ascent RCS propellant, the vehicle GLOW and liftoff thrust must be modified to reflect the additional vehicle mass. | | |
|---|---|---|
| Vehicle GLOW= | 2,023,246 | lbm |
| Ascent RCS Propellant= | 0 | lbm |
| Revised Vehicle GLOW= | 2,023,246 | lbm |

| | | | | | | |
|---|---|---|---|---|---|---|
| This is section is used to calculate the velocity required to reach orbit, based on the relationship | | | | | | |
| between total dv requirements and initial burn T/W ratios. | | | | | | |
| dv parametrics were generated by off line trajectory executions. | | | | | | |
| | | | | | | |
| 15x220 NM orbit, i= 28.5 deg, launch due east from KSC. | | | | | | |
| Mission delta v includes 1% FPR | | | | | | |
| Nominal data, No1= 1.565g, No2= 1.423g, dv= 29,911 fps. | | | | | | |
| | | | | | | |
| For purposes of mission velocity requirement, stage 1 acceleration is constrained between 1.150 | | | | | | |
| and 1.475 g. | | | | | | |
| For purposes of mission velocity requirement, stage 2 acceleration is constrained between 0.662 | | | | | | |
| and 1.422 g. | | | | | | |

| | Burn 1 | |
|---|---|---|
| T/W | ddv | counter |
| 1.150 | 630 | 1 |
| 1.216 | 380 | 2 |
| 1.302 | 107 | 3 |
| 1.388 | -93 | 4 |
| 1.475 | -185 | 5 |
| 1.519 | -172 | 6 |
| 1.565 | 0 | 7 |

| | Burn 2 | |
|---|---|---|
| T/W | ddv | counter |
| 0.662 | 1,609 | 1 |
| 0.696 | 1,394 | 2 |
| 0.730 | 1,198 | 3 |
| 0.798 | 918 | 4 |
| 0.867 | 688 | 5 |
| 1.005 | 398 | 6 |
| 1.143 | 201 | 7 |
| 1.238 | 80 | 8 |
| 1.423 | 0 | 9 |

| | | | | | |
|---|---|---|---|---|---|
| Allowable stage initial accelerations: | | | | | |
| | Sizing value | Max Filter | Min Filter | Input Values | |
| No1= | 1.200 | 1.200 | 1.200 | 1.200 | g |
| No2= | 1.400 | 1.400 | 1.400 | 1.400 | g |
| | | | | | |
| Burn 1 throttle setting= | | | 1.000 | % | |
| Burn 2 throttle setting= | | | 1.000 | % | |
| | | | | | |
| Acceleration at the end of burn 1= | | | 3.236 | g | |
| Acceleration at the end of burn 2= | | | 5.576 | g | |

| | | | | | | |
|---|---|---|---|---|---|---|
| Mission velocity correction for the burn initial thrust to weight ratios used | | | | | | |
| Reference mission velocity= | | | 29,911 | ft/sec | | |
| ddv1= | | | 441 | ft/sec | | |
| ddv2= | | | 10 | ft/sec | | |
| Corrected mission velocity= | | | 30,362 | ft/sec | | |

| | | | | | | |
|---|---|---|---|---|---|---|
| Correction to total dv requirement for low burn 1 dv. | | | | | | |
| Additional velocity correction= 2000- .1905*dV1 (do not allow this term to go below zero). | | | | | | |
| Reference mission velocity= | | | 30,362 | ft/sec | | |
| First stage velocity, dV1= | | | 10,505 | ft/sec | | |
| Additional velocity correction= | | | -1 | ft/sec | | |
| Corrected mission velocity= | | | 30,362 | ft/sec | | |

| | | | |
|---|---|---|---|
| The nominal orbit is inclination= 28.5 deg, perigee= 15 NM and apogee= 220 NM. | | | |
| Adjust for a different inclination | | | |
| Mission inclination= | 51.60 | deg | |
| Reference mission velocity= | 30,362 | ft/sec | |
| Velocity of the Earth's equatorial spin= | 1,526 | ft/sec | |
| Velocity of the Earth's spin at i= 28.5= | 1,341 | ft/sec | |
| Vel of the Earth's spin at mission inc.= | 948 | ft/sec | |
| Corrected mission velocity= | 30,755 | ft/sec | |

| | | | |
|---|---|---|---|
| To adjust for a different perigee and apogee, at the reference perigee (15 NM) do a burn to reach the mission apogee. At the nominal apogee, do a burn to bring the perigee up to the mission perigee. | | | |
| Earths radius= | 20,925,673 | ft | |
| Earths gravitational parameter | 1.408E+16 | ft^3/sec^2 | |
| Mission perigee= | 50.00 | NM | |
| Mission apogee= | 100.00 | NM | |
| | | | |
| Nominal mission perigee= | 15.00 | NM | |
| Nominal mission apogee= | 220.00 | NM | |
| Nominal mission perigee= | 21,016,815 | ft | |
| Nominal mission apogee= | 22,262,418 | ft | |
| Nominal mission semi major axis= | 21,639,616 | ft | |
| Nominal mission eccentricity= | 0.0287808 | | |
| Nominal mission apogee velocity= | 24,781 | ft/sec | |
| Nominal mission perigee velocity= | 26,250 | ft/sec | |
| | | | |
| Transfer orbit 1 perigee= | 15.00 | NM | |
| Transfer orbit 1 apogee= | 100.00 | NM | |
| Transfer orbit 1 perigee= | 21,016,815 | ft | |
| Transfer orbit 1 apogee= | 21,533,284 | ft | |
| Transfer orbit 1 semi major axis= | 21,275,050 | ft | |
| Transfer orbit 1 eccentricity= | 0.0121379 | | |
| Transfer orbit 1 apogee velocity= | 25,412 | ft/sec | |
| Transfer orbit 1 perigee velocity= | 26,037 | ft/sec | |
| | | | |
| Transfer orbit 2 perigee= | 50.00 | NM | |
| Transfer orbit 2 apogee= | 100.00 | NM | |
| Transfer orbit 2 perigee= | 21,229,479 | ft | |
| Transfer orbit 2 apogee= | 21,533,284 | ft | |
| Transfer orbit 2 semi major axis= | 21,381,382 | ft | |
| Transfer orbit 2 eccentricity= | 0.0071044 | | |
| Transfer orbit 2 apogee velocity= | 25,477 | ft/sec | |
| Transfer orbit 2 perigee velocity= | 25,843 | ft/sec | |

| | | |
|---|---|---|
| Results from change to a new orbit: | | |
| Reference mission velocity= | 30,755 | ft/sec |
| Perigee burn delta v= | -213 | ft/sec |
| Apogee burn delta v= | 65 | ft/sec |
| Corrected mission velocity= | 30,607 | ft/sec |

| | | | |
|---|---|---|---|
| Results from additional mission velocity: | | | |
| Reference mission velocity= | | 30,607 | ft/sec |
| Additional MPS mission velocity= | | 29 | ft/sec |
| Final mission velocity= | | 30,636 | ft/sec |

### 3.2.4 Weights Spreadsheet

This spreadsheet is for sizing the vehicle (winged rocket VTHL configuration).

| Input data: | | |
|---|---|---|
| Number of crew= | 0 | |
| Crew cabin volume= | 0 | ft^3 |
| Number of days on orbit= | 7 | |
| Average on orbit power usage= | 5 | kw |
| Average on orbit heat rejection requirement= | 10 | kw |
| Maximum axial acceleration (No) | 3.00 | g |
| Maximum normal acceleration= | 2.50 | g |
| Lift-off thrust to weight | 1.2 | g |
| Factor of safety | 1.40 | |
| Main propellant feed line and press sys= | 55 | lbm-sec/lbf |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm |
| Range safety= | 0 | lbm |
| Vehicle body constant= | 1.32 | lbm/(ft^2*g^(1/3)) |
| Crew Cabin Body Constant (Bo)= | 0 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0 | lbm/ft^2 (if hot structure selected) |
| | | |
| Gimbal Actuator unit mass= | 0 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface) | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| | | |
| ECLSS crew cabin constant (Ec)= | 0 | lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200 | lbm/kw |
| | | |
| Personnel waste systems (PPf)= | 0 | lbm |
| Personnel seats and crew related (PPs)= | 0 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| | | |
| Surface control actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Surface control miscellaneous (Spc)= | 200 | lbm |
| | | |
| Payload bay diameter= | 15.00 | ft |
| Vehicle base diameter= | 29.8 | ft |
| Vehicle cone angle= | 0 | deg (vehicle is a cylinder) |

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Position= | 3 | 1 | 2 | |
| Ullage= | 0.05 | 0.05 | 0.05 | |
| Density= | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Propellant mass= | 1,444,523 | 233,540 | 127,512 | lbm |
| Residual A (propellant mass)= | 0.0038 | 0.0038 | 0.0016 | lbm/lbm |
| Residual B (engine thrust)= | 0.0010 | 0.0010 | 0.0012 | lbm/lbf |
| Ullage pressure= | 35.00 | 35.00 | 60.00 | psi |
| TPS unit mass= | 0.25 | 0 | 0.25 | lbm/ft^2 |
| Burn flag= | 3 | | | |
| Number of fuel tanks= | 2 | | | |
| Payload bay location flag= | 2 | | | |
| | | | | |
| | Fwd Tank | Mid Tank | Aft Tank | |
| Forward endcap height coefficient= | 0.7071 | 0.7071 | 0.3300 | |
| Aft endcap height coefficient= | 0.7071 | 0.7071 | 0.3300 | |
| Tank design= | 3 | 3 | 3 | |
| Forward endcap flag= | 1 | 1 | 1 | |
| Aft endcap flag= | 1 | 1 | 1 | |
| | | | | |
| Propellant tank materials: | Fwd Tank | Mid Tank | Aft Tank | |
| Density= | 0.098 | 0.057 | 0.098 | lbm/in^3 |
| Ftu= | 65,600 | 90,400 | 65,600 | psi |

| | Tank | Endcap | | (for elliptical endcaps) | |
|---|---|---|---|---|---|
| Calculated geometry: | Radius (ft) | Radius (ft) | Height (ft) | k | e |
| Base radius= | 14.90 | | | | |
| Aft skirt height= | 5.88 | | | | |
| Aft tank aft radius= | 14.90 | 14.90 | 4.92 | 3.030303 | 0.943981 |
| Aft tank barrel section height= | 0.41 | | | | |
| Aft tank fwd radius= | 14.90 | 14.90 | 4.92 | 3.030303 | 0.943981 |
| Aft intertank height= | 18.54 | | | | |
| Mid tank aft radius= | 14.90 | 14.90 | 10.54 | 1.414214 | 0.707107 |
| Mid tank barrel section height= | 29.28 | | | | |
| Mid tank fwd radius= | 14.90 | 14.90 | 10.54 | 1.414214 | 0.707107 |
| Fwd intertank height= | 40.29 | | | | |
| Fwd tank aft radius= | 14.90 | 14.90 | 10.54 | 1.414214 | 0.707107 |
| Fwd tank barrel section height= | 16.50 | | | | |
| Fwd tank fwd radius= | 14.90 | 14.90 | 10.54 | 1.414214 | 0.707107 |

| Propellant tank geometry: | Forward | Middle | Aft | |
|---|---|---|---|---|
| Propellant mass= | 1,444,523 | 127,512 | 233,640 | lbm |
| Density= | 71.20 | 4.43 | 50.60 | lbm/ft^3 |
| Ullage= | 0.05 | 0.05 | 0.05 | % |
| Volume= | 21,303 | 30,223 | 4,858 | ft^3 |
| Tank upper radius= | 14.90 | 14.90 | 14.90 | ft |
| Tank upper endcap radius= | 14.90 | 14.90 | 14.90 | ft |
| Tank upper endcap height= | 10.54 | 10.54 | 4.92 | ft |
| Upper Endcap volume= | 4,899 | 4,899 | 2,286 | ft^3 |
| Upper Endcap area= | 568 | 568 | 420 | ft^2 |
| Tank lower radius= | 14.90 | 14.90 | 14.90 | ft |
| Tank lower endcap radius= | 14.90 | 14.90 | 14.90 | ft |
| Tank lower endcap height= | 10.54 | 10.54 | 4.92 | ft |
| Lower Endcap volume= | 4,899 | 4,899 | 2,286 | ft^3 |
| Lower Endcap area= | 568 | 568 | 420 | ft^2 |
| Barrel section volume= | 11,505 | 20,425 | 286 | ft^3 |
| Barrel section length= | 16.50 | 29.28 | 0.41 | ft |
| Barrel section area 1= | 1,544 | 2,742 | 38 | ft^2 |
| Barrel section area 2= | 0.00 | 0.00 | 0.00 | ft^2 |
| Total area= | 2,676 | 3,874 | 878 | ft^2 |
| Cone half angle= | 0.00 | 0.00 | 0.00 | deg |
| Tank insulation= | 0.25 | 0.25 | 0.000 | lbm/ft^2 |

| Propellant tank mass data: | Forward | Middle | Aft | |
|---|---|---|---|---|
| Ullage (upper endcap) pressure= | 35.00 | 50.00 | 35.00 | psi |
| Lower endcap pressure= | 44.79 | 51.08 | 35.17 | psi |
| Average barrel section pressure= | 39.89 | 50.54 | 35.09 | psi |
| Upper endcap thickness= | 0.081 | 0.115 | 0.135 | in |
| Lower endcap thickness= | 0.103 | 0.085 | 0.135 | in |
| Barrel thickness 1= | 0.152 | 0.140 | 0.134 | in |
| Barrel thickness 2= | 0.000 | 0.000 | 0.000 | in |
| Endcap (pressurized structure) mass= | 1,468 | 932 | 1,599 | lbm |
| Barrel section (pressurized structure) mass= | 3,317 | 3,149 | 72 | lbm |
| Total mass (based on pressurized structure)= | 4,785 | 4,081 | 1,672 | lbm |
| Delta mass (semi empirical correction)= | 3,886 | 6,375 | 874 | lbm |
| Delta mass (density correction)= | 3,808 | 3,634 | 857 | lbm |
| Total mass = | 8,594 | 7,715 | 2,528 | lbm |
| Unit mass= | 0.4034 | 0.2653 | 0.5205 | lbm/ft^3 |

| Unpressurized Structures: | | |
|---|---|---|
| Fwd intertank unit mass= | 2.004 | lbm/ft^2 |
| Aft intertank unit mass= | 2.004 | lbm/ft^2 |
| Aft skirt unit mass= | 2.004 | lbm/ft^2 |
| | | |
| Fwd intertank area= | 3,772 | ft^2 |
| Aft intertank area= | 1,736 | ft^2 |
| Aft skirt area= | 644 | ft^2 |

| Engine data: | Number | Total Fv(lbf) | Length (ft) | Unit mass | Iv |
|---|---|---|---|---|---|
| | | (lbf) | (ft) | (lbm) | (sec) |
| Mode 1 engine= | 6 | 2,803,546 | 11.66 | 851 | 385.10 |
| Mode 2 engines= | 6 | 1,213,250 | 11.66 | 5,023 | 452.70 |
| Maximum Vacuum thrust= | | 2,803,546 | lbf | | |

| OMS/RCS system | | | | | | |
|---|---|---|---|---|---|---|
| RCS system= | | | 0.000151 | lbm/lbm-ft | | |
| RCS thruster specific impulse (on-orbit)= | | | 422.00 | sec | | |
| RCS thruster specific impulse (entry)= | | | 410.00 | sec | | |
| RCS thruster specific impulse (ascent)= | | | 350.00 | sec | | |
| RCS ascent mission velocity= | | | 0.00 | ft/sec | | |
| RCS on orbit mission velocity= | | | 155.00 | ft/sec | | |
| RCS entry mission velocity= | | | 40.00 | ft/sec | | |
| | | | | | | |
| OMS system thrust to weight= | | | 0.04 | g | | |
| OMS engine constant= | | | 0.035 | lbm/lbf | | |
| OMS prop system weights= | | | 0.152 | lbm/lbm | | |
| OMS thruster specific impulse= | | | 462.00 | sec | | |
| OMS system on orbit mission velocity= | | | 140 | ft/sec | | |
| | | | | | | |
| RCS Propellant: | RCS Ascent | OMS | RCS on orbit | RCS Entry | Total | |
| Mass Ratio= | 1.0000 | 1.0797 | 1.0115 | 1.0030 | | |
| Start Burn Mass= | 2,023,246 | 208,000 | 192,644 | 190,458 | | lbm |
| End Burn Mass= | 2,023,246 | 192,644 | 190,458 | 189,881 | | lbm |
| Propellant load= | 0 | 15,356 | 2,187 | 577 | 18,119 | lbm |
| | | | | | | |
| RCS System mass= | 4,365 | lbm | | | | |
| OMS Engine mass= | 291 | lbm | | | | |
| OMS System Mass= | 2,754 | lbm | | | | |
| Total= | 7,410 | lbm | | | | |

| | | |
|---|---|---|
| Wings: | | |
| Theoretical Wing Loading= | 80 | lbm/ft^2 |
| Wing Planform Ratio (Sexp/Sref)= | 0.54 | |
| Ratio of Exposed Wing Wetted Area/Planform Area= | 2.064 | |
| Chord Thickness Ratio (Height/Cord Length)= | 0.20 | |
| Ratio Tip Fin/Wing Planform Area= | 0.17 | |
| | | |
| Wing Surface Area Distribution: | | |
| Leading Edge= | 0.10 | |
| Elevon= | 0.15 | |
| Windward Side= | 0.35 | |
| Leeward Side= | 0.40 | |
| Wing/Body efficiency factor (f)= | 0.1500 | |
| Wing Carry through Constant (Wc)= | 0.0267 | |
| Exposed wing Material/Configuration Constant (Wm)= | 0.2140 | |
| Exposed Wing Aspect Ratio= | 1.7800 | |
| Exposed Wing Taper Ratio= | 0.2360 | |
| Body Carry Through Ratio (Carry Through Width/Body Width)= | 0.8290 | |
| | | |
| Wing Geometry: | | |
| Theoretical Wing Planform Area= | 3,211 | ft^2 |
| Exposed wing Planform Area (Sw)= | 1,734 | ft^2 |
| Body planform area (Sb)= | 4,523 | ft^2 |
| Exposed Total Structural Wing Span (Lw)= | 55.55 | ft |
| Exposed wing Root Chord= | 50.50 | ft |
| Exposed Wing Tip Width= | 11.92 | ft |
| Exposed wing root chord max thickness (Tr)= | 10.10 | ft |
| Body width at wing body junction (Lb)= | 24.53 | ft |
| | | |
| Exposed Wing Wetted Area= | 3,579 | ft^2 |
| Leading Edge Wetted Area= | 358 | ft^2 |
| Elevon Wetted Area= | 537 | ft^2 |
| Wing Windward Side Wetted Area= | 1,252 | ft^2 |
| Wing Leeward Side Wetted Area= | 1,431 | ft^2 |
| | | |
| Wing Mass= | 6,783 | lbm |

| | | |
|---|---|---|
| Tip Fins: | | |
| Ratio of Tip Fin wetted area to wing wetted area= | 0.17 | |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) |
| Tip Fin planform area= | 295 | ft*2 |
| Tip Fin Mass= | 977 | lbm |

| Nose: (assume a biconic nose cone with hemispherical tip) | | |
|---|---|---|
| (nose radius= cone 1 base radius= r1, cone 2 base radius= r2, tip radius= r3) | | |
| Exterior angle 1= | 17.00 | deg |
| Exterior angle 2= | 30.00 | deg |
| Ratio r2/r1= | 0.80 | |
| Ratio r3/r1= | 0.20 | |
| Aft radius (r1)= | 14.90 | ft |
| Cone 2 base radius (r2)= | 11.92 | ft |
| Nose tip radius (r3)= | 2.98 | ft |
| Cone 1 length (l1)= | 9.75 | ft |
| Cone 2 length (l2)= | 15.48 | ft |
| Total nose length= | 28.21 | ft |
| Lower cone (1) surface area= | 859 | ft^2 |
| Upper cone (2) surface area= | 837 | ft^2 |
| Tip (3) surface area= | 56 | ft^2 |
| Total nose surface area= | 1,791 | ft^2 |
| Unit weight= | 0.25 | lbm/ft |
| Nose weight= | 448 | lbm |

| TPS unit Masses: | | |
|---|---|---|
| Nose= | 2.20 | lbm/ft^2 |
| Windward Fwd tank= | 0.90 | lbm/ft^2 |
| Leeward Fwd tank= | 0.40 | lbm/ft^2 |
| Windward Fwd Intertank= | 0.90 | lbm/ft^2 |
| Leeward Fwd Intertank= | 0.40 | lbm/ft^2 |
| Windward Mid tank= | 0.90 | lbm/ft^2 |
| Leeward Mid tank= | 0.40 | lbm/ft^2 |
| Windward Aft Intertank= | 0.90 | lbm/ft^2 |
| Leeward Aft Intertank= | 0.40 | lbm/ft^2 |
| Windward Aft tank= | 0.90 | lbm/ft^2 |
| Leeward Aft tank= | 0.40 | lbm/ft^2 |
| Windward Aft Skirt= | 0.90 | lbm/ft^2 |
| Leeward Aft Skirt= | 0.40 | lbm/ft^2 |
| Body Flaps= | 2.00 | lbm/ft^2 |
| Tip Fin= | 2.00 | lbm/ft^2 |
| Elevon= | 2.00 | lbm/ft^2 |
| Wing Leading Edge= | 2.00 | lbm/ft^2 |
| Wing Leeward Side= | 0.40 | lbm/ft^2 |
| Wing Windward Side= | 1.30 | lbm/ft^2 |

| Body Flap: (length is larger of engine length or vehicle control requirement) | | |
|---|---|---|
| Ratio Body Flap Width/Diameter= | 0.25 | |
| Body flap constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 |
| Body Flap Planform Area= | 347 | ft^2 |
| Body Flap Mass= | 978 | lbm |

| | | | |
|---|---|---|---|
| TPS Mass: | | | |
| Nose= | | 3,853 | lbm |
| Fwd tank mass= | | 1,004 | lbm |
| Fwd intertank mass= | | 2,452 | lbm |
| Mid tank mass= | | 1,782 | lbm |
| Aft intertank mass= | | 1,128 | lbm |
| Aft tank mass= | | 25 | lbm |
| Aft skirt mass= | | 1,114 | lbm |
| Body Flap= | | 695 | lbm |
| Wing mass= | | 3,990 | lbm |
| Tip Fins mass= | | 1,179 | lbm |
| Total TPS mass= | | 17,221 | lbm |

| | | | |
|---|---|---|---|
| Engine heat shield mass: | | | |
| Base engine heat shield unit mass= | | 1.64 | lbm/ft^2 |
| Vehicle base diameter= | | 29.80 | ft |
| Vehicle base surface area= | | 697 | ft^2 |
| Engine heat shield mass= | | 1,144 | lbm |

| | | | | | | |
|---|---|---|---|---|---|---|
| Vehicle height: | | | | | | |
| Nose= | | 26.21 | ft | | | |
| Fwd tank= | | 18.50 | ft | | | |
| Fwd intertank= | | 40.29 | ft (zero if number of tanks is set to 1) | | | |
| Mid tank= | | 29.28 | ft | | | |
| Aft intertank= | | 18.54 | ft | | | |
| Aft tank= | | 0.41 | ft | | | |
| Aft skirt= | | 6.88 | ft | | | |
| Engine= | | 11.65 | ft | | | |
| Total= | | 151.77 | ft | | | |

| | | | | | | |
|---|---|---|---|---|---|---|
| Propellant feed and pressurization system: | | | | | | |
| Included are the main propellant feed system, main tank pressurization system, | | | | | | |
| and the purge and vent system. | | | | | | |
| Rpf= | | | | 55.00 | lbm-sec/ft^3 | |
| Mass flow rate= | | | | 7,280 | lbm/sec | |
| Density= | | | | 46.39 | lbm/ft^3 | |
| Feed/press sys= | | | | 8,632 | lbm | |

| Vehicle mass: | | |
|---|---|---|
| Nose= | 438 | lbm |
| Oxidizer Tank= | 8,594 | lbm |
| Oxidizer Tank Insulation= | 669 | lbm |
| Fwd Intertank= | 7,559 | lbm |
| Aft Intertank= | 3,479 | lbm |
| Fuel Tank 1= | 2,528 | lbm |
| Fuel Tank 1 Insulation= | 0 | lbm |
| Fuel Tank 2= | 7,715 | lbm |
| Fuel Tank 2 Insulation= | 968 | lbm |
| Aft skirt= | 1,292 | lbm |
| Range Safety= | 0 | lbm |
| Gimbal Actuators= | 0 | lbm |
| Engines= | 29,287 | lbm |
| Thrust Structure= | 11,270 | lbm |
| Avionics= | 3,187 | lbm |
| Prime Power= | 1,985 | lbm |
| Power Conv & Dist= | 3,798 | lbm |
| Feed/Press System= | 8,632 | lbm |
| OMS/RCS System= | 7,410 | lbm |
| Crew Cabin= | 0 | lbm |
| Environmental Control= | 2,701 | lbm |
| Landing Gear= | 5,696 | lbm |
| Body Insulation= | 0 | lbm |
| Wings= | 6,783 | lbm |
| Tip Fins= | 977 | lbm |
| Body Flap= | 978 | lbm |
| Control Surface Actuators= | 3,277 | lbm |
| Payload bay= | 5,786 | lbm |
| Engine Bay Heat Shield= | 1,144 | lbm |
| TPS Tile Mass= | 17,221 | lbm |
| Dry Mass Contingency= | 21,506 | lbm |
| Dry Mass= | 164,881 | lbm |
| | | |
| On-orbit & Entry OMS/RCS Propellant= | 18,119 | lbm |
| Residual Ascent Propellant= | 9,571 | lbm |
| Personnel Provisions= | 0 | lbm |
| Personnel= | 0 | lbm |
| Burnout Mass (w/o payload)= | 192,572 | lbm |
| | | |
| Ascent RCS Propellant= | 0 | lbm |
| Total Usable Ascent Propellant= | 1,805,674 | lbm |
| Payload= | 25,000 | lbm |
| Gross Lift Off Weight (GLOW)= | 2,023,246 | lbm |
| Main Engine Cut Off (MECO)= | 217,572 | lbm |
| Landed Vehicle Mass= | 189,881 | lbm |

### 3.3 Lifting Body SSTO Vehicle Configuration Sizing Tool Test Case Files

This section contains the example input data files and printouts of the sizing tool spreadsheets for the lifting body SSTO vehicle configuration sizing tool. The first input data file uses the Option 3 Access to Space Report version of an evolved SSME for the vehicle's main engines. This input data file was selected as an example of a bipropellant oxygen/hydrogen vehicle configuration. The second input data file uses the Option 3 Access to Space Report version of the Russian RD-701 engine. This input data file was selected as an example of a tripropellant oxygen/hydrogen/kerosene vehicle configuration. The following vehicle configuration spreadsheets used the RD-701 input data file for the vehicle configuration definition.

#### 3.3.1 Option 3 Evolved SSME Input Data File

| |
|---|
| This is the input data file for the lifting body VTHL configuration. |
| The engine used is an evolved SSME used in the Access to Space Option 3 final report. |
| |
| If number of fuel tanks is set to 1, do not use fuel tank 1. |
| |
| Engine flag, 1= engines with an known thrust and weight are used, 2= engines with a known thrust to weight are used. |
| Burn flag, 1= series burn, 2= parallel burn, 3= dual mode parallel burn |

| | | |
|---|---|---|
| Input Data: | | |
| Payload (Wpay) | 25,000 | lbm |
| Number of crew | 0 | |
| Crew cabin volume | 0 | ft^3 |
| Number of days on-orbit | 7 | |
| Average on-orbit power usage | 5 | kw |
| Average on-orbit heat rejection requirement | 10 | kw |
| Maximum acceleration (No) | 3.000 | g |
| Maximum normal acceleration (Nz) | 1.600 | g |
| Factor of safety | 1.40 | |
| Orbit inclination | 51.60 | deg |
| Orbit perigee | 50.00 | NM |
| Orbit apogee | 100.00 | NM |

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Ullage= | 0.05 | | 0.05 | |
| Density= | 71.20 | | 4.43 | lbm/ft^3 |
| Residual A (propellant mass)= | 0.0038 | | 0.0016 | lbm/lbm |
| Residual B (engine thrust)= | 0.001 | | 0.0012 | lbm/lbf |
| Ullage pressure= | 20.00 | | 20.00 | psi |
| TPS unit mass= | 0.250 | | 0.250 | lbm/ft^2 |

| Vehicle Materials: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Density= | 0.098 | | 0.057 | lbm/ft^3 |
| Ftu= | 65,600 | | 90,400 | psi |

| | | |
|---|---|---|
| Mode 2 burn: | | |
| Mode 2 mission velocity | 19,800 | ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 | sec |
| Iv2 | 447.30 | sec |
| Mixture ratio (% Oxidizer) | 85.73 | % |
| Mixture ratio (% Fuel 1) | 0.00 | % |
| Mixture ratio (% Fuel 2) | 14.27 | % |
| Engine height | 18.06 | ft |
| Number of engines | 5 | |
| Engine flag | 2 | |
| Engine mass (if engine flag= 1) | 0 | lbm |
| Engine vac thrust (if engine flag= 1) | 0 | lbf |
| Engine unit mass (if engine flag= 2) | 71.06 | lbf/lbm (vac) |
| No2 (if engine flag= 2) | 1.400 | g |

| | | |
|---|---|---|
| Mode 1 burn: | | |
| Isl1 | 390.40 | sec |
| Iv1 | 447.30 | sec |
| Mixture ratio (% Oxidizer) | 85.73 | % |
| Mixture ratio (% Fuel 1) | 0.00 | % |
| Mixture ratio (% Fuel 2) | 14.27 | % |
| Engine height | 18.03 | ft |
| Number of engines | 5 | |
| Engine flag | 2 | |
| Engine mass (if engine flag= 1) | 0 | lbm |
| Engine sl thrust (if engine flag= 1) | 0 | lbf |
| Engine unit mass (if engine flag= 2) | 62.02 | lbf/lbm (sl) |
| No1 (if engine flag= 2) | 1.200 | g |

| | |
|---|---|
| Burn flag= | 3 |
| Number of fuels used= | 1 |

| | | |
|---|---|---|
| Vehicle sizing coefficients: | | |
| Main propellant feed line and press sys= | 55.00 | lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm/lbm^(1/8) |
| Range safety= | 0.00 | lbm |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 | lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 | lbm/ft^2 (if hot structure is selected) |
| | | |
| Base engine heat shield unit mass= | 1.64 | lbm/ft^2 |
| | | |
| Gimbal actuator unit mass= | 0.00129 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 3.73E-05 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface) | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| | | |
| ECLSS crew cabin constant (Ec)= | 0.00 | lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 | lbm/kw |
| | | |
| Personnel waste systems (PPf)= | 0.00 | lbm |
| Personnel seats and crew related (PPs)= | 0.00 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| | | |
| Control surface constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Control surface miscellaneous hardware (Spc)= | 200 | lbm |
| | | |
| Payload bay mass= | 3,925 | lbm(from Langley SSTO(R)RD-701 case) |

| | | | |
|---|---|---|---|
| OMS and RCS systems mass coefficients: | | | |
| RCS system mass coefficient= | | 0.000151 | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | | 422.00 | sec |
| RCS thruster specific impulse (reentry)= | | 410.00 | sec |
| RCS thruster specific impulse (ascent)= | | 350.00 | sec |
| | | | |
| OMS system thrust-to-weight= | | 0.04 | g |
| OMS engine mass coefficient= | | 0.035 | lbm/lbf |
| OMS propellant system mass coefficient= | | 0.152 | lbm/lbm |
| OMS thruster specific impulse= | | 462 | sec |

| | | |
|---|---|---|
| The main propulsion system additional delta v may be used for additional on-orbit maneuvers | | |
| and to adjust the mission velocity requirements based on trajectory analysis results. | | |
| The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential | | |
| throttling is selected for vehicle thrust vector control. | | |
| RCS ascent mission velocity (applied to GLOW)= | 0 | ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 | ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 | ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1140 | ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -280 | ft/sec |

| | | |
|---|---|---|
| Vehicle layout: | | |
| Nose length= | 5.00 | ft |
| Oxidizer tank fwd radius= | 16.76 | ft |
| Oxidizer tank aft radius= | 16.76 | ft |
| Fuel tank half angle= | 6.00 | deg |
| Payload bay diameter= | 15.00 | ft |
| Payload bay length= | 30.00 | ft |
| Payload bay/Oxidizer tank standoff= | 5.00 | ft |
| Engine bay height= | 26.30 | ft |
| Oxidizer tank/engine standoff distance= | 10.00 | ft |
| | | |
| Crew cabin length= | 12.50 | ft |
| Crew cabin fwd width= | 17.50 | ft |
| Crew cabin aft width= | 17.50 | ft |
| Crew Cabin/payload bay standoff= | 10.00 | ft |
| | | |
| Oxidizer/Fuel 2 tank standoff= | 0.50 | ft |
| Aeroshell standoff= | 0.50 | ft |

| | | |
|---|---|---|
| Body distribution: (fraction of horizontal tip fin planform area) | | |
| Horizontal tip fin outboard angle= | 11.970 | deg |
| Tip fins= | 1.534 | |
| Elevon= | 0.480 | |
| Rudder= | 0.431 | |

| | | | | | |
|---|---|---|---|---|---|
| | | Body | | TPS | |
| Body Element: | | Coefficient | | Coefficient | |
| Nose | | 2.587 | ft^2/ft^2 | 2.20 | lbm/ft^2 |
| | | | | | |
| Forward Section | | | | | |
| Glove | | 0.710 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| Leeward Surface | | 0.670 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.631 | ft^2/ft^2 | 1.30 | lbm/ft^2 |
| | | | | | |
| Forward Fuel Tank Cone Section | | | | | |
| Glove | | 0.902 | ft^2/ft^2 | 0.65 | lbm/ft^2 |
| Leeward Surface | | 0.706 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.655 | ft^2/ft^2 | 1.00 | lbm/ft^2 |
| | | | | | |
| Aft Fuel Tank Cone Section | | | | | |
| Glove | | 0.883 | ft^2/ft^2 | 0.60 | lbm/ft^2 |
| Leeward Surface | | 0.713 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.655 | ft^2/ft^2 | 0.90 | lbm/ft^2 |
| | | | | | |
| Thrust Structure Section | | | | | |
| Glove | | 0.879 | ft^2/ft^2 | 0.55 | lbm/ft^2 |
| Leeward Surface | | 0.720 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.687 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| | | | | | |
| Body flaps= | | | | 2.00 | lbm/ft^2 |
| Tip fin leading edge= | | | | 2.00 | lbm/ft^2 |
| Tip fin windward= | | | | 0.80 | lbm/ft^2 |
| Tip fin leeward= | | | | 0.65 | lbm/ft^2 |
| Elevons= | | | | 2.00 | lbm/ft^2 |
| Rudder= | | | | 2.00 | lbm/ft^2 |

| | | |
|---|---|---|
| Tip fin surface area breakdown: | | |
| Tip fin surface area/tip fin planform area= | 2.06 | ft^2/ft^2 |
| Tip fin leading edge area/tip fin surface area= | 0.10 | ft^2/ft^2 |
| Tip fin windward surface area/tip fin surface area= | 0.42 | ft^2/ft^2 |
| Tip fin leeward surface area/tip fin surface area= | 0.48 | ft^2/ft^2 |

### 3.3.2 RD-701 Input Data File

| |
|---|
| This is the input data file for the lifting body VTHL configuration. |
| The engine used is a rubber RD-701. |
| |
| If number of fuel tanks is set to 1, do not use fuel tank 1. |
| |
| |
| Engine flag, 1= engines with an known thrust and weight are used, 2= engines with a known |
| thrust to weight are used. |
| Burn flag, 1= series burn, 2= parallel burn, 3= dual mode parallel burn |

| Input Data: | | |
|---|---|---|
| Payload (Wpay) | 25,000 | lbm |
| Number of crew | 0 | |
| Crew cabin volume | 0 | ft^3 |
| Number of days on-orbit | 7 | |
| Average on-orbit power usage | 5 | kw |
| Average on-orbit heat rejection requirement | 10 | kw |
| Maximum acceleration (No) | 3.000 | g |
| Maximum normal acceleration (Nz) | 1.600 | g |
| Factor of safety | 1.40 | |
| Orbit inclination | 51.60 | deg |
| Orbit perigee | 50.00 | NM |
| Orbit apogee | 100.00 | NM |

| Tank definition: | | | | |
|---|---|---|---|---|
| | Ox tank | Fuel 1 tank | Fuel 2 tank | |
| Ullage= | 0.05 | 0.05 | 0.05 | |
| Density= | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Residual A (propellant mass)= | 0.0038 | 0.0038 | 0.0016 | lbm/lbm |
| Residual B (engine thrust)= | 0.001 | 0.001 | 0.0012 | lbm/lbf |
| Ullage pressure= | 20.00 | 20.00 | 20.00 | psi |
| TPS unit mass= | 0.250 | 0.000 | 0.250 | lbm/ft^2 |

| Vehicle Materials: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Density= | 0.098 | 0.098 | 0.057 | lbm/ft^3 |
| Ftu= | 65,600 | 65,600 | 90,400 | psi |

| Mode 2 burn: | | |
|---|---|---|
| Mode 2 mission velocity | 19,830 | ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 | sec |
| Iv2 | 452.70 | sec |
| Mixture ratio (% Oxidizer) | 85.70 | % |
| Mixture ratio (% Fuel 1) | 0.00 | % |
| Mixture ratio (% Fuel 2) | 14.30 | % |
| Engine height | 13.26 | ft |
| Number of engines | 5 | |
| Engine flag | 2 | |
| Engine mass (if engine flag= 1) | 0 | lbm |
| Engine vac thrust (if engine flag= 1) | 0 | lbf |
| Engine unit mass (if engine flag= 2) | 40.26 | lbf/lbm (vac) |
| No2 (if engine flag= 2) | 1.400 | g |

| | | | | |
|---|---|---|---|---|
| Mode 1 burn: | | | | |
| Isl1 | | | 333.50 | sec |
| Iv1 | | | 385.10 | sec |
| Mixture ratio (% Oxidizer) | | | 76.80 | % |
| Mixture ratio (% Fuel 1) | | | 20.20 | % |
| Mixture ratio (% Fuel 2) | | | 3.00 | % |
| Engine height | | | 13.26 | ft |
| Number of engines | | | 5 | |
| Engine flag | | | 2 | |
| Engine mass (if engine flag= 1) | | | 0 | lbm |
| Engine sl thrust (if engine flag= 1) | | | 0 | lbf |
| Engine unit mass (if engine flag= 2) | | | 82.9 | lbf/lbm (sl) |
| No1 (if engine flag= 2) | | | 1.200 | g |

| | | | |
|---|---|---|---|
| Burn flag= | | | 3 |
| Number of fuels used= | | | 2 |

| | | |
|---|---|---|
| Vehicle sizing coefficients: | | |
| Main propellant feed line and press sys= | 55.00 | lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm/lbm^(1/8) |
| Range safety= | 0.00 | lbm |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 | lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 | lbm/ft^2 (if hot structure is selected) |
| | | |
| Base engine heat shield unit mass= | 1.64 | lbm/ft^2 |
| | | |
| Gimbal actuator unit mass= | 0 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface) | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| | | |
| ECLSS crew cabin constant (Ec)= | 0.00 | lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0.00 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 | lbm/kw |
| | | |
| Personnel waste systems (PPf)= | 0.00 | lbm |
| Personnel seats and crew related (PPs)= | 0.00 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| | | |
| Control surface constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Control surface miscellaneous hardware (Spc)= | 200 | lbm |
| | | |
| Payload bay mass= | 3,925 | lbm(from Langley SSTO(R)RD-701 case) |

| | | | |
|---|---|---|---|
| OMS and RCS systems mass coefficients: | | | |
| RCS system mass coefficient= | | 0.000151 | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | | 422.00 | sec |
| RCS thruster specific impulse (reentry)= | | 410.00 | sec |
| RCS thruster specific impulse (ascent)= | | 350.00 | sec |
| | | | |
| OMS system thrust-to-weight= | | 0.04 | g |
| OMS engine mass coefficient= | | 0.035 | lbm/lbf |
| OMS propellant system mass coefficient= | | 0.152 | lbm/lbm |
| OMS thruster specific impulse= | | 462 | sec |

| | | |
|---|---|---|
| The main propulsion system additional delta v may be used for additional on-orbit maneuvers | | |
| and to adjust the mission velocity requirements based on trajectory analysis results. | | |
| The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential | | |
| throttling is selected for vehicle thrust vector control. | | |
| RCS ascent mission velocity (applied to GLOW)= | 0 | ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 | ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 | ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1140 | ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -280 | ft/sec |

| | | |
|---|---|---|
| Vehicle layout: | | |
| Nose length= | 5.00 | ft |
| Oxidizer tank fwd radius= | 13.57 | ft |
| Oxidizer tank aft radius= | 13.57 | ft |
| Fuel tank half angle= | 3.95 | deg |
| Payload bay diameter= | 15.00 | ft |
| Payload bay length= | 30.00 | ft |
| Payload bay/Oxidizer tank standoff= | 5.00 | ft |
| Engine bay height= | 17.75 | ft |
| Oxidizer tank/engine standoff distance= | 10.00 | ft |
| | | |
| Crew cabin length= | 12.50 | ft |
| Crew cabin fwd width= | 10.00 | ft |
| Crew cabin aft width= | 17.00 | ft |
| Crew Cabin/payload bay standoff= | 10.00 | ft |
| | | |
| Oxidizer/Fuel 2 tank standoff= | 0.50 | ft |
| Aeroshell standoff= | 0.50 | ft |

| | | |
|---|---|---|
| Body distribution: (fraction of horizontal tip fin planform area) | | |
| Horizontal tip fin outboard angle= | 11.970 | deg |
| Tip fins= | 1.534 | |
| Elevon= | 0.480 | |
| Rudder= | 0.431 | |

| | | Body | | TPS | |
|---|---|---|---|---|---|
| Body Element: | | Coefficient | | Coefficient | |
| Nose | | 2.587 | ft^2/ft^2 | 2.20 | lbm/ft^2 |
| | | | | | |
| Forward Section | | | | | |
| Glove | | 0.710 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| Leeward Surface | | 0.670 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.631 | ft^2/ft^2 | 1.30 | lbm/ft^2 |
| | | | | | |
| Forward Fuel Tank Cone Section | | | | | |
| Glove | | 0.902 | ft^2/ft^2 | 0.65 | lbm/ft^2 |
| Leeward Surface | | 0.706 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.655 | ft^2/ft^2 | 1.00 | lbm/ft^2 |
| | | | | | |
| Aft Fuel Tank Cone Section | | | | | |
| Glove | | 0.883 | ft^2/ft^2 | 0.60 | lbm/ft^2 |
| Leeward Surface | | 0.713 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.655 | ft^2/ft^2 | 0.90 | lbm/ft^2 |
| | | | | | |
| Thrust Structure Section | | | | | |
| Glove | | 0.879 | ft^2/ft^2 | 0.55 | lbm/ft^2 |
| Leeward Surface | | 0.720 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.687 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| | | | | | |
| Body flaps= | | | | 2.00 | lbm/ft^2 |
| Tip fin leading edge= | | | | 2.00 | lbm/ft^2 |
| Tip fin windward= | | | | 0.80 | lbm/ft^2 |
| Tip fin leeward= | | | | 0.65 | lbm/ft^2 |
| Elevons= | | | | 2.00 | lbm/ft^2 |
| Rudder= | | | | 2.00 | lbm/ft^2 |

| | | |
|---|---|---|
| Tip fin surface area breakdown: | | |
| Tip fin surface area/tip fin planform area= | 2.06 | ft^2/ft^2 |
| Tip fin leading edge area/tip fin surface area= | 0.10 | ft^2/ft^2 |
| Tip fin windward surface area/tip fin surface area= | 0.42 | ft^2/ft^2 |
| Tip fin leeward surface area/tip fin surface area= | 0.48 | ft^2/ft^2 |

### 3.3.3 Performance Spreadsheet

| |
|---|
| This spread sheet is used to calculate the size of a dual burn single stage to orbit vehicle. |
| To find the best velocity split between two modes, vary the mode two velocity. |
| |
| This model is a lifting body VTHL configuration. |

| Input Data: | | |
|---|---|---|
| Payload (Wpay) | 25,000 | lbm |
| Number of crew | 0 | |
| Crew cabin volume | 0 | ft^3 |
| Number of days on-orbit | 7 | |
| Average on-orbit power usage | 5 | kw |
| Average on-orbit heat rejection requirement | 10 | kw |
| Maximum acceleration (No) | 3.000 | g |
| Maximum normal acceleration (Nz) | 1.600 | g |
| Factor of safety | 1.40 | |
| Orbit inclination | 51.60 | deg |
| Orbit perigee | 50.00 | NM |
| Orbit apogee | 100.00 | NM |

| Tank definition: | | | | |
|---|---|---|---|---|
| | Ox tank | Fuel 1 tank | Fuel 2 tank | |
| Ullage= | 0.05 | 0.05 | 0.05 | |
| Density= | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Residual A (propellant mass)= | 0.0038 | 0.0038 | 0.0016 | lbm/lbm |
| Residual B (engine thrust)= | 0.001 | 0.001 | 0.0012 | lbm/lbf |
| Ullage pressure= | 20.00 | 20.00 | 20.00 | psi |
| TPS unit mass= | 0.250 | 0.000 | 0.250 | lbm/ft^2 |

| Vehicle Materials: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Density= | 0.098 | 0.098 | 0.057 | lbm/ft^3 |
| Ftu= | 65,600 | 65,600 | 90,400 | psi |

| Mode 2 burn: | | |
|---|---|---|
| Mode 2 mission velocity | 19,830 | ft/sec |
| Isl2 (if Burn flag= 2) | 0.00 | sec |
| Iv2 | 452.70 | sec |
| Mixture ratio (% Oxidizer) | 85.70 | % |
| Mixture ratio (% Fuel 1) | 0.00 | % |
| Mixture ratio (% Fuel 2) | 14.30 | % |
| Engine height | 13.26 | ft |
| Number of engines | 5 | |
| Engine flag | 2 | |
| Engine mass (if engine flag= 1) | 0 | lbm |
| Engine vac thrust (if engine flag= 1) | 0 | lbf |
| Engine unit mass (if engine flag= 2) | 40.26 | lbf/lbm (vac) |
| No2 (if engine flag= 2) | 1.400 | g |

| | | | | | |
|---|---|---|---|---|---|
| Mode 1 burn: | | | | | |
| Isl1 | | | | 333.50 | sec |
| Iv1 | | | | 385.10 | sec |
| Mixture ratio (% Oxidizer) | | | | 76.80 | % |
| Mixture ratio (% Fuel 1) | | | | 20.20 | % |
| Mixture ratio (% Fuel 2) | | | | 3.00 | % |
| Engine height | | | | 13.26 | ft |
| Number of engines | | | | 5 | |
| Engine flag | | | | 2 | |
| Engine mass (if engine flag= 1) | | | | 0 | lbm |
| Engine sl thrust (if engine flag= 1) | | | | 0 | lbf |
| Engine unit mass (if engine flag= 2) | | | | 82.9 | lbf/lbm (sl) |
| No1 (if engine flag= 2) | | | | 1.200 | g |

| | | | |
|---|---|---|---|
| Burn flag= | | | 3 |
| Number of fuels used= | | | 2 |

| | | |
|---|---|---|
| Vehicle sizing coefficients: | | |
| Main propellant feed line and press sys= | 55.00 | lbm-sec/ft^3 |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm/lbm^(1/8) |
| Range safety= | 0.00 | lbm |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) |
| Body Constant= | 1.32 | lbm/ft-g^(1/3) |
| Crew Cabin Body Constant (Bo)= | 0.00 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0.00 | lbm/ft^2 (if hot structure is selected) |
| | | |
| Base engine heat shield unit mass= | 1.64 | lbm/ft^2 |
| | | |
| Gimbal actuator unit mass= | 0 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface) | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| | | |
| ECLSS crew cabin constant (Ec)= | 0.00 | lbm/(ft^(1/3)^0.75) |
| ECLSS crew supplies constant (Eo)= | 0.00 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 | lbm/kw |
| | | |
| Personnel waste systems (PPf)= | 0.00 | lbm |
| Personnel seats and crew related (PPs)= | 0.00 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0.00 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| | | |
| Control surface constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 |
| Control surface actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Control surface miscellaneous hardware (Spc)= | 200 | lbm |
| | | |
| Payload bay mass= | 3,925 | lbm(from Langley SSTO(R)RD-701 case) |

| | | | |
|---|---|---|---|
| OMS and RCS systems mass coefficients: | | | |
| RCS system mass coefficient= | | 0.000151 | lbm/lbm-ft |
| RCS thruster specific impulse (on-orbit)= | | 422.00 | sec |
| RCS thruster specific impulse (reentry)= | | 410.00 | sec |
| RCS thruster specific impulse (ascent)= | | 350.00 | sec |
| | | | |
| OMS system thrust-to-weight= | | 0.04 | g |
| OMS engine mass coefficient= | | 0.035 | lbm/lbf |
| OMS propellant system mass coefficient= | | 0.152 | lbm/lbm |
| OMS thruster specific impulse= | | 462 | sec |

| | | |
|---|---|---|
| The main propulsion system additional delta v may be used for additional on-orbit maneuvers | | |
| and to adjust the mission velocity requirements based on trajectory analysis results. | | |
| The ascent RCS mission velocity requirement is used for ascent vehicle roll control if differential | | |
| throttling is selected for vehicle thrust vector control. | | |
| RCS ascent mission velocity (applied to GLOW)= | 0 | ft/sec |
| RCS on-orbit mission velocity (applied to MECO- residuals)= | 155 | ft/sec |
| RCS reentry mission velocity (applied to dry mass+ payload)= | 40 | ft/sec |
| OMS system on-orbit mission velocity (applied to MECO- residuals)= | 1140 | ft/sec |
| Main propulsion system additional dv (applied to MECO)= | -280 | ft/sec |

| | | | |
|---|---|---|---|
| Vehicle layout: | | | |
| Nose length= | | 5.00 | ft |
| Oxidizer tank fwd radius= | | 13.57 | ft |
| Oxidizer tank aft radius= | | 13.57 | ft |
| Fuel tank half angle= | | 3.95 | deg |
| Payload bay diameter= | | 15.00 | ft |
| Payload bay length= | | 30.00 | ft |
| Payload bay/Oxidizer tank standoff= | | 5.00 | ft |
| Engine bay height= | | 17.75 | ft |
| Oxidizer tank/engine standoff distance= | | 10.00 | ft |
| | | | |
| Crew cabin length= | | 12.50 | ft |
| Crew cabin fwd width= | | 10.00 | ft |
| Crew cabin aft width= | | 17.00 | ft |
| Crew Cabin/payload bay standoff= | | 10.00 | ft |
| | | | |
| Oxidizer/Fuel 2 tank standoff= | | 0.50 | ft |
| Aeroshell standoff= | | 0.50 | ft |

| | | |
|---|---|---|
| Body distribution: (fraction of horizontal tip fin planform area) | | |
| Horizontal tip fin outboard angle= | 11.970 | deg |
| Tip fins= | 1.534 | |
| Elevon= | 0.480 | |
| Rudder= | 0.431 | |

| | | | | | |
|---|---|---|---|---|---|
| | | Body | | TPS | |
| Body Element: | | Coefficient | | Coefficient | |
| Nose | | 2.587 | ft^2/ft^2 | 2.20 | lbm/ft^2 |
| | | | | | |
| Forward Section | | | | | |
| Glove | | 0.710 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| Leeward Surface | | 0.670 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.631 | ft^2/ft^2 | 1.30 | lbm/ft^2 |
| | | | | | |
| Forward Fuel Tank Cone Section | | | | | |
| Glove | | 0.902 | ft^2/ft^2 | 0.65 | lbm/ft^2 |
| Leeward Surface | | 0.706 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.655 | ft^2/ft^2 | 1.00 | lbm/ft^2 |
| | | | | | |
| Aft Fuel Tank Cone Section | | | | | |
| Glove | | 0.883 | ft^2/ft^2 | 0.60 | lbm/ft^2 |
| Leeward Surface | | 0.713 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.655 | ft^2/ft^2 | 0.90 | lbm/ft^2 |
| | | | | | |
| Thrust Structure Section | | | | | |
| Glove | | 0.879 | ft^2/ft^2 | 0.55 | lbm/ft^2 |
| Leeward Surface | | 0.720 | ft^2/ft^2 | 0.30 | lbm/ft^2 |
| Windward Surface | | 0.687 | ft^2/ft^2 | 0.80 | lbm/ft^2 |
| | | | | | |
| Body flaps= | | | | 2.00 | lbm/ft^2 |
| Tip fin leading edge= | | | | 2.00 | lbm/ft^2 |
| Tip fin windward= | | | | 0.80 | lbm/ft^2 |
| Tip fin leeward= | | | | 0.65 | lbm/ft^2 |
| Elevons= | | | | 2.00 | lbm/ft^2 |
| Rudder= | | | | 2.00 | lbm/ft^2 |

| | | |
|---|---|---|
| Tip fin surface area breakdown: | | |
| Tip fin surface area/tip fin planform area= | 2.06 | ft^2/ft^2 |
| Tip fin leading edge area/tip fin surface area= | 0.10 | ft^2/ft^2 |
| Tip fin windward surface area/tip fin surface area= | 0.42 | ft^2/ft^2 |
| Tip fin leeward surface area/tip fin surface area= | 0.48 | ft^2/ft^2 |

| | | | |
|---|---|---|---|
| Calculated Data: | | | |
| Mission velocity (dV) (required) | | 30,327 | ft/sec |

| Engine data: | | Mode 1 | Mode 2 | Total | |
|---|---|---|---|---|---|
| Engine Thrust (single engine) (sl) | | 508,913 | 0 | | lbf |
| Engine Thrust (single engine) (vac) | | 587,653 | 254,481 | | lbf |
| Engine masses (single engine) | | 6,139 | 6,321 | | lbm |
| Number of engines | | 5 | 5 | | |
| Total sea level thrust | | 2,544,564 | 0 | 2,544,564 | lbf |
| Total vacuum thrust | | 2,938,266 | 1,272,405 | 2,938,266 | lbf |

| | | | | |
|---|---|---|---|---|
| Burn one vacuum Isp (Iv1) correction based on burn flag) | | | | |
| Thrust split (Fv2/Fvtotal) | | | 0 | |
| Iv1 | | | 385.10 | sec |

| | | | | |
|---|---|---|---|---|
| Burn Two: | | | | |
| Stage velocity | | | 19,830 | ft/sec |
| Iv2 | | | 452.70 | sec |
| Mass ratio (r2) | | | 3.9019 | |
| Burn out weight (W2i) | | | 232,927 | lbm |
| Initial weight (W2o) | | | 908,861 | lbm |
| Propellant (Wf2) | | | 675,934 | lbm |
| Initial acceleration (No2) | | | 0.400 | g |
| | | | | |
| Burn One: | | | | |
| Structure (Wst1) | | | 207,927 | lbm |
| Stage velocity (dV1) (required) | | | 10,497 | ft/sec |
| Iv1 | | | 385.10 | sec |
| Mass ratio (r1) (required) | | | 2.3331 | |
| Burn out weight (W1i) | | | 908,861 | lbm |
| Gross Loft Off Weight | | | 2,120,470 | lbm |
| Propellant (Wf1) | | | 1,211,610 | lbm |
| Initial accelerating (No1) | | | 1.200 | g |
| Mode 1 propellant | | | 1,211,610 | lbm |
| Mode 2 propellant | | | 0 | lbm |
| Mass flow rate | | | 7,630 | lbm/sec |
| Propellant density | | | 46.39 | lbm/ft^3 |

| Results: | | |
|---|---|---|
| Gross Lift Off Weight (GLOW) | 2,120,470 | lbf |
| Total Structural Mass (includes residuals) | 207,927 | lbf |
| Total Propellant Mass | 1,887,543 | lbf |
| Total mode 1 propellant | 1,211,610 | lbm |
| Total mode 2 propellant | 675,934 | lbm |
| Total oxidizer | 1,508,791 | lbm |
| Total fuel 1 | 244,745 | lbm |
| Total fuel 2 | 133,007 | lbm |
| | | |
| Burn 1: | | |
| Initial throttle setting= | 100.00 | % |
| End of burn thrust to weight= | 3.233 | g |
| Initial propellant: | | |
| Oxidizer= | 930,516 | lbm |
| Fuel 1= | 244,745 | lbm |
| Fuel 2= | 36,348 | lbm |
| | | |
| Burn 2: | | |
| Initial throttle setting= | 100.00 | % |
| End of burn thrust to weight= | 5.463 | g |
| Initial propellant: | | |
| Oxidizer= | 579,275 | lbm |
| Fuel 1= | 0 | lbm |
| Fuel 2= | 2,836 | lbm |
| | | |
| If the vehicle has ascent RCS propellant, the vehicle GLOW and liftoff thrust must be modified to reflect the additional vehicle mass. | | |
| Vehicle GLOW= | 2,120,470 | lbm |
| Ascent RCS Propellant= | 0 | lbm |
| Revised Vehicle GLOW= | 2,120,470 | lbm |

| |
|---|
| This section is used to calculate the velocity required to reach orbit, based on the relationship between total dv requirements and initial burn T/W ratios. |
| dv parametrics were generated by off line trajectory executions. |
| |
| 15x220 NM orbit, i= 28.5 deg, launch due east from KSC. |
| Mission delta v includes 1% FPR |
| Nominal data, No1= 1.565g, No2= 1.423g, dv= 29,911 fps. |
| |
| For purposes of mission velocity requirement, stage 1 acceleration is constrained between 1.150 and 1.475 g and stage 2 acceleration is con strained between 0.662 and 1.422 g. |

| | Burn 1 | | | | Burn 2 | |
|---|---|---|---|---|---|---|
| T/W | ddv | counter | | T/W | ddv | counter |
| 1.150 | 630 | 1 | | 0.662 | 1,609 | 1 |
| 1.216 | 380 | 2 | | 0.696 | 1,394 | 2 |
| 1.302 | 107 | 3 | | 0.730 | 1,198 | 3 |
| 1.388 | -93 | 4 | | 0.798 | 918 | 4 |
| 1.475 | -185 | 5 | | 0.867 | 688 | 5 |
| 1.519 | -172 | 6 | | 1.005 | 398 | 6 |
| 1.565 | 0 | 7 | | 1.143 | 201 | 7 |
| | | | | 1.238 | 80 | 8 |
| | | | | 1.423 | 0 | 9 |

| Allowable stage initial accelerations: | | | | | |
|---|---|---|---|---|---|
| | Sizing value | Max Filter | Min Filter | Input Values | |
| No1= | 1.200 | 1.300 | 1.200 | 1.200 | g |
| No2= | 1.400 | 1.400 | 1.400 | 1.400 | g |
| | | | | | |
| Burn 1 throttle setting= | | | 1.000 | % | |
| Burn 2 throttle setting= | | | 1.000 | % | |
| | | | | | |
| Acceleration at the end of burn 1= | | | 3.233 | g | |
| Acceleration at the end of burn 2= | | | 5.463 | g | |

| Mission velocity correction for the burn initial thrust to weight ratios used | | | | |
|---|---|---|---|---|
| Reference mission velocity= | | | 29,911 | ft/sec |
| ddv1= | | | 441 | ft/sec |
| ddv2= | | | 10 | ft/sec |
| Corrected mission velocity= | | | 30,362 | ft/sec |

| The nominal orbit is inclination= 28.5 deg, perigee= 15 NM and apogee= 220 NM. | | | |
|---|---|---|---|
| Adjust for a different inclination | | | |
| Mission inclination= | | 51.60 | deg |
| Reference mission velocity= | | 30,362 | ft/sec |
| Velocity of the Earth's equatorial spin= | | 1,526 | ft/sec |
| Velocity of the Earth's spin at i= 28.5= | | 1,341 | ft/sec |
| Vel of the Earth's spin at mission inc.= | | 948 | ft/sec |
| Corrected mission velocity= | | 30,756 | ft/sec |

| | | | | |
|---|---|---|---|---|
| To adjust for a different perigee and apogee, at the reference perigee (15 NM) do a burn to reach the mission apogee. At the nominal apogee, do a burn to bring the perigee up to the mission perigee. | | | | |
| Earths radius= | 20,925,673 | ft | | |
| Earths gravitational parameter | 1.408E+16 | ft^3/sec^2 | | |
| Mission perigee= | 50.00 | NM | | |
| Mission apogee= | 100.00 | NM | | |
| | | | | |
| Nominal mission perigee= | 15.00 | NM | | |
| Nominal mission apogee= | 220.00 | NM | | |
| Nominal mission perigee= | 21,016,815 | ft | | |
| Nominal mission apogee= | 22,262,418 | ft | | |
| Nominal mission semi major axis= | 21,639,616 | ft | | |
| Nominal mission eccentricity= | 0.02878063 | | | |
| Nominal mission apogee velocity= | 24,781 | ft/sec | | |
| Nominal mission perigee velocity= | 26,250 | ft/sec | | |
| | | | | |
| Transfer orbit 1 perigee= | 15.00 | NM | | |
| Transfer orbit 1 apogee= | 100.00 | NM | | |
| Transfer orbit 1 perigee= | 21,016,815 | ft | | |
| Transfer orbit 1 apogee= | 21,533,284 | ft | | |
| Transfer orbit 1 semi major axis= | 21,275,050 | ft | | |
| Transfer orbit 1 eccentricity= | 0.012137922 | | | |
| Transfer orbit 1 apogee velocity= | 25,412 | ft/sec | | |
| Transfer orbit 1 perigee velocity= | 26,037 | ft/sec | | |
| | | | | |
| Transfer orbit 2 perigee= | 50.00 | NM | | |
| Transfer orbit 2 apogee= | 100.00 | NM | | |
| Transfer orbit 2 perigee= | 21,229,479 | ft | | |
| Transfer orbit 2 apogee= | 21,533,284 | ft | | |
| Transfer orbit 2 semi major axis= | 21,381,382 | ft | | |
| Transfer orbit 2 eccentricity= | 0.007104446 | | | |
| Transfer orbit 2 apogee velocity= | 25,477 | ft/sec | | |
| Transfer orbit 2 perigee velocity= | 25,841 | ft/sec | | |

| | | |
|---|---|---|
| Results from change to a new orbit: | | |
| Reference mission velocity= | 30,756 | ft/sec |
| Perigee burn delta v= | -213 | ft/sec |
| Apogee burn delta v= | 65 | ft/sec |
| Corrected mission velocity= | 30,607 | ft/sec |

| | | |
|---|---|---|
| Results from additional mission velocity: | | |
| Reference mission velocity= | 30,607 | ft/sec |
| Additional MPS mission velocity= | -280 | ft/sec |
| Final mission velocity= | 30,327 | ft/sec |

### 3.3.4 Tank Sizing Spreadsheet

| |
|---|
| This spreadsheet is used to size the propellant tanks for the lifting body vehicle. |
| The oxidizer is stored in a cylindrical tank with ellipsoidal endcaps. |
| The oxidizer tank is aft of the payload bay. |
| |
| Fuel is stored in two bent biconical tanks with ellipsoidal endcaps. |
| The biconical fuel tanks lay along side the oxidizer tank and stretch from the payload bay |
| to the oxidizer tank aft endcap. |
| The maximum fuel tank thickness is at the oxidizer tank forward endcap. |
| |
| In the tri propellant case, there are four fuel tanks. Fuel 1 is stored in the forward |
| two conical fuel tanks and fuel 2 is stored in the aft two bent biconical fuel tanks. |
| |
| In the bi propellant case, fuel 2 is stored in two bent biconical fuel tanks. There |
| are no fuel 1 tanks. |
| |
| An iterative approach is used to match the length of the fuel tanks to the length of the |
| oxidizer tank and the payload bay. |

| | | |
|---|---|---|
| Input data: | | |
| Oxidizer mass= | 1,509,742 | lbm |
| Oxidizer density= | 71.20 | lbm/ft^3 |
| Oxidizer ullage= | 0.05 | |
| Fuel 1 mass= | 244,732 | lbm |
| Fuel 1 density= | 50.50 | lbm/ft^3 |
| Fuel 1 ullage= | 0.05 | |
| Fuel 2 mass= | 133,005 | lbm |
| fuel 2 density= | 4.43 | lbm/ft^3 |
| Fuel 2 ullage= | 0.05 | |
| Oxidizer tank fwd radius= | 13.57 | ft |
| Oxidizer tank aft radius= | 13.57 | ft |
| Fuel tank half angle= | 3.95 | deg |
| Payload bay diameter= | 15.00 | ft |
| Payload bay length= | 30.00 | ft |
| Payload bay stand off distance= | 5.00 | ft |
| Engine length= | 13.26 | ft |
| Engine bay height= | 17.75 | ft |
| Oxidizer tank/engine standoff distance= | 10.00 | ft |

| | | |
|---|---|---|
| The oxidizer tank is a cone with ellipsoidal endcaps. | | |
| Oxidizer tank: | | |
| Volume= | 22,264 | ft^3 |
| Fwd endcap radius= | 13.57 | ft |
| Fwd endcap height= | 9.60 | ft |
| Fwd endcap Volume= | 3,701 | ft^3 |
| Barrel section length= | 25.69 | ft |
| Aft endcap radius= | 13.57 | ft |
| Aft endcap depth= | 9.60 | ft |
| Aft endcap Volume= | 3,701 | ft^3 |
| Barrel Volume= | 14,863 | ft^3 |
| | | |
| Barrel section length= | 25.69 | ft |

| | | |
|---|---|---|
| (from payload bay fwd edge to fwd endcap/barrel interface) | | |
| Length to fwd endcap= | 44.60 | ft |

| | | |
|---|---|---|
| Length of the ox tank fwd endcap+ the ox tank barrel+ the ox tank aft endcap depth+ the payload bay length+ the payload bay standoff distance: | | |
| distance= | 79.88 | ft |

| | | |
|---|---|---|
| Distance from fuel tank midpoint to engine nozzle exit plane= | 58.55 | ft |
| Nozzle radius= | 8.88 | ft |

| | | |
|---|---|---|
| Fuel 1 tank: | | |
| Volume= | 2,544 | ft^3 |
| Tank half angle= | 3.95 | deg |
| Fwd endcap radius= | 5.66 | ft |
| Fwd endcap height= | 4.00 | ft |
| Fwd endcap Volume= | 268 | ft^3 |
| Barrel section length= | 15.20 | ft |
| Aft endcap radius= | 6.71 | ft |
| Aft endcap depth= | 4.74 | ft |
| Aft endcap Volume= | 447 | ft^3 |
| Barrel Volume= | 1,829 | ft^3 |
| | | |
| Barrel section length= | 15.20 | ft |

| | | |
|---|---|---|
| Fuel tank standoff distance= | 15.45 | ft |

| | | |
|---|---|---|
| Fuel 2 tank: | | |
| Volume= | 15,762 | ft^3 |
| Tank half angle= | 3.95 | deg |
| Fwd endcap radius= | 8.72 | ft |
| Fwd endcap height= | 6.17 | ft |
| Fwd endcap Volume= | 983 | ft^3 |
| Total barrel section length= | 49.35 | ft |
| Midpoint (barrel 1) length= | 13.96 | ft |
| Midpoint radius= | 9.60 | ft |
| Barrel 1 volume= | 3,716 | ft^3 |
| Aft endcap radius= | 9.20 | ft |
| Aft endcap depth= | 6.50 | ft |
| Aft endcap Volume= | 1,151 | ft^3 |
| Barrel 2 Volume= | 9,913 | ft^3 |
| Barrel section 2 length= | 35.40 | ft |
| | | |
| Total barrel section length= | 49.35 | ft |

| | | |
|---|---|---|
| Fuel tank axial length: | | |
| Fu1 tank barrel length= | 15.20 | ft |
| Fuel tank standoff distance= | 15.45 | ft |
| Fu2 tank barrel length= | 49.35 | ft |
| Fu2 tank aft endcap height= | 6.50 | ft |
| Total= | 86.49 | ft |

Since the centerline of the fuel tank is at an angle to the vehicle centerline, the effective fuel tank length is cos(tank half angle)*axial fuel tank length.

| | | | |
|---|---|---|---|
| The forward fuel tank cone: | | | |
| Tank half angle= | 3.95 | deg | |
| Tank axial length= | 44.50 | ft | |
| Tank effective length= | | 44.49 | ft |
| | | | |
| Aft fuel tank cone: | | | |
| Tank half angle= | 0.79 | deg | |
| Tank axial length= | 35.40 | ft | |
| Tank effective length= | | 35.39 | ft |
| | | | |
| Total tank effective length= | | 79.88 | ft |

The fuel tank fwd endcap radius is revised to force convergence between oxidizer and fuel tank lengths.

| | | |
|---|---|---|
| Fwd endcap radius= | 5.66 | ft |

### 3.3.5 Weights Spreadsheet

This spreadsheet is for sizing the vehicle (lifting body VTHL).

| | | |
|---|---|---|
| Input data: | | |
| Number of crew= | 0 | |
| Crew cabin volume= | 0 | ft^3 |
| Number of days on orbit= | 7 | |
| Average on orbit power usage= | 5.00 | kw |
| Average on orbit heat rejection requirement= | 10.00 | kw |
| Maximum axial acceleration (No) | 3.00 | g |
| Maximum normal acceleration= | 1.60 | g |
| Lift-off thrust to weight | 1.200 | g |
| Factor of safety | 1.40 | |
| Main propellant feed line and press sys= | 65 | lbm-sec/lbf |
| Vehicle mass prop contingency factor= | 0.15 | |
| Avionics= | 710 | lbm |
| Range safety= | 0 | lbm |
| Vehicle body constant= | 1.32 | lbm/(ft^2*g^(1/3)) |
| Crew Cabin Body Constant (Bo)= | 0 | lbm/number of crew^(1/2) |
| Landing gear constant (Kl)= | 0.03 | lbm/lbm |
| Body insulation constant (Kbi)= | 0 | lbm/ft^2 (if hot structure selected) |
| | | |
| Gimbal Actuator unit mass= | 0 | lbm/lbf |
| Thrust structure (max thrust)= | 0.00207 | lbm/lbf |
| Thrust structure (number of engines)= | 0.00039 | lbm/lbf |
| | | |
| Prime Power (PWc) (aero surface)= | 0.274 | lbm/ft^2 |
| Prime Power (PWe) (engine gimbaling)= | 0.00E+00 | lbm/lbf |
| Prime Power (PWa) (avionics)= | 0.155 | lbm/lbm |
| Electrical Power Conv & Dist= | 0.020 | lbm/lbm |
| Hydraulic Power Conv & Dist (aero surface)= | 0.000 | lbm/ft^2 |
| Hydraulic Power Conv & Dist (engine gimbaling)= | 0.000 | lbm/lbf |
| Fuel cell unit mass (FCw)= | 28.71 | lbm/kw |
| Fuel cell reactants unit mass (FCc)= | 29.26 | lbm/kw-day |
| | | |
| ECLSS crew cabin constant (Ec)= | 0 | lbm/(ft^(1/3))^0.75 |
| ECLSS crew supplies constant (Eo)= | 0 | lbm/crew member-day |
| ECLSS avionics waste heat (Ea)= | 0.22 | lbm/lbm |
| Active thermal control loop unit mass (Ew)= | 200.00 | lbm/kw |
| | | |
| Personnel waste systems (PPf)= | 0 | lbm |
| Personnel seats and crew related (PPs)= | 0 | lbm/crew member |
| Personnel miscellaneous (Pm)= | 0 | lbm |
| Personnel mass (Pp)= | 0.00 | lbm/crew member |
| | | |
| Surface control actuator constant (Ssc)= | 2.61 | lbm/ft^2 |
| Surface control miscellaneous (Spc)= | 200 | lbm |

| Tank definition: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
|---|---|---|---|---|
| Propellant mass= | 1,509,742 | 244,732 | 133,005 | lbm |
| Residual A (propellant mass)= | 0.0038 | 0.0038 | 0.0018 | lbm/lbm |
| Residual B (engine thrust)= | 0.0010 | 0.0010 | 0.0012 | lbm/lbf |
| Ullage pressure= | 20.00 | 20.00 | 20.00 | psi |
| TPS unit mass= | 0.25 | 0.000 | 0.25 | lbm/ft^2 |
| Burn flag= | 3 | | | |
| Number of fuel tanks= | 2 | | | |
| | | | | |
| Propellant tank materials: | Ox tank | Fuel 1 tank | Fuel 2 tank | |
| Density= | 0.098 | 0.098 | 0.057 | lbm/in^3 |
| Ftu= | 65,600 | 65,600 | 90,400 | psi |

| | Endcap | | (for elliptical endcaps) | |
|---|---|---|---|---|
| Calculated geometry: (tanks are conical) | Radius (ft) | Height (ft) | k | e |
| Oxidizer tank fwd radius= | 13.57 | 9.60 | 1.41421 | 0.70711 |
| Oxidizer tank aft radius= | 10.57 | 9.60 | 1.41421 | 0.70711 |
| Fuel 1 tank fwd radius= | 5.66 | 4.00 | 1.41421 | 0.70711 |
| Fuel 1 tank aft radius= | 6.71 | 4.74 | 1.41421 | 0.70711 |
| Fuel 2 tank fwd radius= | 8.72 | 6.17 | 1.41421 | 0.70711 |
| Fuel 2 tank aft radius= | 9.20 | 6.50 | 1.41421 | 0.70711 |

The oxidizer is in one cylindrical tank, fuel 1 is in two conical tanks and fuel 2 is in two bent double cone tanks. All tanks have elliptical endcaps.

| Propellant tank geometry: | Oxidizer | Fuel 1 | Fuel 2 | |
|---|---|---|---|---|
| Density= | 71.20 | 50.50 | 4.43 | lbm/ft^3 |
| Tank Volume= | 22,264 | 2,544 | 15,762 | ft^3 |
| Tank upper endcap radius= | 13.57 | 5.66 | 8.72 | ft |
| Tank upper endcap health= | 9.60 | 4.00 | 6.17 | ft |
| Midpoint radius= | | | 9.69 | ft |
| Tank lower endcap radius= | 13.57 | 6.71 | 9.20 | ft |
| Tank lower endcap health= | 9.60 | 4.74 | 6.50 | ft |
| Barrel section 1 length= | 25.69 | 15.20 | 13.95 | ft |
| Barrel section 2 length= | | | 35.40 | ft |
| | | | | |
| Cone 1 half angle= | 0.00 | 3.96 | 3.96 | deg |
| Cone 2 half angle= | | | 0.79 | deg |
| Upper endcap area= | 470 | 82 | 194 | ft^2 |
| Lower endcap area= | 470 | 115 | 216 | ft^2 |
| Barrel section area 1= | 2,191 | 592 | 809 | ft^2 |
| Barrel section area 2= | | | 2,100 | ft^2 |
| Total area= | 3,130 | 788 | 3,318 | ft^2 |
| Tank insulation= | 0.250 | 0.000 | 0.250 | lbm/ft^2 |

Oxidizer is in one cylinder, fuel 1 is in two conical tanks and fuel 2 is in two bent conical tanks.

| Propellant tank mass data: | | Oxidizer | Fuel 1 | Fuel 2 | |
|---|---|---|---|---|---|
| Ullage (upper endcap) pressure= | | 20.00 | 20.00 | 20.00 | psi |
| Lower endcap pressure= | | 36.24 | 26.40 | 21.82 | psi |
| Average barrel section pressure= | | 27.62 | 23.20 | 20.91 | psi |
| Upper endcap thickness= | | 0.042 | 0.017 | 0.020 | in |
| Lower endcap thickness= | | 0.074 | 0.027 | 0.023 | in |
| Barrel thickness 1= | | 0.096 | 0.040 | 0.036 | in |
| Barrel thickness 2= | | | | 0.036 | in |
| Endcap (pressurized structure) mass= | | 768 | 64 | 71 | lbm |
| Barrel section (pressurized structure) mass= | | 2,967 | 334 | 854 | lbm |
| Total mass (based on pressurized structure)= | | 3,735 | 398 | 925 | lbm |
| Delta mass (semi empirical correction)= | | 4,791 | 766 | 5,181 | lbm |
| Delta mass (density correction)= | | 4,596 | 741 | 2,953 | lbm |
| Total mass = | | 8,431 | 1,139 | 3,878 | lbm |
| Unit mass= | | 0.3787 | 0.4476 | 0.2460 | lbm/ft^3 |

| Vehicle geometry: | | | | | |
|---|---|---|---|---|---|
| Body dimensions: | | | | Length | |
| Nose length= | | 5.00 | ft | 5.00 | ft |
| | | | | | |
| Crew cabin length= | | 12.50 | ft | 12.50 | ft |
| Crew cabin fwd width= | | 10.00 | ft | | |
| Crew cabin aft width= | | 17.00 | ft | | |
| Crew Cabin/payload bay standoff= | | 10.00 | ft | 10.00 | ft |
| | | | | | |
| Payload bay length= | | 30.00 | ft | 30.00 | ft |
| Payload bay diameter= | | 15.00 | ft | | |
| Payload bay/Oxidizer tank standoff= | | 5.00 | ft | 5.00 | ft |
| | | | | | |
| Oxidizer tank length= | | 25.69 | ft | 44.88 | ft |
| Oxidizer tank fwd diameter= | | 27.14 | ft | | |
| Oxidizer tank aft diameter= | | 27.14 | ft | | |
| Oxidizer tank fwd endcap health= | | 9.60 | ft | | |
| Oxidizer tank aft endcap health= | | 9.60 | ft | | |
| Oxidizer tank/engine standoff= | | 10.00 | ft | 10.00 | ft |
| Oxidizer/Fuel 2 tank standoff= | | 0.50 | ft | | |
| Engine length= | | 13.26 | ft | 13.26 | ft |
| Engine bay health= | | 17.75 | ft | | |
| | | | | | |
| Aeroshell standoff= | | 0.50 | ft | | |
| Fuel tank fwd diameter= | | 11.32 | ft | | |
| Fuel tank mid diameter= | | 19.37 | ft | | |
| Fuel tank aft diameter= | | 18.39 | ft | | |
| | | | | | |
| Total length= | | | | 130.64 | ft |

| | | | | | | |
|---|---|---|---|---|---|---|
| Vehicle exterior angles: | | | | | | |
| (The groundrules are theta1 should be between 10 and 15 degrees larger than | | | | | | |
| theta2. Allowable value for theta4 is a function of aft body length. Ratio of aft body | | | | | | |
| length/total body length is greater than 0.450) | | | | | | |
| | | | | | | |
| Theta1 (nose): | | | | | | |
| r1= | 5.60 | ft | | | | |
| r2= | 19.82 | ft | | | | |
| l= | 22.50 | ft | | | | |
| Theta1= | 32.47 | deg | | | | |
| | | | | | | |
| Theta2 (fwd fuel cone): | | | | | | |
| r1= | 19.82 | ft | | | | |
| r2= | 33.94 | ft | | | | |
| l= | 44.60 | ft | | | | |
| Theta2= | 17.58 | deg | Theta1- Theta2= | | 14.89 | deg |
| | | | | | | |
| Theta3 (aft fuel cone): | | | | | | |
| r1= | 33.94 | ft | | | | |
| r2= | 32.96 | ft | | | | |
| l= | 28.79 | ft | | | | |
| Theta3= | 1.95 | deg | | | | |
| | | | | | | |
| Theta4 | | | | | | |
| Theta2= | 17.58 | deg | | | | |
| Theta3= | 1.95 | deg | | | | |
| Theta4= | 19.52 | deg | | | | |
| | | | | | | |
| Aft body length: | | | | | | |
| Oxidizer tank barrel section= | | | 25.69 | ft | | |
| Oxidizer tank aft endcap= | | | 9.60 | ft | | |
| Oxidizer tank/engine standoff= | | | 10.00 | ft | | |
| Engine length= | | | 13.26 | ft | | |
| Total= | | | 58.55 | ft | | |

| | | | | |
|---|---|---|---|---|
| Define body planform area: | | | | |
| Nose is a semi ellipse. | | | | |
| Base diameter= | 11.00 | ft | | |
| Length= | 5.00 | ft | | |
| Area= | | | 43 | ft^2 |
| | | | | |
| The nose to fuel tank fwd barrel section is a trapezoid. | | | | |
| Fwd diameter= | 11.00 | ft | | |
| Aft diameter= | 39.63 | ft | | |
| Length= | 22.50 | ft | | |
| Area= | | | 570 | ft^2 |
| | | | | |
| The fuel tank fwd barrel to the oxidizer tank fwd barrel is a trapezoid. | | | | |
| Fwd diameter= | 39.63 | ft | | |
| Aft diameter= | 67.88 | ft | | |
| Length= | 44.50 | ft | | |
| Area= | | | 2,397 | ft^2 |
| | | | | |
| The oxidizer tank fwd barrel to the oxidizer tank aft endcap is a trapezoid. | | | | |
| Fwd diameter= | 67.88 | ft | | |
| Aft diameter= | 65.92 | ft | | |
| Length= | 28.79 | ft | | |
| Area= | | | 1,926 | ft^2 |
| | | | | |
| The thrust structure is a trapezoid. | | | | |
| Fwd diameter= | 65.92 | ft | | |
| Length= | 16.50 | ft | | |
| Aft body angle (theta3)= | 1.95 | deg | | |
| Aft diameter= | 64.80 | ft | | |
| Area= | | | 1,079 | ft^2 |
| | | | | |
| The engine bay is a trapezoid (also body flap planform area). | | | | |
| Fwd diameter= | 64.80 | ft | | |
| Length= | 13.26 | ft | | |
| Aft body angle (theta3)= | 1.95 | deg | | |
| Aft diameter= | 63.90 | ft | | |
| Area= | | | 853 | ft^2 |
| | | | | |
| Horizontal tip fin is two triangles. | | | | |
| Length= | 58.56 | ft | | |
| Outboard angle= | 11.97 | deg | | |
| Width= | 14.41 | ft | | |
| Area= | | | 843 | ft |
| | | | | |
| Vehicle body planform area= | | | 7,711 | ft^2 |

| | | | | |
|---|---|---|---|---|
| Body distribution: (fraction of horizontal tip fin planform area) | | | | |
| Horizontal tip fin outboard angle= | | | 11.970 | deg |
| Tip fins= | | | 1.534 | |
| Elevon= | | | 0.480 | |
| Rudder= | | | 0.431 | |

| | | |
|---|---|---|
| Body Flap planform area (also engine bay planform area): | | |
| For engine protection, body flap is engine length and | | |
| width is fuel tank centerline. | | |
| Engine bay planform area= | 853 | ft^2 |

| | | | |
|---|---|---|---|
| Tip Fins: | | | |
| Tip Fin Constant= | 1.00 | lbm/ft^2-g^(1.24) | |
| Tip Fin planform area= | 1,294 | ft^2 | |
| Tip Fin Mass= | 5,116 | lbm | |
| Tip fin surface area/tip fin planform area= | | 2.06 | ft^2/ft^2 |
| Tip fin leading edge area/tip fin surface area= | | 0.10 | ft^2/ft^2 |
| Tip fin windward surface area/tip fin surface area= | | 0.42 | ft^2/ft^2 |
| Tip fin leeward surface area/tip fin surface area= | | 0.48 | ft^2/ft^2 |
| Tip fin surface area= | 2,665 | ft^2 | |
| Tip fin leading edge area= | 267 | ft^2 | |
| Tip fin windward surface area= | 1119 | ft^2 | |
| Tip fin leeward surface area= | 1279 | ft^2 | |

| | | | | |
|---|---|---|---|---|
| Control surface masses: | | | | |
| Both elevons and rudder are split into two sections. | | | | |
| Control surface constant (Bbf)= | 1.17 | lbm/(ft^2)^1.15 | | |
| | | | | |
| | Body Flap | Elevons | Rudder | |
| Planform Area= | 853 | 405 | 364 | ft^2 |
| Mass= | 2,747 | 1,051 | 928 | lbm |

| | | Body Coefficient (ft^2/ft^2) | Planform Area (ft^2) | Body Surface Area (ft^2) | TPS Coefficient (lbm/ft^2) | TPS Mass (lbm) |
|---|---|---|---|---|---|---|
| Body Element: | | | | | | |
| Nose | | 2.587 | 43 | 112 | 2.20 | 246 |
| | | | | | | |
| Forward Section | | | 570 | | | |
| Glove | | 0.710 | | 404 | 0.80 | 324 |
| Leeward Surface | | 0.670 | | 382 | 0.30 | 114 |
| Windward Surface | | 0.631 | | 359 | 1.30 | 467 |
| | | | | | | |
| Forward Fuel Tank Cone Section | | | 2,397 | | | |
| Glove | | 0.902 | | 2,162 | 0.65 | 1,406 |
| Leeward Surface | | 0.706 | | 1,692 | 0.30 | 508 |
| Windward Surface | | 0.655 | | 1,570 | 1.00 | 1,570 |
| | | | | | | |
| Aft Fuel Tank Cone Section | | | 1,926 | | | |
| Glove | | 0.883 | | 1,700 | 0.60 | 1,020 |
| Leeward Surface | | 0.713 | | 1,373 | 0.30 | 412 |
| Windward Surface | | 0.655 | | 1,261 | 0.90 | 1,135 |
| | | | | | | |
| Thrust Structure Section | | | 1,079 | | | |
| Glove | | 0.879 | | 948 | 0.55 | 521 |
| Leeward Surface | | 0.720 | | 777 | 0.30 | 233 |
| Windward Surface | | 0.687 | | 741 | 0.80 | 593 |
| | | | | | | |
| Total | | | | 13,483 | | 8,549 |

| | | |
|---|---|---|
| Aeroshell mass: | | |
| Aeroshell unit mass= | 1.727 | lbm/ft^2 |
| Aeroshell mass= | 23,287 | lbm |

| Vehicle TPS: | | Unit mass | | TPS Area | | Mass | |
|---|---|---|---|---|---|---|---|
| Body= | | | | | | 8,549 | lbm |
| Body flaps= | | 2.00 | lbm/ft^2 | 853 | ft^2 | 1,706 | lbm |
| Tip fin leading edge= | | 2.00 | lbm/ft^2 | 267 | ft^2 | 533 | lbm |
| Tip fin windward side= | | 0.80 | lbm/ft^2 | 1119 | ft^2 | 896 | lbm |
| Tip fin leeward side= | | 0.65 | lbm/ft^2 | 1279 | ft^2 | 832 | lbm |
| Elevons= | | 2.00 | lbm/ft^2 | 405 | ft^2 | 810 | lbm |
| Rudder= | | 2.00 | lbm/ft^2 | 364 | ft^2 | 727 | lbm |
| Total= | | | | | | 14,053 | lbm |

| | | | |
|---|---|---|---|
| Engine heat shield mass: | | | |
| (engine bay is to centerline of fuel tanks) | | | |
| Base engine heat shield unit mass= | | 1.64 | lbm/ft^2 |
| Engine bay diameter= | | 63.90 | ft |
| Engine bay health= | | 17.75 | ft |
| Vehicle base surface area= | | 1,134 | ft^2 |
| Engine heat shield mass= | | 1,860 | lbm |

| Body insulation: | | | |
|---|---|---|---|
| (insulation covers body surface area and will be used | | | |
| only if TPS does not include sufficient insulation) | | | |
| Body insulation coefficient= | | 0.00 | lbm/ft^2 |
| Body insulation surface area= | | 13,483 | ft^2 |
| Body insulation mass= | | 0 | lbm |

| Engine data: | | Number | Total Fv | Length | Unit mass | Iv |
|---|---|---|---|---|---|---|
| | | | (lbf) | (ft) | (lbm) | (sec) |
| Mode 1 engine= | | 6 | 2,938,176 | 13.26 | 6,139 | 365.10 |
| Mode 2 engines= | | 5 | 1,272,405 | 13.26 | 5,321 | 492.70 |
| Maximum Vacuum thrust= | | | 2,938,176 | lbf | | |
| Engine length= | | | 13.26 | | | |

| Propellant feed and pressurization system: | | | | | | |
|---|---|---|---|---|---|---|
| Included are the main propellant feed system, main tank pressurization system, | | | | | | |
| and the purge and vent system. | | | | | | |
| RPM= | | | | 55.00 | lbm-sec/ft^3 | |
| Mass flow rate= | | | | 7,830 | lbm/sec | |
| Density= | | | | 46.39 | lbm/ft^3 | |
| Feed/press sys= | | | | 9,047 | lbm | |

| OMS/RCS system | | | | | | | |
|---|---|---|---|---|---|---|---|
| RCS system= | | | | 0.000151 | lbm/lbm-ft | | |
| RCS thruster specific impulse (on-orbit)= | | | | 422.00 | sec | | |
| RCS thruster specific impulse (entry)= | | | | 410.00 | sec | | |
| RCS thruster specific impulse (ascent)= | | | | 350.00 | sec | | |
| RCS ascent mission velocity= | | | | 0.00 | ft/sec | | |
| RCS on orbit mission velocity= | | | | 155.00 | ft/sec | | |
| RCS entry mission velocity= | | | | 40.00 | ft/sec | | |
| | | | | | | | |
| OMS system thrust to weight= | | | | 0.04 | g | | |
| OMS engine constant= | | | | 0.035 | lbm/lbf | | |
| OMS prop system weights= | | | | 0.152 | lbm/lbm | | |
| OMS thruster specific impulse= | | | | 462.00 | sec | | |
| OMS system on orbit mission velocity= | | | | 1140 | ft/sec | | |
| | | | | | | | |
| RCS Propellant: | | RCS Ascent | OMS | RCS on orbit | RCS Entry | Total | |
| Mass Ratio= | | 1.0000 | 1.0797 | 1.0115 | 1.0030 | | |
| Start Burn Mass= | | 2,120,406 | 222,913 | 206,456 | 204,113 | | lbm |
| End Burn Mass= | | 2,120,405 | 206,456 | 204,113 | 203,495 | | lbm |
| Propellant load= | | 0 | 16,457 | 2,344 | 618 | 19,418 | lbm |
| | | | | | | | |
| RCS System mass= | | 4,027 | lbm | | | | |
| OMS Engine mass= | | 312 | lbm | | | | |
| OMS System Mass= | | 2,952 | lbm | | | | |
| Total= | | 7,290 | lbm | | | | |

| | | |
|---|---|---|
| Vehicle mass: | | |
| Oxidizer Tank= | 8,431 | lbm |
| Oxidizer Tank Insulation= | 782 | lbm |
| Fuel Tank 1= | 2,278 | lbm |
| Fuel Tank 1 Insulation= | 0 | lbm |
| Fuel Tank 2= | 7,756 | lbm |
| Fuel Tank 2 Insulation= | 1,659 | lbm |
| Aeroshell= | 23,287 | lbm |
| Range Safety= | 0 | lbm |
| Gimbal Actuators= | 0 | lbm |
| Engines= | 30,693 | lbm |
| Thrust structure= | 10,868 | lbm |
| Avionics= | 3,219 | lbm |
| Prime Power= | 2,111 | lbm |
| Power Conv & Dist= | 4,070 | lbm |
| Feed/press sys= | 9,047 | lbm |
| OMS/RCS System= | 7,290 | lbm |
| Crew Cabin= | 0 | lbm |
| Environmental Control= | 2,708 | lbm |
| Landing Gear= | 6,105 | lbm |
| Body Insulation= | 0 | lbm |
| Tip Fins/Chines= | 6,116 | lbm |
| Body Flap= | 2,747 | lbm |
| Elevator= | 1,051 | lbm |
| Rudder= | 928 | lbm |
| Control Surface Actuators= | 4,432 | lbm |
| Payload Bay= | 3,925 | lbm |
| Engine Bay Heat Shield= | 1,860 | lbm |
| TPS= | 14,053 | lbm |
| Dry Mass Contingency= | 23,282 | lbm |
| Dry Mass= | 178,495 | lbm |
| | | |
| On-orbit & Entry OMS/RCS Propellant= | 19,418 | lbm |
| Residual Ascent Propellant= | 10,014 | lbm |
| Personnel Provisions= | 0 | lbm |
| Personnel= | 0 | lbm |
| Burn-out Mass (w/o payload)= | 207,927 | lbm |
| | | |
| Ascent RCS Propellant= | 0 | lbm |
| Total Usable Ascent Propellant= | 1,887,478 | lbm |
| Payload= | 25,000 | lbm |
| Gross Lift Off Weight (GLOW)= | 2,120,405 | lbm |
| | | |
| Main Engine Cut Off (MECO)= | 232,927 | lbm |
| Landed Vehicle mass= | 203,495 | lbm |

| Vehicle cg (distance from vehicle nose) | | | |
|---|---|---|---|
| (vehicle dry mass) | | | |
| | mass (lbm) | distance (ft) | (ft*lbm) |
| Oxidizer Tank= | 8,431 | 84.94 | 716,128 |
| Oxidizer Tank Insulation= | 782 | 84.94 | 66,459 |
| Fuel Tank 1= | 2,278 | 35.10 | 79,958 |
| Fuel Tank 1 Insulation= | 0 | 35.10 | 0 |
| Fuel Tank 2= | 7,756 | 91.39 | 708,805 |
| Fuel Tank 2 Insulation= | 1,659 | 91.39 | 151,609 |
| Aeroshell= | 23,287 | 84.92 | 1,977,443 |
| Range Safety= | 0 | 22.50 | 0 |
| Gimbal Actuators= | 0 | 117.38 | 0 |
| Engines= | 30,693 | 124.01 | 3,806,384 |
| Thrust Structure= | 10,666 | 112.38 | 1,198,629 |
| Avionics= | 3,219 | 11.25 | 36,213 |
| Prime Power= | 2,111 | 11.25 | 23,748 |
| Power Conv & Dist= | 4,070 | 11.25 | 45,788 |
| Feed/Press Sys= | 9,047 | 89.91 | 813,401 |
| OMS/RCS System= | 7,290 | 98.61 | 718,867 |
| Crew Cabin= | 0 | 11.25 | 0 |
| Environmental Control= | 2,708 | 11.25 | 30,487 |
| Personnel Provisions= | 0 | 11.25 | 0 |
| Personnel= | 0 | 11.25 | 0 |
| Landing Gear= | 6,105 | 72.70 | 443,806 |
| Body Insulation= | 0 | 84.92 | 0 |
| Tip Fins/Chines= | 6,116 | 111.13 | 679,648 |
| Body Flap= | 2,747 | 128.64 | 353,430 |
| Elevator= | 1,051 | 128.64 | 135,142 |
| Rudder= | 928 | 128.64 | 119,402 |
| Control Surface Actuators= | 4,432 | 128.64 | 570,189 |
| Payload bay= | 3,926 | 42.50 | 166,813 |
| Engine Bay Heat Shield= | 1,860 | 130.64 | 242,986 |
| TPS= | 14,053 | 65.32 | 917,932 |
| Contingency= | 23,282 | 65.32 | 1,520,809 |
| Payload= | 0 | 42.50 | 0 |
| Total= | 178,495 | | 15624041 |
| | | | |
| cg= 86.97 ft | | | |

# 9.0 SSTO Turnaround Assessment Report

This section contains a copy of the final report regarding an SSTO ground processing turnaround time assessment that was performed by Mr. Wally Eshleman of the Skunk Works under an intercompany work transfer for LMSC, at the request of Gene Austin of the Marshall Space Flight Center. The turnaround time assessment utilized methodologies and tools that have been standardized by the Air Force for reliability, maintainability, and supportability assessments of aircraft in the Air Force inventory.

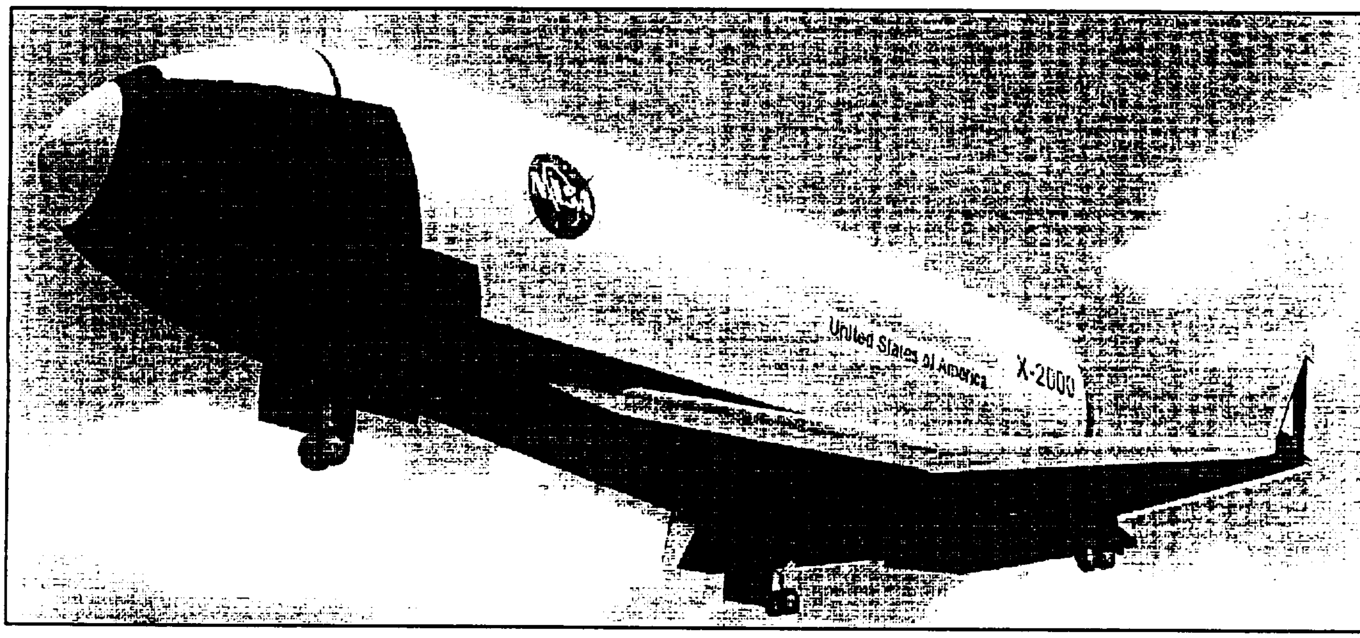
# NASA SSTO TURNAROUND PROCESSING ASSESSMENT

LOCKHEED ADVANCED DEVELOPMENT COMPANY
SUPPORT TO
MSFC CONTRACT NAS8–39208
1 JUNE 1994

HEAVY LIFT LAUNCH VEHICLE CONCEPT REFINEMENT
FOR EARTH TO ORBIT TRANSPORTATION
(OPTION 1)

# LOCKHEED ADVANCED DEVELOPMENT COMPANY
# SUPPORT TO
# MSFC CONTRACT NAS8–39208
# 1 JUNE 1994

## HEAVY LIFT LAUNCH VEHICLE CONCEPT REFINEMENT FOR EARTH TO ORBIT TRANSPORTATION (OPTION 1)

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## 1.0 INTRODUCTION

The purpose of this contract is to perform a standard aircraft Reliability and Maintainability (R&M) engineering evaluation to determine a turnaround time for MFSC's conceptual Single Stage To Orbit (SSTO) reusable space vehicle. The two primary R&M performance indicators are System Reliability (safety, mission success, etc.) and System Availability (ability to readily utilize the system). In this study, only turnaround time and its association with vehicle availability will be analyzed. The standard methodology utilized in this analysis is addressed in a multitude of military specifications, most of which were initiated during the 1950's and have evolved to the point of being contractually applied to all hardware elements of modern U.S. weapons systems.

The math model developed for this conceptual evaluation is addressed, specifically, in USAF MIL-STD-1388, Task 203; Baseline Comparative System (BCS) analysis. Conceptual BCS analysis and predictions are to be used to identify order-of-magnitude, as opposed to specific metrics, for elapsed and total-time processing requirements and implied logistics resources (such as manpower). This is a bottom up approach which employs like-equipment historical failure and maintenance data as the foundation for determining serial and total turnaround maintenance hours required between missions. The processing turnaround burden of any vehicle is a reflection of vehicle system R&M performance as well as functional processing requirements (postflight recovery operations, etc.). Both contribute to the total processing burden.

Existing historical data bases for the Space Shuttle Orbiter and USAF modern military aircraft were interrogated to determine failure rates and maintenance repair times for like vehicle systems R&M performance. The two driving R&M math model parameters are system failure rates and maintenance repair times. Ultimately, Orbiter system failure rates and US military aircraft system repair times are combined to determine SSTO vehicle systems contributions to the total burden. This approach best considers the actual operational mission environment (atmospheric assent, on-orbit time, and reentry) and a modern aircraft type design-for-maximum-availability philosophy. Due to the scope of this study, all analysis will be performed at the system level without consideration for equipment level parameters. The fidelity of this assessment, however, is greatest when performed at the component level, which better reflects the actual conceptual vehicle R&M performance characteristics. Future high resolution studies should be considered which will more accurately consider component technology improvements and integrated systems.

This study includes identification of analogous modern aircraft subsystems which are similar in configuration and componentry to that of the existing Shuttle Orbiter, associated system failure rates, comparison of aircraft and Shuttle Orbiter failure rates for like systems, aircraft maintenance philosophy discussion, MSFC's SSTO turnaround prediction and a hardware demonstration of LADC's internally developed R&M Tool Kit.

### 1.1 R&M HISTORICAL DATA BASES

The historical Shuttle Orbiter R&M performance, at the system level (two digit Work Unit Code (WUC)) is derived from several existing databases and reports. Orbiter system failure rates are identified utilizing the Orbiter Reliability Centered Maintenance Report, dated February 1994, developed by Johnson Controls World Services (JCWS) under contract to KSC-NASA. Although this failure data (Problem Reports per 100 operating hours) is derived from other KSC trending databases, and not an actual record of equipment failures for an actual tracked equipment exposure time (operating time), this data has the highest accuracy found to date by LADC. Additional reference material included JSC's In-flight Anomaly Reports and Orbiter Autoland Reliability Analysis Report, dated April 1993, and LSOC's 1993 Orbiter Processing Study (APU/Hydraulics Baseline Assessment). JSC's Orbiter Autoland Reliability Analysis Report contains component level failure rates and bottom up aircraft-type reliability predictions for subsystem performance. This report reflects typical R&M engineering data and tasks performed for all modern military weapons systems. Data collected in the AFR 66-1 (USAF Automated Historical R&M database) allows detailed studies such as JSC's risk assessment to be performed rapidly without labor intensive investigation of individual problem reports. In general, most Shuttle R&M related reports/studies do not include (or are not based on) R&M data which reflects all pertinent parameters that will significantly effect the actual failure rates. For example, actual equipment exposure time should be tracked rather than total mission time added to total recorded ground-power-on time to derive a generic total operating time for vehicle systems and equipment. Environmental characteristics during a mission and ground processing are radically different. These times and associated failure rates (and causes) should be tracked independently. Also, many systems, both mechanical and avionic, have actual operating times far different than total mission and ground power-on time (SSME,Propulsion, RCS & OMS, Flight Control, etc.). The need for actual tracked failure data reflecting all exposure time, compiled from all missions, such as contained in the USAF 66-1 military aircraft historical R&M data base, was discussed at a meeting held at LSOC during a recent LADC visit to KSC. In attendance was a JCWS representative, a LSOC R&M engineer and the LSOC Advanced Programs group. All agreed that a high priority should be placed on obtaining quality R&M data and discussed different means of effecting this desire. The final presentation for this study will highlight the O&S benefits of having such data.

Shuttle Orbiter maintenance repair times have been identified utilizing the Launch Vehicle Maintenance Analysis Final Report, dated November 1992, by Martin Marietta Manned Astronautics Group under contract to LaRC-NASA. This report derives maintenance Mean Time To Repair (MTTR) for Orbiter systems by identifying the duration (in days) which Problem Reports were statused as open and then adjusts the total time by a percent factor to reflect active time as opposed to inactive time. This is the only document found by LADC specifically addressing MTTR. Many scheduling reports exist which reflect total time predicted (or actual in the As-Run reports) for accomplishing top level maintenance tasks. Actual labor times and type maintenance are not recorded for the purpose of evaluating elements of each maintenance task performed during processing. The benefits of collecting this type data will be highlighted at the final presentation.

The source for aircraft R&M data is the USAF's large historical failure and maintenance data base. The AFM 66-1 Maintenance Data Collection System was originally devised as a tool for base level management. It was designed to assure effective management, at the base level, of all Air Force resources, tools, equipment, skills and manpower. Portions of the data collected were also furnished to the Air Force Logistics Command (AFLC) for logistics support requirements. As the potential of the system was recognized, its scope was enlarged and its use expanded. From its introduction in July 1958, the information has been used for cost analysis, budget computations, material improvement, training, improved maintenance, potential critical items, requirements, and many other areas including configuration management.

Today all failure and maintenance history for all existing military aircraft is collected at the component level. AFM 66-1 specifies the format for the raw data and the major R&M parameters derived from the data. Data is collected for all three defined levels of maintenance. These different levels of maintenance are addressed in task four of this report. Figure 1-1 depicts typical USAF base level aircraft maintenance cycle and the type data collected under AFM 66-1.

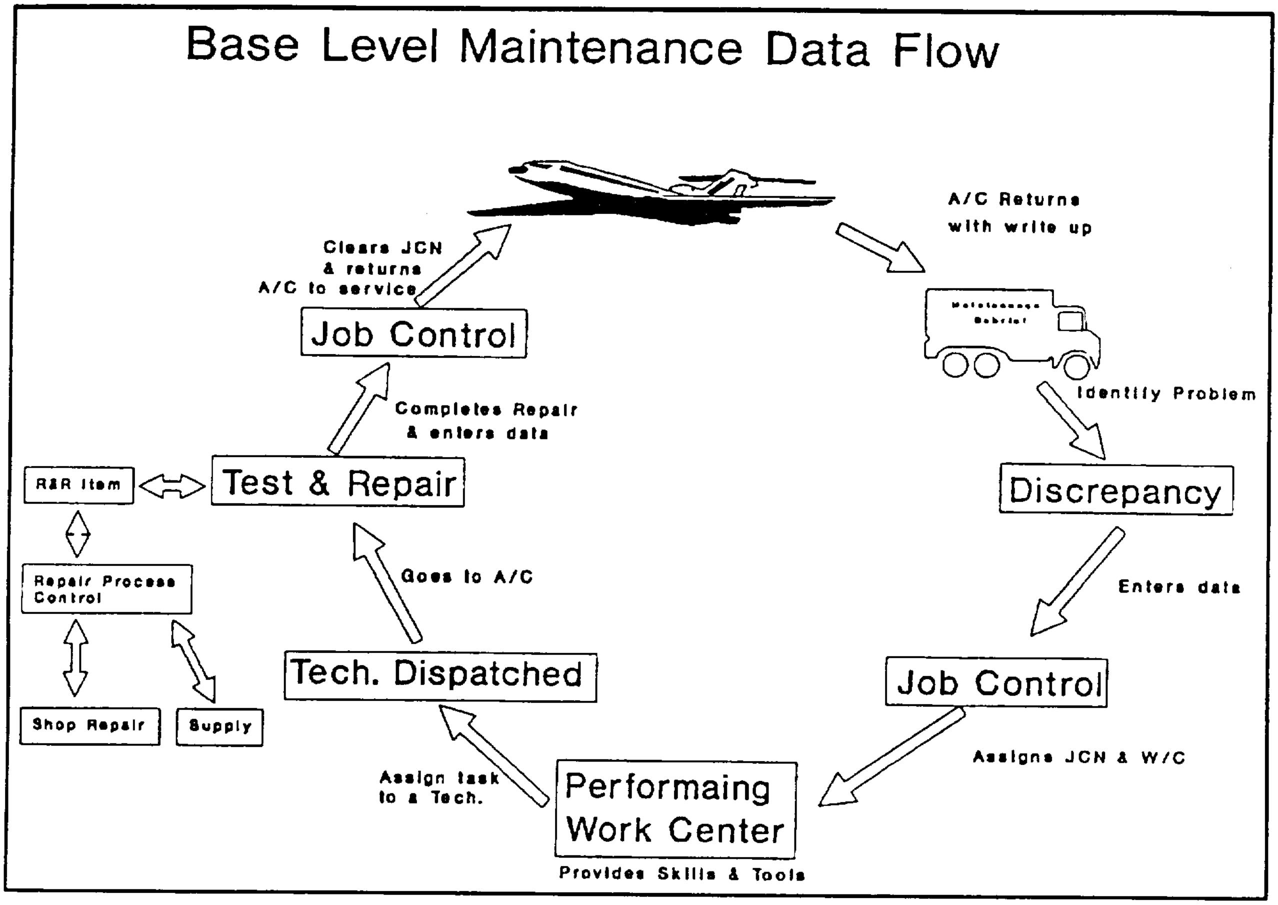



*Figure 1-1*

## 2.0 R&M ANALYSIS TASKS

### 2.1 TASK 1, ANALOGOUS AIRCRAFT SUBSYSTEM DEFINITION

Figure 2-1-1 is a listing of Shuttle Orbiter subsystems, their associated WUCs, MFSC's work breakdown structure, comparative USAF military aircraft type, systems and their associated WUCs. Selected analogous aircraft systems are generally similar in function, configuration, componentry and complexity. Aircraft systems WUCs are at the two-digit level unless Shuttle WUC break down require aircraft systems to be divided to lower level of indenture to allow a one to one comparison. Figures 2-1-2 through 2-1-6 are WUC listings for aircraft used in this study.

### 2.2 TASK 2, AIRCRAFT SUBSYSTEM FAILURE RATE IDENTIFICATION

Aircraft systems in bold print on the system listing in Task 1 are considered to have comparable physical or functional characteristics which would allow some degree of application to the next generation reusable space vehicle.

## ANALOGOUS AIRCRAFT SUBSYSTEMS

| ORBITER SYSTEMS | McCURRY'S WBS | STS WUC V | AIRCRAFT WUC A | BCS ANALYSIS AIRCRAFT | AIRCRAFT SYSTEMS |
|---|---|---|---|---|---|
| PRUGE, VENT & DRAIN. | 1.6.5 | V05 | 41 | E-3C | ENVIRONMENTAL CONTROL SYS |
| THERMAL PROTECTION | 1.7 | V09 | N/A | N/A | N/A |
| STRUCTURE | 1.5 | V10-35 | 11 | C-5A | STRUCTURE |
| MAIN PROPULSION SYS | 1.3.1-5 | V41 | 23 | F-15 | PROPULSION SYSTEM |
| REACTION CONTROL SYS | 1.4.2 | V42 | N/A | N/A | N/A |
| ORBITAL MANEUVERING SYS | 1.4.1 | V43 | N/A | N/A | N/A |
| ELECT POWER GENERATION | 1.2.1/2/3 | V45 | 42 | B-1B | ELECTRICAL POWER |
| AUXILIARY POWER UNIT | 1.6.3 | V46 | 24 | F-117 | AUXILIARY POWER UNIT |
| LANDING & DECELERATION | 1.6.6 | V51 | 13 | B-1B | LANDING GEAR |
| BRAKES & RECOVERY SYS | 1.6.6 | V52 | 13 | B-1B | LANDING GEAR |
| PAYLOAD & RETENTION / DEPLOY | 1.6.8 | V54 | 75 | X-XX | WEAPONS SYS |
| PYRO & RANGE SAFETY | 1.9 / 1.12 | V55 | 12 | B-1B | CREW EJECTION SYSTEM/PYRO |
| AERO SURFACE CONTROL (FLT) | 1.6.1 | V57 | 14K | F-16C | FLIGHT CONTROL SYS |
| HYDRAULIC POWER | 1.6.3 | V58 | 45 | B-1B | HYDRAULIC STSTEM |
| ACTUATION MECHANISM | 1.6.3 | V59 | 14K | F-16C | HYD / FLT CONT |
| ATMOSPHERIC REVITIALIZATION | 1.10.1.1 | V61 | 41 | E-3C | ENVIRONMENTAL CONTROL SYS |
| ACTIVE THERMAL CONTROL | 1.10.1.3 | V63 | 41 | N/A | N/A |
| NAVIGATION | 1.1.1 | V71 | 71,72,73 | B-1B | NAVIGATION |
| DATA PROCESSING | 1.1.4 | V72 | 82 | B-1B | DATA PROCESSING |
| COMMUNICATIONS | 1.1.3 | V74 | 67-69 | B-1B | COMMUNICATIONS |
| INSTRUMENTATION | 1.1.6/7/8 | V75 | 51-55 | C-5A | MAINTENANCE REPORTING |
| ELECTICAL POWER DIST | 1.1.2 | V76 | 42 | B-1B | ELECTRICAL SYS |
| FLIGHT CONTROL (AVIONICS) | 1.1.1.3 | V79 | 14A-J | F-16C | FLT CONT - AVIONICS |

*Figure 2-1-1*

Failure rates for analogous systems from aircraft selected in Task 1 are identified in Figure 2-2-1. Aircraft system inherent Mean Time Between Failures (MTBF) are converted to failures per hour for the purpose of comparison with Orbiter system failures. Failure rate data is only part of the information needed to develop a conceptual BCS math model which allows development of a turnaround prediction (Task 5). Maintenance data, Mean Time To Repair (MTTR), for each analogous aircraft subsystem is required to understand how long it will take to restore a system to operational status after a failure has been encountered. MTTR reflects all serial time required to 1) identify a failure (or requirement for maintenance), 2) isolate and access the failed component, 3) complete a repair or replacement action, 4) test component and/or system to insure integrity restoration, 5) close out area of vehicle in which maintenance was performed, 6) logistics delay time. The AFM 66-1 data collection system compiles data, independently, for each element of maintenance reflected in MTTR to allow rapid identification of excessive labor intensive elements. Figure 2-2-2 lists the analogous aircraft systems and their corresponding MTTRs.

LADC is currently developing a stand-alone PC based tool for performing rapid R&M analysis. This R&M Tool Kit automates the manual process of manipulating and analyzing raw R&M data and provides a graphic illustration for all typical aircraft R&M performance parameters (figures of merit). Currently, most raw AFM 66-1 historical data is available and data manipulation programs have been written and embedded the R&M Tool Kit. Graphic print programs are developed last and are not all currently available. An assortment of R&M Tool Kit data and graphics for the analogous aircraft systems are depicted in Figures 2-2-3 through 2-2-12.

# HIERARCHY OF SUBSYSTEM WORK UNIT CODES

**Aircraft : B001B**

| Wuc | Nomenclature |
|---|---|
| 00000 | AIRCRAFT LEVEL |
| 11000 | STRUCTURES |
| 12000 | EQUIPMENT/FURNISHN |
| 13000 | LANDING GEAR |
| 14000 | FLIGHT CONTROLS |
| 16000 | CREW ESCAPE/SAFETY |
| 19000 | ENGINE STARTING |
| 23000 | PROPULSION |
| 24000 | AUXILIARY PWR |
| 27000 | ACCESORY GEAR BOXE |
| 39000 | ICE AND RAIN PROTC |
| 41000 | AIR CONDITIONING |
| 42000 | ELECTRICAL POWER |
| 43000 | ELECTRICAL MULTIPL |
| 44000 | LIGHTING |
| 45000 | HYDRAULIC POWER |
| 46000 | FUEL |
| 47000 | OXYGEN |
| 48000 | INDICATING/RECORDN |
| 49000 | FIRE PROTECTION |
| 51000 | ELECT/ELEC, MULTPU |
| 52000 | AUTO FLIGHT |
| 55000 | CEN INTEG TEST SYS |
| 59000 | CREW COMMUNICATION |
| 73000 | NAVIGATION |
| 75000 | WEAPONS |
| 76000 | ELECTRONIC WARFARE |

*Figure 2–1–2*

# HIERARCHY OF SUBSYSTEM WORK UNIT CODES

Aircraft : C005A

| Wuc | Nomenclature |
|---|---|
| 00000 | AIRCRAFT LEVEL |
| 11000 | AIR FRAME |
| 12000 | CKPT & FUSLAG COMP |
| 13000 | LANDING GEAR |
| 14000 | FLIGHT CONTROLS |
| 23000 | TURBFAN PWR PLT SY |
| 24000 | AUX POWER PLANT AS |
| 41000 | AIR COND PRES&ICE |
| 42000 | ELECTRICAL PWR SUP |
| 44000 | LIGHTING SYSTEM |
| 45000 | HYD&PNEU PWR SUP |
| 46000 | FUEL SYSTEM |
| 47000 | OXYGEN SYSTEM |
| 49000 | MISC UTILITIES |
| 51000 | INSTRUMENTS |
| 52000 | AUTO PILOT |
| 55000 | MALFNCT DET ANLS/R |
| 61000 | HF COMMUNICATIONS |
| 62000 | VHF COMMUNICATION |
| 63000 | UHF COMMUNICATIONS |
| 64000 | INTERPHONE |
| 65000 | IFF |
| 66000 | EMER COMMUNICATION |
| 71000 | RADIO NAVIGATION |
| 72000 | RADAR NAVIGATION |
| 91000 | EMERGENCY EQUIP |
| 97000 | EXPLOSIVE DEV&COMP |

*Figure 2-1-3*

# HIERARCHY OF SUBSYSTEM WORK UNIT CODES

Aircraft : E003C

| Wuc | Nomenclature |
|---|---|
| 00000 | AIRCRAFT LEVEL |
| 03000 | LOOK PHASE SCH INS |
| 04000 | SPECIAL INSPECTION |
| 11000 | AIRFRAME |
| 12000 | COCKPIT/FUSELAGE |
| 13000 | LANDING GEAR |
| 14000 | FLT CONT SYS |
| 23000 | TURBOFAN POW PLT |
| 24000 | AUX POWER PLANT |
| 41000 | AIR COND PRESS |
| 42000 | ELECT POWER SUPPLY |
| 44000 | LIGHTING |
| 45000 | HYD PNEUMATIC POWE |
| 46000 | FUEL SYSTEM |
| 47000 | OXYGEN SYSTEM |
| 49000 | MISC UTILITIES |
| 51000 | INSTRUMENTS |
| 52000 | AUTOPILOT SYSTEM |
| 55000 | MALF ANAL REC EQUI |
| 61000 | HF COMM ANARC167 |
| 62000 | VHF COMM SYS |
| 63000 | UHF COM SYS ARC 16 |
| 64000 | INTERPHONE SYS |
| 65000 | IDENT FRIEND FOE |
| 66000 | EMERGENCY COMM SYS |
| 69000 | MISC COMM EQUIP |
| 71000 | RADIO NAVIGATION |
| 72000 | RADAR NAVIGATION |
| 76000 | ECM SYSTEM |
| 81000 | SURVEILLANCE RADAR |
| 82000 | COMPT DATA DISP SY |
| 91000 | EMERGENCY EQUIPMEN |
| 96000 | PERSONL MISC EQUIP |
| 97000 | EXP DEVICES EQUIP |

*Figure 2–1–4*

# HIERARCHY OF SUBSYSTEM WORK UNIT CODES

Aircraft : F015E

| Wuc | Nomenclature |
|---|---|
| 00000 | AIRCRAFT LEVEL |
| 03000 | LOOK PH OF SCH INS |
| 04000 | SPECIAL INSPECTION |
| 11000 | AIRFRAME |
| 12000 | CKPT & FUSE COMPTS |
| 13000 | LANDING GEAR |
| 14000 | FLIGHT CONTROLS |
| 23000 | POWER PLANT |
| 24000 | AUX PWR PLANT |
| 41000 | AIR COND P AN SI C |
| 42000 | ELECTRICAL SYSTEM |
| 44000 | LIGHTING SYSTEM |
| 45000 | HYD AN PNEU PWR SU |
| 46000 | FUEL SYSTEM |
| 47000 | OXYGEN SYSTEM |
| 49000 | MISC UTILITIES |
| 51000 | INSTRUMENTS |
| 52000 | AUTOPILOT |
| 55000 | MAL ANAL AN REC EQ |
| 57000 | INT GUID AN FLTCON |
| 63000 | UHF COMMUNICATIONS |
| 65000 | IFF |
| 71000 | RADIO NAVIGATION |
| 74000 | FIE CONTROL |
| 75000 | WPN DLVRY SYS |
| 76000 | TAC ELEC WRFRE SYS |
| 91000 | EMERG EQUIPMENT |
| 92000 | TOW TARGET EQUIPME |
| 97000 | EXPLOSIVE DEVICES |

*Figure 2–1–5*

# HIERARCHY OF SUBSYSTEM WORK UNIT CODES

**Aircraft : F016C**

| Wuc | Nomenclature |
|---|---|
| 00000 | AIRCRAFT LEVEL |
| 01000 | GROUND HANDLING SR |
| 02000 | AIRCRAFT CLEANING |
| 05000 | A/C ENGINE STORAGE |
| 06000 | GROUND SAFETY |
| 07000 | AIRCRAFT RECORDS |
| 09000 | SHOP SUPPORT GENER |
| 11000 | AIRFRAME |
| 12000 | CREW STATION SYSTE |
| 13000 | LANDING GEAR SYSTE |
| 14000 | FLIGHT CONTROL SYS |
| 23000 | TURBO FAN PWR PLAN |
| 24000 | AUX POWER PLANT JF |
| 27000 | TURBOFAN POWR PLAN |
| 41000 | ENVIR CONT SYSTEM |
| 42000 | ELECT POWER SYSTEM |
| 44000 | LIGHTING SYSTEM |
| 45000 | HYD AND PNEU SYSTE |
| 46000 | FUEL SYSTEM |
| 47000 | OXYGEN SYSTEM |
| 49000 | MISCELLAN UTILITIE |
| 51000 | FLIGHT INSTRUMENTS |
| 55000 | MALFCT ANLYS REC E |
| 62000 | VHF COMMUNICATIONS |
| 63000 | UHF COMMUNICATIONS |
| 64000 | INTERPHONE SYSTEM |
| 65000 | IFF SYSTEM |
| 69000 | MISC COMM EQUIP |
| 71000 | RADIO NAVIGATION |
| 74000 | FIRE CONTROL SYSTE |
| 75000 | WEAPONS DELIVERY |
| 76000 | PENETR AIDS AND EC |
| 91000 | EMERGENCY EQUIPMEN |
| 92000 | TOW TARGET EQUIP |
| 96000 | PERS AND MISC EQUI |
| 97000 | EXPLSVE DVCS / CMP |

*Figure 2–1–6*

## ORBITER AND AIRCRAFT FAILURE DATA COMPARISON

| ORBITER SYSTEMS | WBS | PR/100HR | F/100HR | AIRCRAFT SYSTEMS |
|---|---|---|---|---|
| PRUGE, VENT & DRAIN. | 1.6.5 | 0.14 | .578 | ENVIRONMENTAL CONTROL SY |
| THERMAL PROTECTION | 1.7 | 54.00 | | N/A |
| STRUCTURE | 1.5 | 11.99 | 1.075 | STRUCTURE |
| MAIN PROPULSION SYS | 1.3.1-5 | 3.19 | 6.854 | PROPULSION SYSTEM |
| REACTION CONTROL SYS | 1.4.2 | 0.23 | | N/A |
| ORBITAL MANEUVERING SYS | 1.4.1 | 0.17 | | N/A |
| ELECT POWER GENERATION | 1.2.1/2/3 | 0.57 | .644 | ELECTRICAL POWER |
| AUXILIARY POWER UNIT | 1.6.3 | 1.66 | 1.502 | AUXILIARY POWER UNIT |
| LANDING & DECELERATION | 1.6.6 | 0.81 | 5.672 | LANDING GEAR |
| BRAKES & RECOVERY SYS | 1.6.6 | 0.05 | 3.979 | LANDING GEAR |
| PAYLOAD & RETENTION / DEPLOY | 1.6.8 | 0.26 | 1.258 | WEAPONS SYS |
| PYRO & RANGE SAFETY | 1.9 / 1.12 | 0.19 | .100 | CREW EJETION SYSTEM |
| AERO SURFACE CONTROL (FLT) | 1.6.1 | 0.02 | .140 | FLIGHT CONTROL SYS |
| HYDRAULIC POWER | 1.6.3 | 1.46 | 4.168 | HYDRAULIC STSTEM |
| ACTUATION MECHANISM | 1.6.3 | 0.61 | .026 | HYD / FLT CONT ACT |
| ATMOSPHERIC REVITIALIZATION | 1.10.1.1 | 0.38 | 1.640 | ENVIRONMENTAL CONTROL SY |
| ACTIVE THERMAL CONTROL | 1.10.1.3 | 2.74 | | N/A |
| NAVIGATION | 1.1.1 | 0.07 | 16.181 | NAVIGATION |
| DATA PROCESSING | 1.1.4 | 0.24 | 3.322 | DATA PROCESSING |
| COMMUNICATIONS | 1.1.3 | 0.86 | 1.659 | COMMUNICATIONS |
| INSTRUMENTATION | 1.1.6/7/8 | 0.66 | 3.140 | MAINTENANCE REPORTING |
| ELECTICAL POWER DIST | 1.1.2 | 0.26 | .494 | ELECTRICAL SYS |
| FLIGHT CONTROL (AVIONICS) | 1.1.1.3 | 0.09 | .285 | FLT CONT - AVIONICS |

*Figure 2–2–1*

## ORBITER AND AIRCRAFT MAINTENANCE DATA COMPARISON

| ORBITER SYSTEMS | WBS | MTTR | MTTR | AIRCRAFT SYSTEMS |
|---|---|---|---|---|
| PRUGE, VENT & DRAIN. | 1.6.5 | 16.20 | 0.72 | ENVIRONMENTAL CONTROL SYS |
| THERMAL PROTECTION | 1.7 | 35.54 | N/A | N/A |
| STRUCTURE | 1.5 | 5.72 | 1.02 | STRUCTURE |
| MAIN PROPULSION SYS | 1.3.1-5 | 13.04 | 1.06 | PROPULSION SYSTEM |
| REACTION CONTROL SYS | 1.4.2 | 15.17 | N/A | N/A |
| ORBITAL MANEUVERING SYS | 1.4.1 | 23.01 | N/A | N/A |
| ELECT POWER GENERATION | 1.2.1/2/3 | 27.33 | 1.05 | ELECTRICAL POWER |
| AUXILIARY POWER UNIT | 1.6.3 | 15.17 | 2.7 | AUXILIARY POWER UNIT |
| LANDING & DECELERATION | 1.6.6 | 10.00 | 0.99 | LANDING GEAR |
| BRAKES & RECOVERY SYS | 1.6.6 | 10.00 | 0.93 | LANDING GEAR |
| PAYLOAD & RETENTION / DEPLOY | 1.6.8 | 10.00 | 2.84 | WEAPONS SYS |
| PYRO & RANGE SAFETY | 1.9 / 1.12 | 58.84 | 2.5 | CREW EJECTION SYSTEM |
| AERO SURFACE CONTROL (FLT) | 1.6.1 | 10.00 | 1.24 | FLIGHT CONTROL SYS |
| HYDRAULIC POWER | 1.6.3 | 15.25 | 1.04 | HYDRAULIC STSTEM |
| ACTUATION MECHANISM | 1.6.3 | 77.52 | 1.58 | HYD / FLT CONT |
| ATMOSPHERIC REVITIALIZATION | 1.10.1.1 | 44.19 | 0.72 | ENVIRONMENTAL CONTROL SYS |
| ACTIVE THERMAL CONTROL | 1.10.1.3 | 7.54 | N/A | N/A |
| NAVIGATION | 1.1.1 | 77.52 | 1.03 | NAVIGATION |
| DATA PROCESSING | 1.1.4 | 17.10 | 2.55 | DATA PROCESSING |
| COMMUNICATIONS | 1.1.3 | 17.10 | 1 | COMMUNICATIONS |
| INSTRUMENTATION | 1.1.6/7/8 | 44.19 | 0.9 | MAINTENANCE REPORTING |
| ELECTICAL POWER DIST | 1.1.2 | 7.54 | 0.77 | ELECTRICAL SYS |
| FLIGHT CONTROL (AVIONICS) | 1.1.1.3 | 17.10 | 0.88 | FLT CONT - AVIONICS |

*Figure 2–2–2*

*R&M TOOL KIT PRINTOUT*

# SUMMARIZED MDC DATA

Aircraft : E003C Fh : 3,476.00 Sorties : 368.00

| Year | Ind | Wuc | Nomenclature | Inherent | Induced | Nodefect | Total_hrs |
|---|---|---|---|---|---|---|---|
| 91 | 2 | 11000 | AIRFRAME | 453.00 | 291.50 | 146.00 | 890.50 |
| 91 | 2 | 12000 | COCKPIT/FUSELAGE | 228.42 | 177.50 | 227.50 | 633.42 |
| 91 | 2 | 13000 | LANDING GEAR | 137.50 | 141.00 | 52.33 | 330.83 |
| 91 | 2 | 14000 | FLT CONT SYS | 262.67 | 225.67 | 231.67 | 720.00 |
| 91 | 2 | 23000 | TURBOFAN POW PLT | 460.92 | 323.67 | 456.33 | 1,240.92 |
| 91 | 2 | 24000 | AUX POWER PLANT | 69.92 | 41.00 | 30.00 | 140.92 |
| 91 | 2 | 41000 | AIR COND PRESS | 514.75 | 554.67 | 574.17 | 1,643.58 |
| 91 | 2 | 42000 | ELECT POWER SUPPLY | 217.17 | 159.83 | 210.00 | 587.00 |
| 91 | 2 | 44000 | LIGHTING | 93.58 | 83.25 | 58.33 | 235.17 |
| 91 | 2 | 45000 | HYD PNEUMATIC POWE | 120.08 | 97.42 | 105.67 | 323.17 |
| 91 | 2 | 46000 | FUEL SYSTEM | 74.83 | 115.92 | 92.58 | 283.33 |
| 91 | 2 | 47000 | OXYGEN SYSTEM | 96.00 | 48.92 | 124.50 | 269.42 |
| 91 | 2 | 49000 | MISC UTILITIES | 37.50 | 63.42 | 41.00 | 141.92 |
| 91 | 2 | 51000 | INSTRUMENTS | 78.08 | 79.00 | 134.17 | 291.25 |
| 91 | 2 | 52000 | AUTOPILOT SYSTEM | 64.00 | 50.00 | 66.67 | 180.67 |
| 91 | 2 | 55000 | MALF ANAL REC EQUI | 33.50 | 27.00 | 27.00 | 87.50 |
| 91 | 2 | 61000 | HF COMM ANARC167 | 241.17 | 188.00 | 167.75 | 596.92 |
| 91 | 2 | 62000 | VHF COMM SYS | 11.00 | 11.00 | 10.00 | 32.00 |
| 91 | 2 | 63000 | UHF COM SYS ARC 16 | 229.67 | 165.50 | 195.25 | 590.42 |
| 91 | 2 | 64000 | INTERPHONE SYS | 160.08 | 178.00 | 135.25 | 473.33 |
| 91 | 2 | 65000 | IDENT FRIEND FOE | 70.17 | 76.67 | 78.17 | 225.00 |
| 91 | 2 | 66000 | EMERGENCY COMM SYS | 8.00 | 26.00 | 16.00 | 50.00 |
| 91 | 2 | 69000 | MISC COMM EQUIP | 348.42 | 327.08 | 308.75 | 984.25 |
| 91 | 2 | 71000 | RADIO NAVIGATION | 113.67 | 93.00 | 128.25 | 334.92 |
| 91 | 2 | 72000 | RADAR NAVIGATION | 25.00 | 66.00 | 24.17 | 115.17 |
| 91 | 2 | 76000 | ECM SYSTEM | 91.17 | 89.50 | 39.50 | 220.17 |
| 91 | 2 | 81000 | SURVEILLANCE RADAR | 619.25 | 292.17 | 365.92 | 1,277.33 |
| 91 | 2 | 82000 | COMPT DATA DISP SY | 374.92 | 276.00 | 340.17 | 991.08 |
| 91 | 2 | 91000 | EMERGENCY EQUIPMEN | 9.50 | 9.00 | 14.00 | 32.50 |
| 91 | 2 | 96000 | PERSONL MISC EQUIP | 2.50 | 4.50 | 10.00 | 17.00 |
| 91 | 2 | 97000 | EXP DEVICES EQUIP | 0.50 | 0.25 | 2.00 | 2.75 |

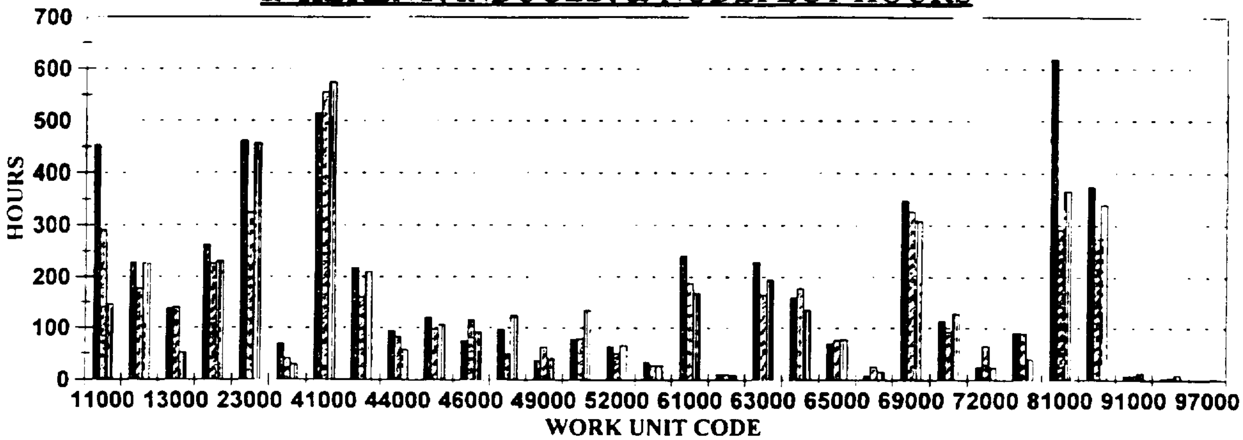
*Figure 2-2-3*

*R&M TOOL KIT PRINTOUT*

# SUMMARIZED MDC DATA

Aircraft : E003C Fh : 3,476.00 Sorties : 368.00

## INHERENT, INDUCED, & NODEFECT HOURS




## BREAK RATE DATA

| Aircraft | Sorties | Break Rate |
|---|---|---|
| E003A | 138.00 | 0.96 |
| E003B | 801.00 | 2.77 |
| E003C | 368.00 | 3.61 |

AIRCRAFT LEVEL BREAK RATES

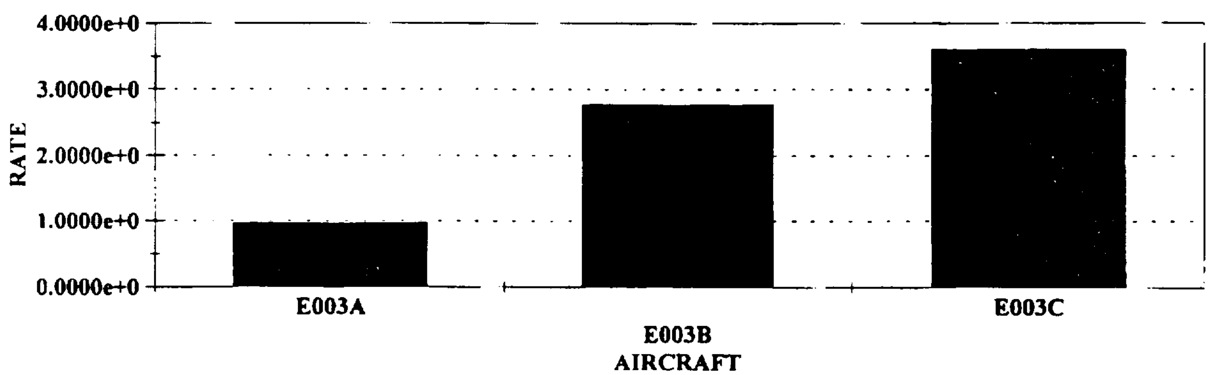


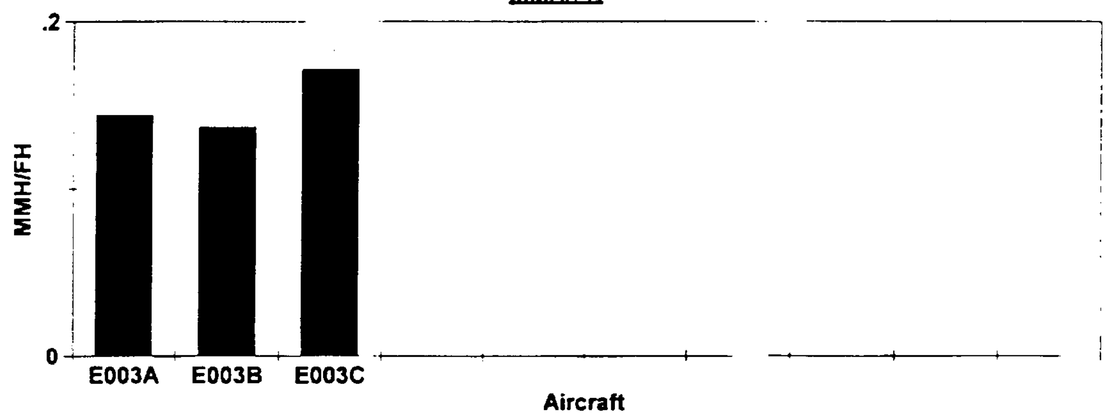

*Figure 2–2–4*

*R&M TOOL KIT PRINTOUT*

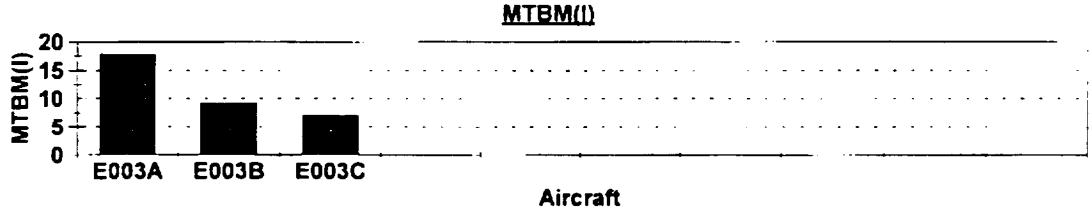
# MANHOURS AND FAILURES BY AIRCRAFT

**41000 AIR CONDITIONING AND PRESSURIZATION**

**MANHOUR UTILIZATION**
**MMH/FH**




**INHERENT FAILURE COUNTS**
**MTBM(I)**




# MDC MAINTENANCE TASK BREAKDOWN

Today : 5/26/94

Aircraft : E003C Flight Hours : 3,476.00

| Wuc | Year | Ind | Open* | Troubleshoot | Repair | Checkout | Close |
|---|---|---|---|---|---|---|---|
| 41000 | 91 | 2 | 0.09 | 0.39 | 0.24 | 0.19 | 0.09 |

*Figure 2-2-5*

*R&M TOOL KIT PRINTOUT*

# HOW MAL BREAKDOWN

**Aircraft : E003B**

**Selection WUC : 13**

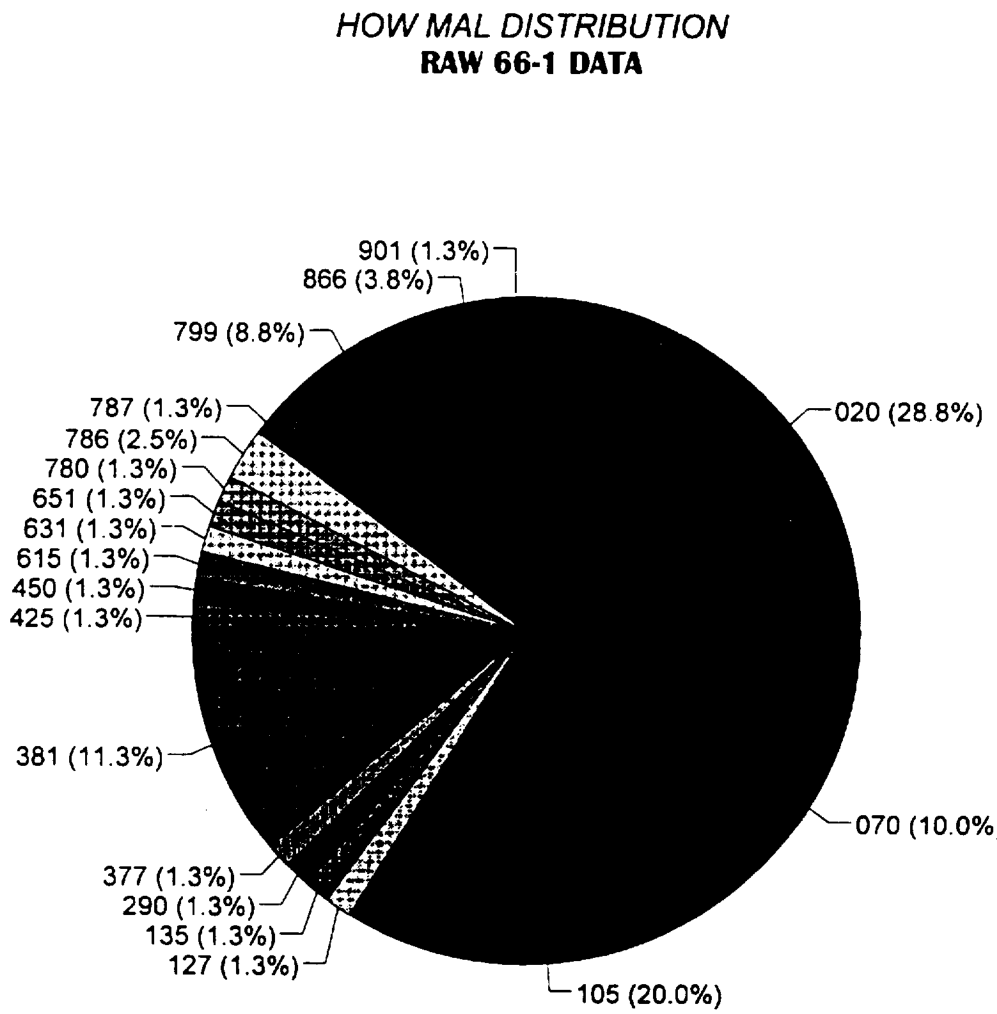


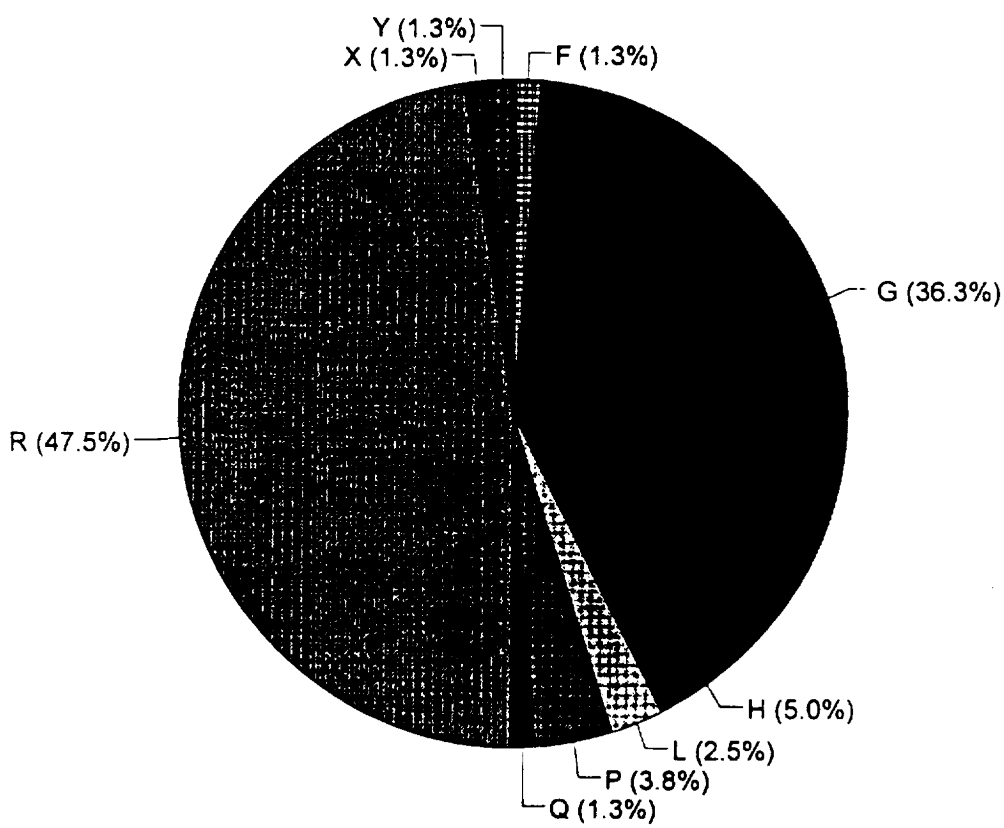

*Figure 2-2-6*

# ACTION TAKEN BREAKDOWN

**Aircraft : E003B**

**Selection WUC : 13**

*ACTION TAKEN TOTAL DISTRIBUTION*
**RAW 66-1 DATA**




**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

*Figure 2–2–7*

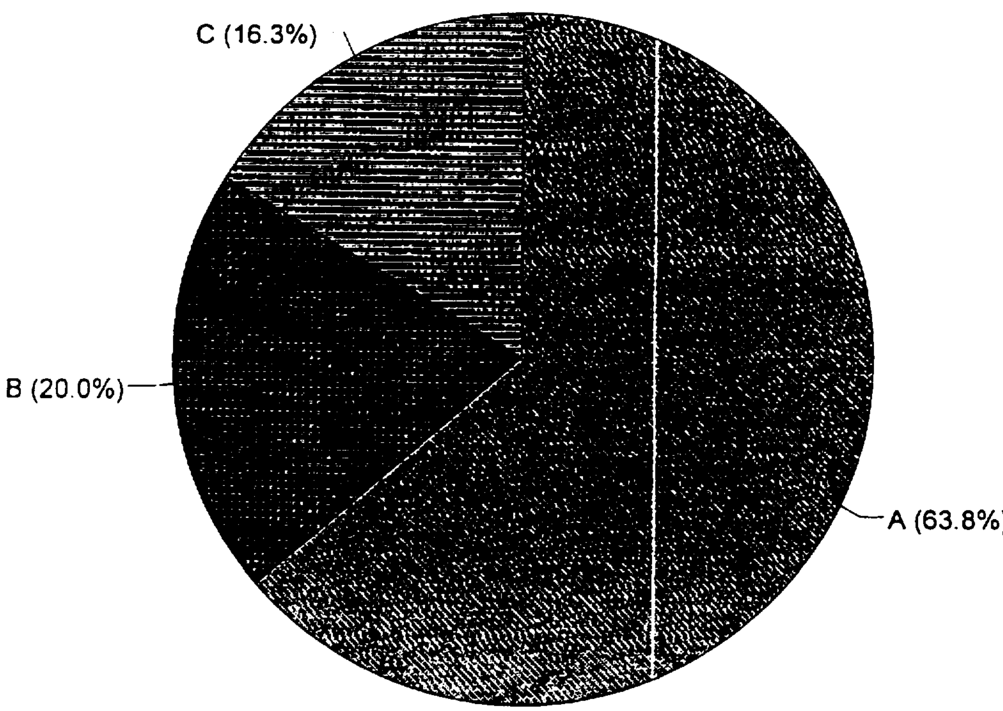
*R&M TOOL KIT PRINTOUT*

# TYPE HOW MAL BREAKDOWN

**Aircraft : E003B**

**Selection WUC : 13**

*TOTAL TYPE HOW MAL DISTRIBUTION*
**RAW 66-1 DATA**



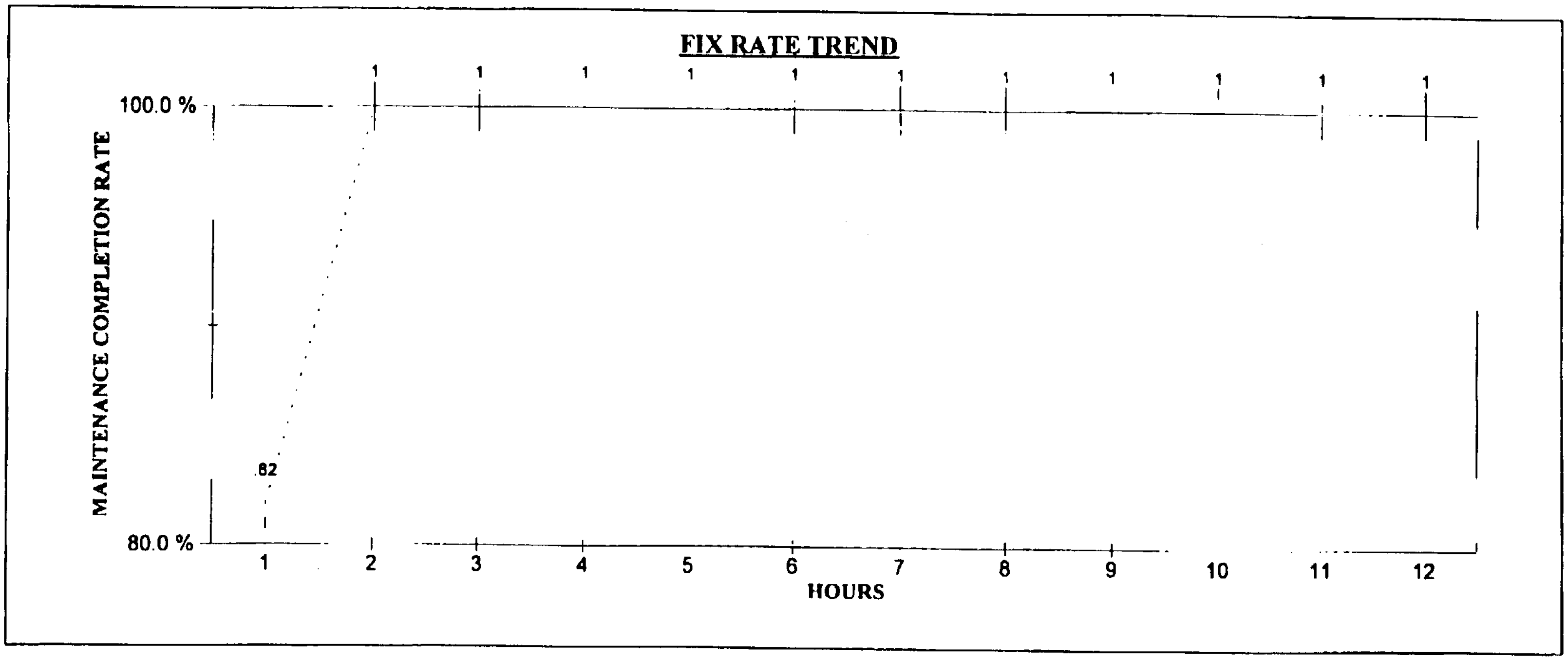

**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

*Figure 2–2–8*

# *R&M TOOL KIT PRINTOUT*

## FIX RATE CHART

Aircraft : **E003C** Wuc : **76000** Nomenclature : **ECM SYSTEM**




**CURRENT DATA**

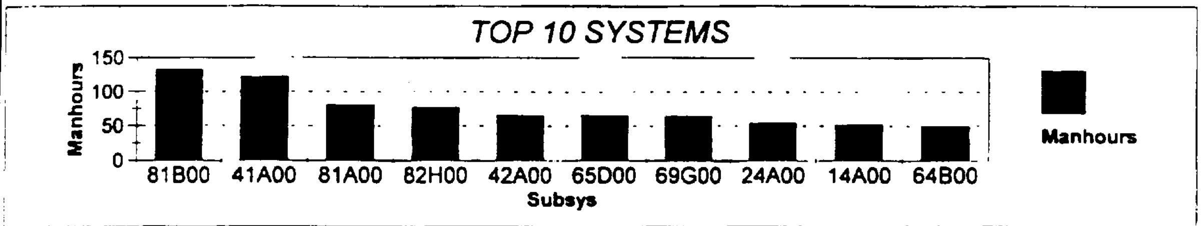
| Hour | Percent | Events |
|---|---|---|
| 1.00 | 0.82 | 18.00 |
| 2.00 | 1.00 | 4.00 |
| 3.00 | 1.00 | 0.00 |
| 4.00 | 1.00 | 0.00 |
| 5.00 | 1.00 | 0.00 |
| 6.00 | 1.00 | 0.00 |
| 7.00 | 1.00 | 0.00 |
| 8.00 | 1.00 | 0.00 |
| 9.00 | 1.00 | 0.00 |
| 10.00 | 1.00 | 0.00 |
| 11.00 | 1.00 | 0.00 |
| 12.00 | 1.00 | 0.00 |

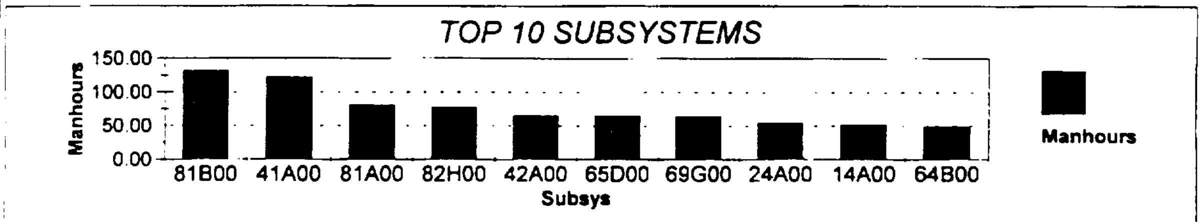
*Figure 2–2–9*

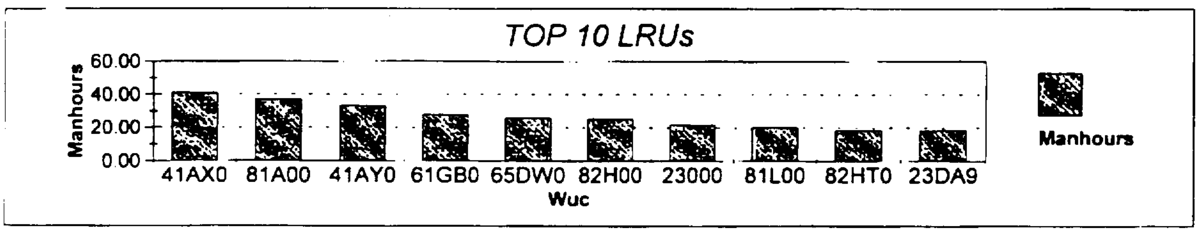
## *R&M TOOL KIT PRINTOUT*

# TOP 10 DRIVERS

# SYSTEM, SUBSYSTEM, AND LRU



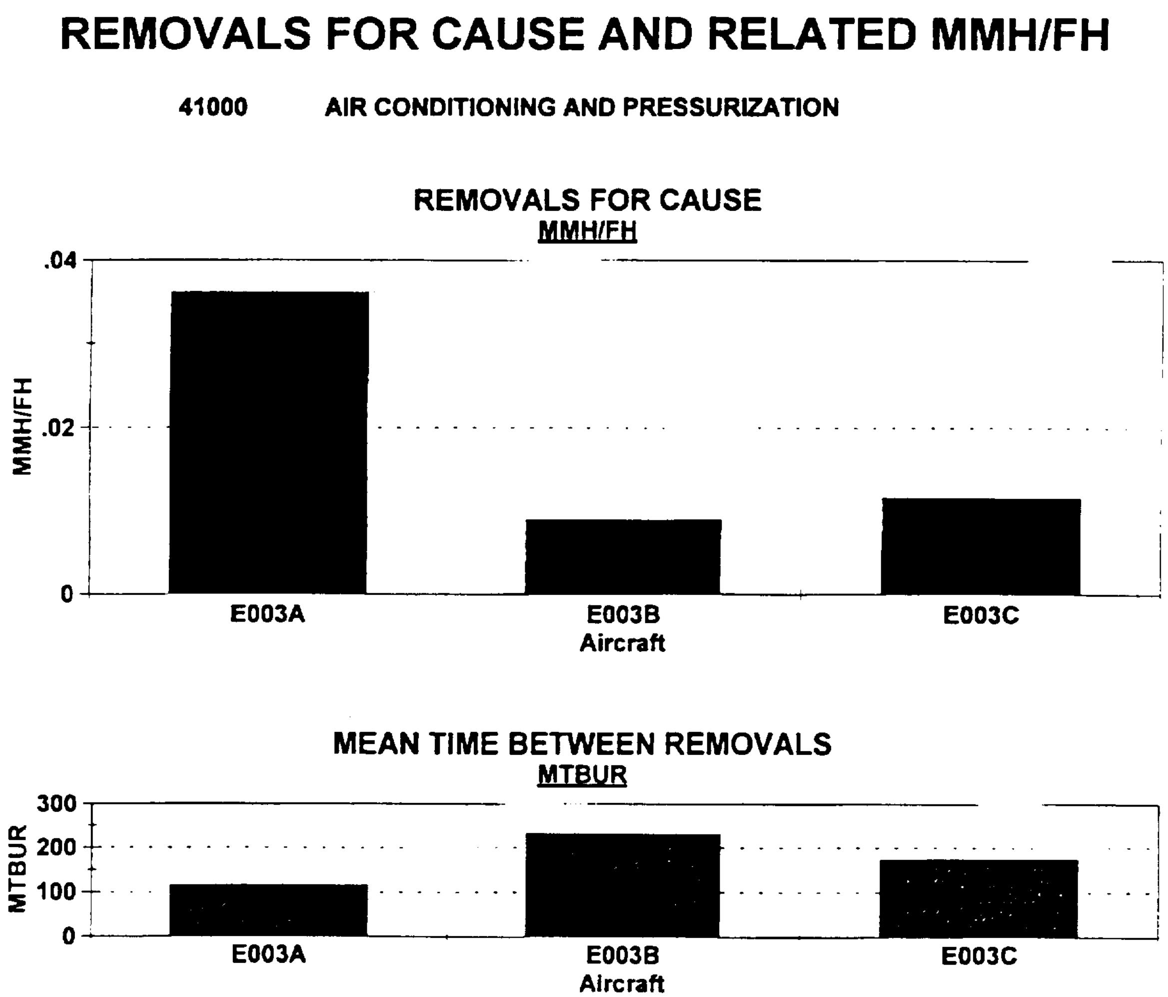

| Flight Hrs | Sorties |
|---|---|
| 3,476.00 | 368.00 |

| Aircraft | Sys | Units | Manhours |
|---|---|---|---|
| E003C | 81B00 | 67.00 | 132.08 |
| E003C | 41A00 | 66.00 | 122.00 |
| E003C | 81A00 | 41.00 | 80.50 |
| E003C | 82H00 | 85.00 | 77.08 |
| E003C | 42A00 | 49.00 | 65.67 |
| E003C | 65D00 | 26.00 | 65.25 |
| E003C | 69G00 | 32.00 | 63.92 |
| E003C | 24A00 | 22.00 | 54.25 |
| E003C | 14A00 | 37.00 | 51.92 |
| E003C | 64B00 | 34.00 | 49.50 |

| Subsys | Units | Manhours |
|---|---|---|
| 81B00 | 67.00 | 132.08 |
| 41A00 | 66.00 | 122.00 |
| 81A00 | 41.00 | 80.50 |
| 82H00 | 85.00 | 77.08 |
| 42A00 | 49.00 | 65.67 |
| 65D00 | 26.00 | 65.25 |
| 69G00 | 32.00 | 63.92 |
| 24A00 | 22.00 | 54.25 |
| 14A00 | 37.00 | 51.92 |
| 64B00 | 34.00 | 49.50 |

| Wuc | Units | Manhours |
|---|---|---|
| 41AX0 | 27.00 | 40.92 |
| 81A00 | 9.00 | 37.00 |
| 41AY0 | 12.00 | 33.17 |
| 61GB0 | 9.00 | 27.92 |
| 65DW0 | 6.00 | 25.75 |
| 82H00 | 13.00 | 25.25 |
| 23000 | 7.00 | 21.67 |
| 81L00 | 7.00 | 20.25 |
| 82HT0 | 7.00 | 18.67 |
| 23DA9 | 6.00 | 18.50 |

*Figure 2–2–10*



REMOVALS FOR CAUSE AND RELATED MMH/FH
41000 AIR CONDITIONING AND PRESSURIZATION
REMOVALS FOR CAUSE
MMH/FH
MMH/FH
.04
.02
0
E003A
E003B
E003C
Aircraft
MEAN TIME BETWEEN REMOVALS
MTBUR
MTBUR
300
200
100
0
E003A
E003B
E003C
Aircraft


*Figure 2-2-11*

## *R&M TOOL KIT PRINTOUT*

## GENERAL R&M PARAMETER REPORT

Aircraft : E003C Flight Hours : 3,476.00 Sorties : 368.00

| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 12000 | COCKPIT/FUSELAGE | 66,455.70 | 6.65 | 0.46 | 434.50 | 82.76 | 15.05 | 119.86 |
| 13000 | LANDING GEAR | 33,659.38 | 3.37 | 0.76 | 347.60 | 139.04 | 29.71 | 133.69 |
| 14000 | FLT CONT SYS | 69,332.57 | 6.93 | 0.48 | 1,738.00 | 289.67 | 14.42 | 56.06 |
| 23000 | TURBOFAN POW PLT | 69,044.88 | 6.90 | 0.90 | 267.38 | 49.66 | 14.48 | 70.94 |
| 24000 | AUX POWER PLANT | 7,479.86 | 0.75 | 1.30 | 579.33 | 217.25 | 133.69 | 248.29 |
| 41000 | AIR COND PRESS | 141,829.69 | 14.18 | 0.72 | 204.47 | 43.45 | 7.05 | 60.98 |
| 42000 | ELECT POWER SUPPLY | 47,468.35 | 4.75 | 0.67 | 869.00 | 112.13 | 21.07 | 217.25 |
| 44000 | LIGHTING | 20,138.09 | 2.01 | 0.68 | 496.57 | 79.00 | 49.66 | 204.47 |
| 45000 | HYD PNEUMATIC POWE | 23,590.33 | 2.36 | 0.77 | 1,738.00 | 217.25 | 42.39 | 173.80 |
| 46000 | FUEL SYSTEM | 18,699.65 | 1.87 | 0.81 | 1,158.67 | 124.14 | 53.48 | 112.13 |
| 47000 | OXYGEN SYSTEM | 11,795.17 | 1.18 | 0.72 | 869.00 | 289.67 | 84.78 | 1,158.67 |
| 49000 | MISC UTILITIES | 7,192.17 | 0.72 | 0.71 | 3,476.00 | 496.57 | 139.04 | 869.00 |
| 51000 | INSTRUMENTS | 16,110.47 | 1.61 | 0.74 | 158.00 | 70.94 | 62.07 | 80.84 |
| 52000 | AUTOPILOT SYSTEM | 8,630.61 | 0.86 | 0.85 | 579.33 | 158.00 | 115.87 | 182.95 |
| 55000 | MALF ANAL REC EQUI | 7,767.55 | 0.78 | 0.80 | | 173.80 | 128.74 | 173.80 |
| 61000 | HF COMM ANARC167 | 33,659.38 | 3.37 | 0.78 | 119.86 | 54.31 | 29.71 | 57.93 |
| 62000 | VHF COMM SYS | 3,164.56 | 0.32 | 0.49 | 1,158.67 | 496.57 | 316.00 | 496.57 |
| 63000 | UHF COM SYS ARC 16 | 45,166.86 | 4.52 | 0.74 | 72.42 | 29.97 | 22.14 | 32.79 |
| 64000 | INTERPHONE SYS | 39,988.49 | 4.00 | 0.60 | 96.56 | 31.60 | 25.01 | 44.00 |
| 65000 | IDENT FRIEND FOE | 14,096.66 | 1.41 | 0.87 | 695.20 | 173.80 | 70.94 | 204.47 |
| 66000 | EMERGENCY COMM SYS | 2,301.50 | 0.23 | 0.66 | 1,158.67 | 869.00 | 434.50 | 1,158.67 |
| 69000 | MISC COMM EQUIP | 70,195.63 | 7.02 | 0.72 | 108.63 | 22.72 | 14.25 | 26.33 |
| 71000 | RADIO NAVIGATION | 22,151.90 | 2.22 | 0.78 | 158.00 | 75.57 | 45.14 | 72.42 |
| 72000 | RADAR NAVIGATION | 5,178.37 | 0.52 | 0.67 | 1,738.00 | 496.57 | 193.11 | 347.60 |
| 76000 | ECM SYSTEM | 10,932.11 | 1.09 | 0.79 | 579.33 | 128.74 | 91.47 | 158.00 |
| 81000 | SURVEILLANCE RADAR | 102,416.57 | 10.24 | 1.10 | 182.95 | 68.16 | 9.76 | 47.62 |
| 82000 | COMPT DATA DISP SY | 100,690.45 | 10.07 | 0.65 | 57.93 | 15.52 | 9.93 | 13.85 |
| 91000 | EMERGENCY EQUIPMEN | 2,013.81 | 0.20 | 1.10 | | 1,158.67 | 496.57 | 3,476.00 |
| 96000 | PERSONL MISC EQUIP | 1,438.43 | 0.14 | 0.40 | | 695.20 | 695.20 | 1,738.00 |
| 97000 | EXP DEVICES EQUIP | 287.69 | 0.03 | 0.25 | | | 3,476.00 | |
| 97000 | EXP DEVICES EQUIP | 287.69 | 0.03 | 0.25 | | | 3,476.00 | |

*Figure 2–2–12*

## 2.3 TASK 3, COMPARISON OF SHUTTLE AND AIRCRAFT FAILURE RATES

A one-to-one comparison of Orbiter and aircraft failure rates and maintenance repair times is contained in Task 2. Task 3 develops a Baseline Comparative System math model which combines these two R&M parameters in order to identify total system-driven turnaround time requirements for MFSC's SSTO vehicle. The combination of system failure rates and associated repair times yields the total burden for corrective maintenance actions. Additionally, estimates (in the form of percentage of corrective maintenance time) are included for scheduled (preventative maintenance) and summed to yield total system-driven maintenance hours required. In addition to vehicle system driven maintenance, functional processing tasks contribute to the total time required for turnaround. Elements of functional requirements include recovery, postflight inspections and tests, servicing and check out, pad integration, and interface tests. Payload installation / removal and integration also falls in the functional category but, is not addressed in this assessment.

Three different BCS math models were developed for this study. The first model reflects Shuttle Orbiter systems identified for the SSTO vehicle. The mission duration is typical of an average Orbiter mission which yields 168 hours of mission exposure time on the vehicle. For a subset of the Orbiter systems, ground power-on time provides a much greater exposure time. This is also reflected in the model as this operation time and associated failures (problem report quantity) is considered in the failure rates contained in the RCM reports. An average ground power-on time of 2000 hours is used. A factor of 50 percent is added to corrective maintenance to reflect the scheduled maintenance burden. Shuttle system failure rates and maintenance times are used to provide corrective maintenance burdens. Each vehicle system is evaluated to produce a system total and then summed to provide a vehicle total maintenance hour requirement. This is added to a SSTO-assumed functional maintenance time requirement to produce the total processing maintenance hour requirement.

The purpose of the first model is to establish a unmanned SSTO base line which reflects current Shuttle failure rates, repair times, ground power-on times and mission times. A long established spacecraft community culture and vehicle maintenance approach is reflected within the data used for this model. The model produced a total maintenance hour burden between missions of approximately 50,000 hours. If the TPS system is ignored the total system burden is only approximately 8,000 hours. A print out of the model is shown in Figure 2-3-1.

The second model developed is identical to the first except the SSTO mission time is reduced to an assumed time of 24 hours. The ground power-on time is reduced to a predicted level of 250 hours.

This yielded a total between-mission processing maintenance time of approximately 7000 hours. The scope of this contract does not allow the investigation of individual system drivers and identification of which phase of the mission or ground processing failures are identified. This is critical for establishing assumptions included in the model which accurately reflect the affect of mission and ground processing time changes on each vehicle system. TPS, Propulsion Feed, Main Engine, RCS and OMS systems yield the most obvious illogical mathematical results. These systems are prime candidates for a R&M investigation to allow more accurate modeling of their failure characteristics. A print out of this model is shown in Figure 2-3-2.

The third model was developed to best reflect the next generation reusable space vehicle. Orbiter system failure rates are used because they best reflect the mission environment. Aircraft mainte-

# ORBITER AND AIRCRAFT FAILURE DATA COMPARISON

| | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| GROUND PROCESSING TIME= | 2000 | HOURS | | | | (PR count = FAIL count) | | | | |
| STS MISSION DURATION= | 168 | HOURS | | JCWSI RCM DATA ( STS61-3 FLT RATE ) EQUIPMENT FAILURE RATE | | | | | MARTIN'S LVMAFR | |
| SYSTEMS & FUNCTIONS | WBS | WUC V | WUC A | PRs / 100 OP HRS | PRs/HR | STS MISSION FAILS / MISSION | STS GROUND FAILS / MISSION | STS TOTAL FAILS / MISSION | STS SYSTEM MTTR | TOTAL BURDEN HOURS |
| PRUGE, VENT & DRAIN. | 1.6.5 | V05 | 41 | 0.14 | 0.0014 | .235 | 2.8 | 3.04 | 16.20 | 49.16 |
| THERMAL PROTECTION | 1.7 | V09 | N/A | 54.00 | 0.54 | *90.720* | *1080* | *1170.72* | 35.54 | 41605.81 |
| STRUCTURE | 1.5 | V10-35 | 11 | 11.99 | 0.1199 | 20.143 | 239.8 | 259.94 | 5.72 | 1486.59 |
| MAIN PROPULSION SYS | 1.3.1-5 | V41 | 23 | 3.19 | 0.0319 | 5.359 | 63.8 | 138.32 | 13.04 | 1803.31 |
| REACTION CONTROL SYS | 1.4.2 | V42 | N/A | 0.23 | 0.0023 | .386 | 4.6 | 4.99 | 15.17 | 75.63 |
| ORBITAL MANEUVERING SYS | 1.4.1 | V43 | N/A | 0.17 | 0.0017 | .286 | 3.4 | 3.69 | 23.01 | 84.80 |
| ELECT POWER GENERATION | 1.2.1/2/3 | V45 | 42 | 0.57 | 0.0057 | .958 | 11.4 | 12.36 | 27.33 | 337.68 |
| AUXILIARY POWER UNIT | 1.6.3 | V46 | 24 | 1.66 | 0.0166 | 2.789 | 33.2 | 35.99 | 15.17 | 545.88 |
| LANDING & DECELERATION | 1.6.6 | V51 | 13 | 0.81 | 0.0081 | 1.361 | 16.2 | 17.56 | 10.00 | 175.61 |
| BRAKES & RECOVERY SYS | 1.6.6 | V52 | 13 | 0.05 | 0.0005 | .084 | 1 | 1.08 | 10.00 | 10.84 |
| PAYLOAD & RETENTION / DEPLOY | 1.6.8 | V54 | 75 | 0.26 | 0.0026 | .437 | 5.2 | 5.64 | 10.00 | 56.37 |
| PYRO & RANGE SAFETY | 1.9 / 1.12 | V55 | 12 | 0.19 | 0.0019 | .319 | 3.8 | 4.12 | 58.84 | 242.37 |
| AERO SURFACE CONTROL (FLT) | 1.6.1 | V57 | 14K | 0.02 | 0.0002 | .034 | 0.4 | 0.43 | 10.00 | 4.34 |
| HYDRAULIC POWER | 1.6.3 | V58 | 45 | 1.46 | 0.0146 | 2.453 | 29.2 | 31.65 | 15.25 | 482.55 |
| ACTUATION MECHANISM | 1.6.3 | V59 | 14K | 0.51 | 0.0051 | .857 | 10.2 | 11.06 | 77.52 | 857.12 |
| ATMOSPHERIC REVITIALIZATION | 1.10.1.1 | V61 | 41 | 0.38 | 0.0038 | .638 | 7.6 | 8.24 | 44.19 | 364.05 |
| ACTIVE THERMAL CONTROL | 1.10.1.3 | V63 | 41 | 2.74 | 0.0274 | 4.603 | 54.8 | 59.40 | 7.54 | 447.74 |
| NAVIGATION | 1.1.1 | V71 | 71,72,73 | 0.07 | 0.0007 | .118 | 1.4 | 1.52 | 77.52 | 117.64 |
| DATA PROCESSING | 1.1.4 | V72 | 82 | 0.24 | 0.0024 | .403 | 4.8 | 5.20 | 17.10 | 88.95 |
| COMMUNICATIONS | 1.1.3 | V74 | 67-69 | 0.86 | 0.0086 | 1.445 | 17.2 | 18.64 | 17.10 | 318.75 |
| INSTRUMENTATION | 1.1.6/7/8 | V75 | 51-55 | 0.66 | 0.0066 | 1.109 | 13.2 | 14.31 | 44.19 | 632.31 |
| ELECTICAL POWER DIST | 1.1.2 | V76 | 42 | 0.26 | 0.0026 | .437 | 5.2 | 5.64 | 7.54 | 42.49 |
| FLIGHT CONTROL (AVIONICS) | 1.1.1.3 | V79 | 14A-J | 0.09 | 0.0009 | .151 | 1.8 | 1.95 | 17.10 | 33.36 |
| | | | | TOTAL VEHICLE DRIVEN MAINTENANCE HOURS = | | | | | | 49863.37 |
| | | | | TOTAL NON-TILE MAINTENANCE HOURS | | | | | | 8257.56 |
| RECOVERY | | | | | | | | | | 80 |
| PRE / POST FLIGHT INSPECTIONS | | | | | | | | | | 240 |
| SERVICING | | | | | | | | | | 200 |
| FINAL SYSTEM TESTING | | | | | | | | | | 80 |
| HPF TO PAD TRANSPORTATION | | | | | | | | | | 160 |
| PAD INTEGRATION | | | | | | | | | | 200 |
| PAD INTERFACE CK OUT | | | | | | | | | | |
| | | | | TOTAL SSTO-TYPE FUNCTION DRIVEN MAINTENANCE HOURS | | | | | | 960 |
| | | | | TOTAL MAINTENANCE HOURS REQUIRED FOR TURNAROUND = | | | | | | 50823.37 |

*Figure 2–3–1*

# ORBITER AND AIRCRAFT FAILURE DATA COMPARISON

GROUND PROCESSING TIME= 250 HOURS
STS MISSION DURATION= 24 HOURS

(PR count = FAIL count)

| SYSTEMS & FUNCTIONS | WBS | WUC V | WUC A | JCWSI RCM DATA ( STS61-3 FLT RATE ) EQUIPMENT FAILURE RATE PRs / 100 OP HRS | PRs/HR | STS MISSION FAILS / MISSION | STS GROUND FAILS / MISSION | STS TOTAL FAILS / MISSION | MARTIN'S LVMAFR STS SYSTEM MTTR | TOTAL BURDEN HOURS |
|---|---|---|---|---|---|---|---|---|---|---|
| PRUGE, VENT & DRAIN. | 1.6.5 | V05 | 41 | 0.14 | 0.0014 | .034 | 0.35 | 0.38 | 16.20 | 6.21 |
| THERMAL PROTECTION | 1.7 | V09 | N/A | 54.00 | 0.54 | *12.960* | *135* | *147.96* | 35.64 | 5258.30 |
| STRUCTURE | 1.5 | V10-35 | 11 | 11.99 | 0.1199 | 2.878 | 29.975 | 32.85 | 5.72 | 187.88 |
| MAIN PROPULSION SYS | 1.3.1-5 | V41 | 23 | 3.19 | 0.0319 | .766 | 7.975 | 17.48 | 13.04 | 227.91 |
| REACTION CONTROL SYS | 1.4.2 | V42 | N/A | 0.23 | 0.0023 | .055 | 0.575 | 0.63 | 15.17 | 9.56 |
| ORBITAL MANEUVERING SYS | 1.4.1 | V43 | N/A | 0.17 | 0.0017 | .041 | 0.425 | 0.47 | 23.01 | 10.72 |
| ELECT POWER GENERATION | 1.2.1/2/3 | V45 | 42 | 0.57 | 0.0057 | .137 | 1.425 | 1.56 | 27.33 | 42.68 |
| AUXILIARY POWER UNIT | 1.6.3 | V46 | 24 | 1.66 | 0.0166 | .398 | 4.15 | 4.55 | 15.17 | 68.99 |
| LANDING & DECELERATION | 1.6.6 | V51 | 13 | 0.81 | 0.0081 | .194 | 2.025 | 2.22 | 10.00 | 22.19 |
| BRAKES & RECOVERY SYS | 1.6.6 | V52 | 13 | 0.05 | 0.0005 | .012 | 0.125 | 0.14 | 10.00 | 1.37 |
| PAYLOAD & RETENTION / DEPLOY | 1.6.8 | V54 | 75 | 0.26 | 0.0026 | .062 | 0.65 | 0.71 | 10.00 | 7.12 |
| PYRO & RANGE SAFETY | 1.9 / 1.12 | V55 | 12 | 0.19 | 0.0019 | .046 | 0.475 | 0.52 | 58.84 | 30.63 |
| AERO SURFACE CONTROL (FLT) | 1.6.1 | V57 | 14K | 0.02 | 0.0002 | .005 | 0.05 | 0.05 | 10.00 | 0.55 |
| HYDRAULIC POWER | 1.6.3 | V58 | 45 | 1.46 | 0.0146 | .350 | 3.65 | 4.00 | 15.25 | 60.99 |
| ACTUATION MECHANISM | 1.6.3 | V59 | 14K | 0.51 | 0.0051 | .122 | 1.275 | 1.40 | 77.52 | 108.33 |
| ATMOSPHERIC REVITIALIZATION | 1.10.1.1 | V61 | 41 | 0.38 | 0.0038 | .091 | 0.95 | 1.04 | 44.19 | 46.01 |
| ACTIVE THERMAL CONTROL | 1.10.1.3 | V63 | 41 | 2.74 | 0.0274 | .658 | 6.85 | 7.51 | 7.54 | 56.59 |
| NAVIGATION | 1.1.1 | V71 | 71,72,73 | 0.07 | 0.0007 | .017 | 0.175 | 0.19 | 77.52 | 14.87 |
| DATA PROCESSING | 1.1.4 | V72 | 82 | 0.24 | 0.0024 | .058 | 0.6 | 0.66 | 17.10 | 11.24 |
| COMMUNICATIONS | 1.1.3 | V74 | 67-69 | 0.86 | 0.0086 | .206 | 2.15 | 2.36 | 17.10 | 40.29 |
| INSTRUMENTATION | 1.1.6/7/8 | V75 | 51-55 | 0.66 | 0.0066 | .158 | 1.65 | 1.81 | 44.19 | 79.91 |
| ELECTICAL POWER DIST | 1.1.2 | V76 | 42 | 0.26 | 0.0026 | .062 | 0.65 | 0.71 | 7.54 | 5.37 |
| FLIGHT CONTROL (AVIONICS) | 1.1.1.3 | V79 | 14A-J | 0.09 | 0.0009 | .022 | 0.225 | 0.25 | 17.10 | 4.22 |
| | | | | TOTAL VEHICLE DRIVEN MAINTENANCE HOURS = | | | | | | 6301.92 |
| | | | | TOTAL NON-TILE MAINTENANCE HOURS | | | | | | 1043.62 |
| RECOVERY | | | | | | | | | | 80 |
| PRE / POST FLIGHT INSPECTIONS | | | | | | | | | | 240 |
| SERVICING | | | | | | | | | | 200 |
| FINAL SYSTEM TESTING | | | | | | | | | | 80 |
| HPF TO PAD TRANSPORTATION | | | | | | | | | | 160 |
| PAD INTEGRATION | | | | | | | | | | 200 |
| PAD INTERFACE CK OUT | | | | TOTAL SSTO-TYPE FUNCTION DRIVEN MAINTENANCE HOURS | | | | | | 960 |
| | | | | TOTAL MAINTENANCE HOURS REQUIRED FOR TURNAROUND = | | | | | | 07261.92 |

*Figure 2–3–2*

nance repair times are used to reflect modern design and maintenance approaches for reusable vehicle. This yielded a total turnaround maintenance burden of approximately 1400 hours which is significantly lower than results from other models. The potential for a turnaround burden of this magnitude will always be questionable by the current culture of space vehicle operators and maintainers. A paper written by LaRC Operations and Support compared maintenance requirements and turnaround times of their conceptual HL-20 vehicle and their X-15 vehicle. They concluded that is was possible to achieve the rapid turnaround time of a reusable space vehicle reflected in the third SSTO math model. A print out of the third model is shown in Figure 2-3-3.

The Shuttle Orbiter failure rates are higher for comparative systems and exposure times. Major improvements in failure rates have little affect on total maintenance burden. The driving factors are current Orbiter repair times, for both corrective and scheduled maintenance, and functional processing requirements.

## 2.4 TASK 4, DISCUSSION OF TOP-LEVEL AIRCRAFT MAINTENANCE PHILOSOPHIES

Maintenance philosophies are driven by two primary factors, 1) vehicle type and design approach, and 2) dictated policy based on vehicle operator's inherent cultural requirements. Aircraft and spacecraft have historically had radically different design and maintenance approaches based on frequency of intended use, as well as many other factors. Both communities emphasize priority on cost, life cycle cost, safety (population and crew members) and utilization. Optimization of design approaches, processes, maintenance and logistics of both communities is a must to achieve the goals (which should evolve into hard program requirements) for the next generation reusable spacecraft.

The following paragraphs describe the standard USAF aircraft maintenance approach and levels.

The maintenance philosophy of the Air Force is based on a multi-level maintenance concept. Typically, an aircraft system can be maintained under a two level maintenance concept or a three level maintenance concept. If there is not some overriding reason (either technical or contractual) that dictates accomplishment of a maintenance task at a specific level, the decision can be made on purely economic considerations. The determination of two level versus three level is made by performing this analysis.

The three levels of maintenance recognized by the Air Force are Organizational, Intermediate and Depot. The Air Force goal is to maintain equipment at the lowest level possible, or closest to the weapon system. This limits the amount of time available to restore the system to an operational condition after it fails which in turn limits the required resources necessary to support the maintenance task.

Organizational level (O-level) maintenance is also referred to as flight line maintenance. It is performed by the user of the equipment. The capabilities of O-level are normally limited to periodic servicing of the equipment, troubleshooting to identify failures, and removing/replacing major components. Again the factors limiting what maintenance can be performed at this level are the tools, test equipment and training of the personnel. Using a piece of electronic equipment as an example, O-level is usually limited to periodic testing of the equipment to ensure it functions, removing the item when it fails and replacing the entire failed item with one that works. At O-level, the user's

# ORBITER AND AIRCRAFT FAILURE DATA COMPARISON

(PR count = FAIL count)

GROUND PROCESSING TIME= 250
SSTO MISSION DURATION= 24

| SYSTEMS & FUNCTIONS | WBS | WUC V | WUC A | JCWSI RCM DATA (STS61-3 FLT RATE) EQUIPMENT FAILURE RATE /100 OP | PRs/HR | SSTO MISSION FAILS / MISSION | SSTO GROUND FAILS / MISSION | SSTO TOTAL FAILS / MISSION | AIRCRAFT SYSTEM MTTR | TOTAL CORR HOURS | PLUS 50% TOTAL BURDEN HOURS |
|---|---|---|---|---|---|---|---|---|---|---|---|
| PRUGE, VENT & DRAIN. | 1.6.5 | V05 | 41 | 0.14 | 0.0014 | .034 | 0.35 | 0.38 | 0.72 | 0.28 | 00.39 |
| THERMAL PROTECTION | 1.7 | V09 | N/A | 54.00 | 0.54 | 12.960 | | 12.96 | 1.00 | 12.96 | 18.14 |
| STRUCTURE | 1.5 | V10-35 | 11 | 11.99 | 0.1199 | 2.878 | 29.975 | 32.85 | 1.02 | 33.51 | 46.91 |
| MAIN PROPULSION SYS | 1.3.1-5 | V41 | 23 | 3.19 | 0.0319 | .766 | 7.975 | 17.48 | 10.00 | 174.81 | 244.74 |
| REACTION CONTROL SYS | 1.4.2 | V42 | N/A | 0.23 | 0.0023 | .055 | 0.575 | 0.63 | 15.17 | 9.56 | 13.38 |
| ORBITAL MANEUVERING SYS | 1.4.1 | V43 | N/A | 0.17 | 0.0017 | .041 | 0.425 | 0.47 | 23.01 | 10.72 | 15.00 |
| ELECT POWER GENERATION | 1.2.1/2/3 | V45 | 42 | 0.57 | 0.0057 | .137 | 1.425 | 1.56 | 1.05 | 1.64 | 02.30 |
| AUXILIARY POWER UNIT | 1.6.3 | V46 | 24 | 1.66 | 0.0166 | .398 | 4.15 | 4.55 | 2.70 | 12.28 | 17.19 |
| LANDING & DECELERATION | 1.6.6 | V51 | 13 | 0.81 | 0.0081 | .194 | 2.025 | 2.22 | 0.99 | 2.20 | 03.08 |
| BRAKES & RECOVERY SYS | 1.6.6 | V52 | 13 | 0.05 | 0.0005 | .012 | 0.125 | 0.14 | 0.93 | 0.13 | 00.18 |
| PAYLOAD & RETENTION / DEPLOY | 1.6.8 | V54 | 75 | 0.26 | 0.0026 | .062 | 0.65 | 0.71 | 2.84 | 2.02 | 02.83 |
| PYRO & RANGE SAFETY | 1.9 / 1.12 | V56 | 12 | 0.19 | 0.0019 | .046 | 0.475 | 0.52 | 2.50 | 1.30 | 01.82 |
| AERO SURFACE CONTROL (FLT) | 1.6.1 | V57 | 14K | 0.02 | 0.0002 | .005 | 0.05 | 0.05 | 1.24 | 0.07 | 00.10 |
| HYDRAULIC POWER | 1.6.3 | V58 | 45 | 1.46 | 0.0146 | .350 | 3.65 | 4.00 | 1.04 | 4.16 | 05.82 |
| ACTUATION MECHANISM | 1.6.3 | V59 | 14K | 0.51 | 0.0051 | .122 | 1.275 | 1.40 | 1.58 | 2.21 | 03.09 |
| ATMOSPHERIC REVITIALIZATION | 1.10.1.1 | V61 | 41 | 0.38 | 0.0038 | .091 | 0.95 | 1.04 | 0.72 | 0.75 | 01.05 |
| ACTIVE THERMAL CONTROL | 1.10.1.3 | V63 | 41 | 2.74 | 0.0274 | .658 | 6.85 | 7.51 | 7.54 | 56.59 | 79.22 |
| NAVIGATION | 1.1.1 | V71 | 71,72,73 | 0.07 | 0.0007 | .017 | 0.175 | 0.19 | 1.03 | 0.20 | 00.28 |
| DATA PROCESSING | 1.1.4 | V72 | 82 | 0.24 | 0.0024 | .058 | 0.6 | 0.66 | 2.55 | 1.68 | 02.35 |
| COMMUNICATIONS | 1.1.3 | V74 | 67-69 | 0.86 | 0.0086 | .206 | 2.15 | 2.36 | 1.00 | 2.36 | 03.30 |
| INSTRUMENTATION | 1.1.6/7/8 | V75 | 51-55 | 0.66 | 0.0066 | .158 | 1.65 | 1.81 | 0.90 | 1.63 | 02.28 |
| ELECTICAL POWER DIST | 1.1.2 | V76 | 42 | 0.26 | 0.0026 | .062 | 0.65 | 0.71 | 0.77 | 0.55 | 00.77 |
| FLIGHT CONTROL (AVIONICS) | 1.1.1.3 | V79 | 14A-J | 0.09 | 0.0009 | .022 | 0.225 | 0.25 | 0.88 | 0.22 | 00.30 |
| | | | | TOTAL VEHICLE DRIVEN MAINTENANCE HOURS = | | | | | | 331.80 | 464.52 |
| RECOVERY | | | | | | | | | | | 80 |
| PRE / POST FLIGHT INSPECTIONS | | | | | | | | | | | 240 |
| SERVICING | | | | | | | | | | | 200 |
| FINAL SYSTEM TESTING | | | | | | | | | | | 80 |
| HPF TO PAD TRANSPORTATION | | | | | | | | | | | 160 |
| PAD INTEGRATION | | | | | | | | | | | 200 |
| PAD INTERFACE CK OUT | | | | TOTAL FUNCTION DRIVEN MAINTENANCE HOURS | | | | | | | 960 |
| | | | | TOTAL MAINTENANCE HOURS REQUIRED FOR TURNAROUND = | | | | | | | 1424.52 |

*Figure 2–3–3*

mission is to fulfill the requirement of using the equipment to accomplish its intended mission; therefore maintenance planning must consider possible short term solutions to remedy the failure. At this level, the maintenance planning is normally accomplished early in a program development phase by performing a detailed Reliability Centered Maintenance (RCM) analysis. It is not uncommon for O- level tasks to be limited to a Mean Time To Repair (MTTR) of thirty (30) minutes or less.

Maintenance actions that are not within the capabilities of O-level are passed to the next higher level-Intermediate (I-Level) maintenance. I-level is more capable of performing maintenance because that is its primary mission. Because the complexity of maintenance tasks increases at this level, it has a greater range of tools and test equipment available and personnel possess higher skill levels. I-level generally consists of testing the items removed at O-level and replacing failed modules. A typical MTTR for I-level maintenance is sixty (60) minutes.

Maintenance actions that cannot be performed at either O-level or I-level are passed on to the Depot level (D-level). D-level has the capability to do anything necessary to repair the failed item. D-level facilities normally have the widest range of tools test equipment and highly skilled maintenance personnel. Fabrication of structural parts, major overhauls and refurbishment, and complete rebuilding of equipment can be done at D-level. Within the Air Force, D-level is normally the responsibility of a single facility for each type of equipment or major assembly. For instance, one depot might be responsible for the repair and overhaul of the F404 jet engine. All aircraft that use this engine would send it to that specific depot for repair. These depots are sometimes referred to as Air Logistics Commands (ALC).

**Maintenance Actions**

The Air Force defines a maintenance action as any action performed by designated maintenance personnel to service, test, calibrate or restore a weapon system to an operational ready state.

There are two types of maintenance actions as recognized by the Air Force. Scheduled maintenance actions and unscheduled maintenance actions.

The determination as to whether a specific task is to be performed on a scheduled or unscheduled basis is made through the performance of a Reliability Centered Maintenance Analysis (RCMA).

The main objective of an RCMA is to examine each failure mode/cause, its consequences, periodicity (as dictated by the FMECA) and determine if a schedule maintenance task could have prevented the failure. This maximizes the inherent reliability of the item under analysis.

The RCMA process considers the significant items that comprise the item under analysis. As stated above, it uses data generated by the FMECA to identify items most critical to the reliability of the item and where a failure would have the greatest negative effect on system availability.

The key to RCMA is the RCM decision logic (refer to MIL-STD-2173) tree. Each failure mode is examined using this logic tree to establish a maintenance task and its periodicity. A scheduled maintenance task is only established if it is applicable (resists or detects an impending failure) and is effective (cost effective). Economic tradeoffs are conducted where needed to determine if the benefits of a scheduled task exceeds the cost of performing that task; or if the cost of a redesign effort offsets the risk of failure for which there is no applicable and effective task (usually safety of flight systems).

The resultant documentation is used to generate a scheduled maintenance program for the weapon system. A by product is a listing of all the maintenance tasks that are to be performed on an unscheduled basis, which aids in performance of maintenance planning activities.

**AFM 66-1**

The AFM 66-1, Maintenance Data Collection System was originally devised as a tool for base level management. It was designed to assure effective management, at the base level, of all Air Force resources, tools, equipment skills and manpower. Portions of the data collected were also furnished to the Air Force Logistics Command for logistics support requirements. As the potential of the system was recognized, it's scope was enlarged and its use expanded. From its introduction in July 1958, the information has been used for cost analysis, budget computations, material improvement, training, improved maintenance, life expectancy, reliability and maintainability, determination of potential critical items, requirements and many other areas including Configuration Management.

## 2.5 TASK 5, PREDICTION OF TYPICAL SSTO TURNAROUND TIME

This task identifies a top-level serial (calendar) turnaround timeline for vehicle processing between missions and the associated availability. Operational Availability ($A_O$) is the primary Maintainability figure-of-merit which provides an understanding of potential utilization of a given system. $A_O$ is evaluated for planning proposes such as fleet sizing. The total turnaround burden (total maintenance hours) and predicted timeline determines the system downtime (unavailable time) which allows computation of $A_O$. The processing timeline is shown in Figure 2-5-1 and $A_O$, based on mission rate, is shown in Figure 2-5-2.

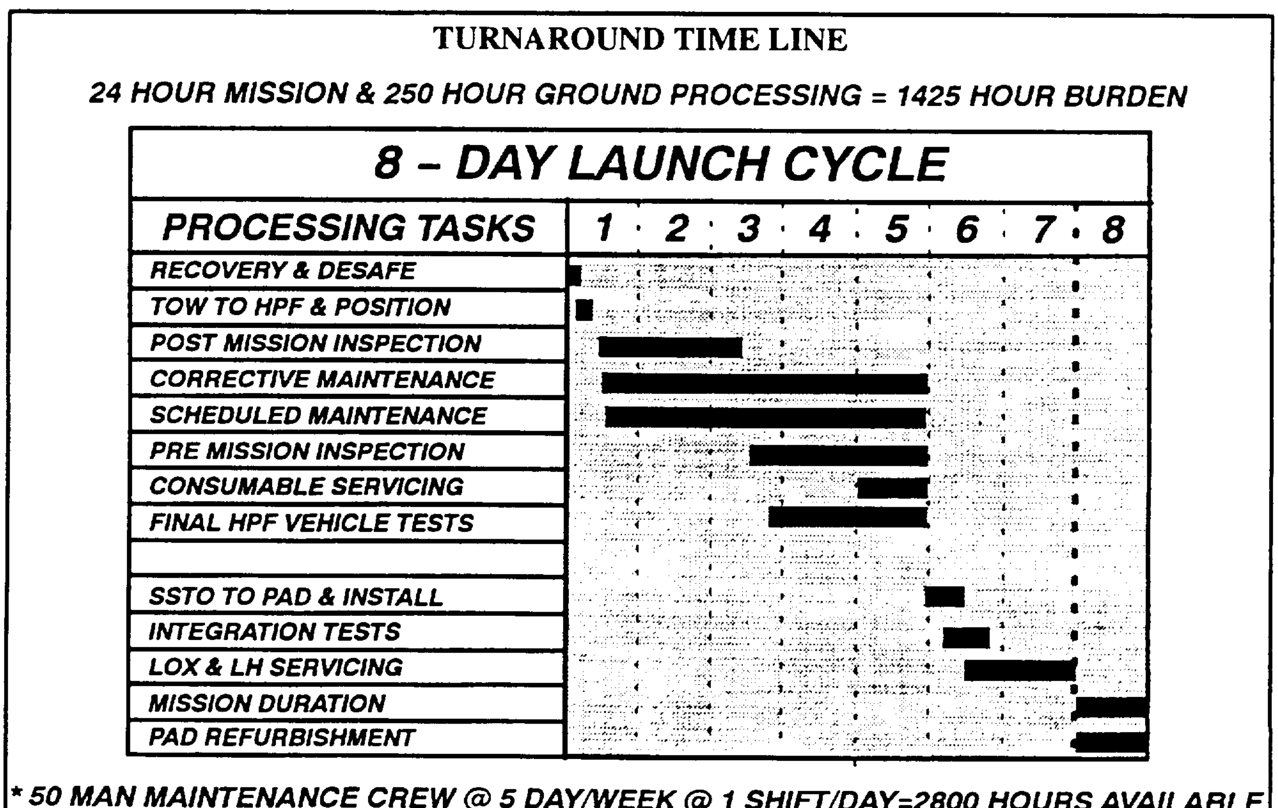



*Figure 2–5–1*

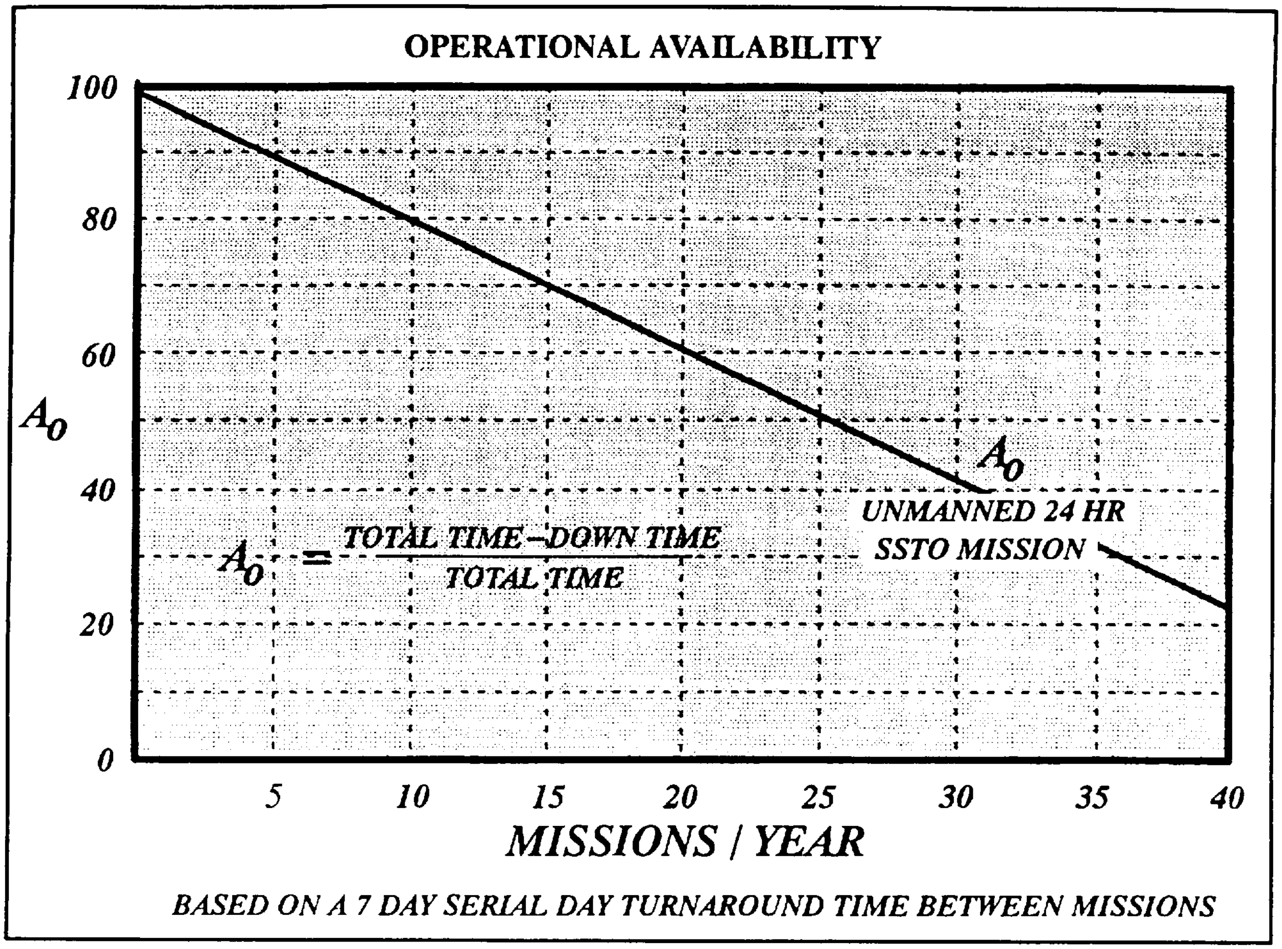



*Figure 2–5–2*

## 2.6 TASK 6, LADC'S R&M TOOL KIT DEMONSTRATION

The attached set of viewgraphs represents the analysis of the environmental control system on the E–3C. In simple terms the analysis takes the reviewer down from a high level perspective (System or Subsystem) down to the individual LRU and SRU. The charts should be arranged consistently, so that the higher level charts are together etc. The objective will be to show the audience just how easy it is to isolate design-related problems using an interactive database management tool. The presentation will move steadily from the general to the specific. In this case we will show how and why our attention was focused on subsystem 41A. That will be followed by charts designed to isolate the faulty SRUs and a brief discussion / explanation of our interpretation will be made as the presentation concludes.

# MDC MAINTENANCE TASK BREAKDOWN

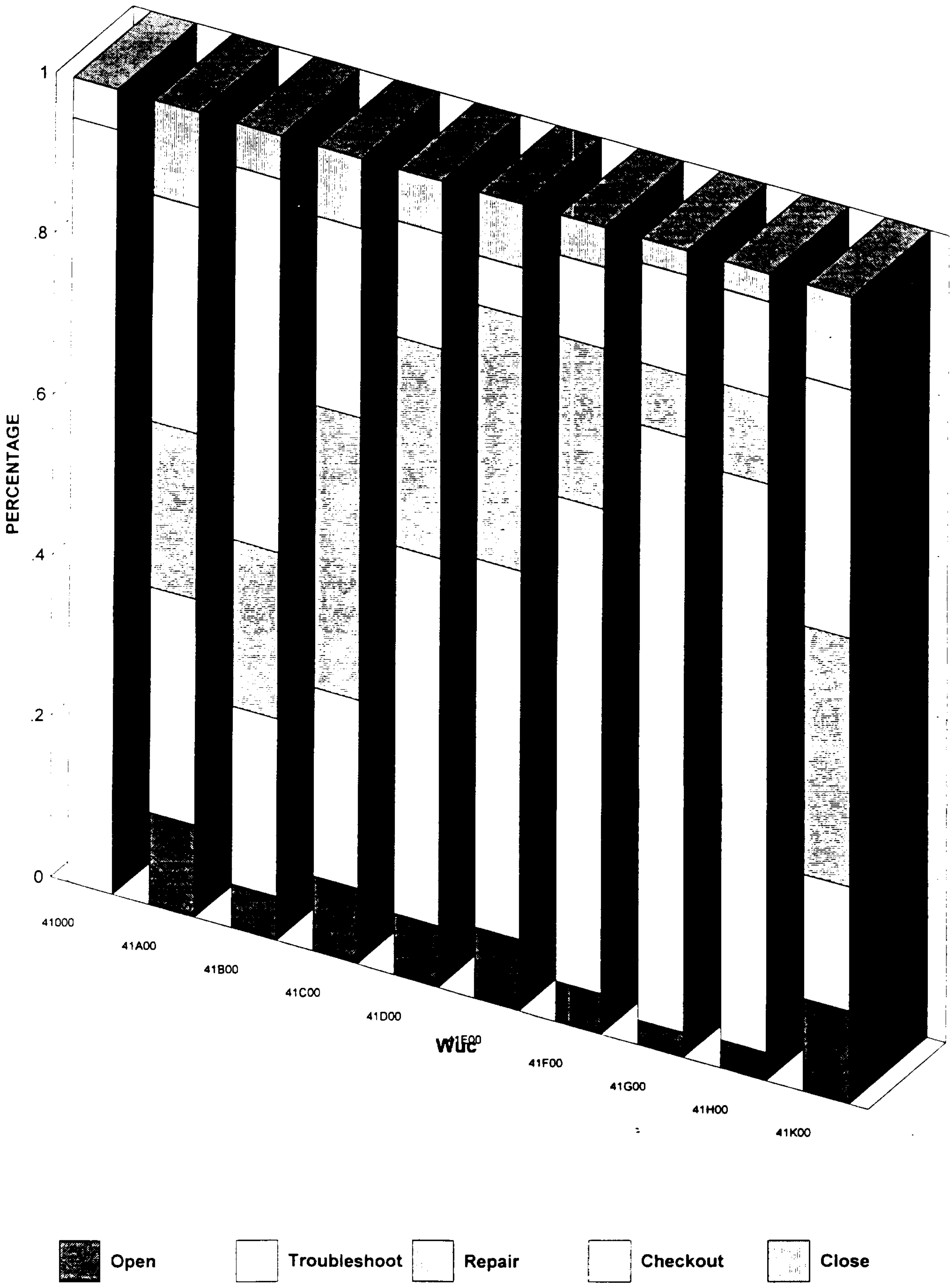
**Today : 6/13/94**

Aircraft : E003C Flight Hours : 3,476.00

| Wuc | Year | Ind | Open* | Troubleshoot | Repair | Checkout | Close |
|---|---|---|---|---|---|---|---|
| 41000 | 91 | 3 | 0.00 | 0.95 | 0.00 | 0.05 | 0.00 |
| 41A00 | 91 | 3 | 0.12 | 0.28 | 0.21 | 0.28 | 0.12 |
| 41B00 | 91 | 3 | 0.06 | 0.22 | 0.21 | 0.46 | 0.06 |
| 41C00 | 91 | 3 | 0.09 | 0.23 | 0.35 | 0.23 | 0.09 |
| 41D00 | 91 | 3 | 0.08 | 0.46 | 0.26 | 0.14 | 0.06 |
| 41E00 | 91 | 3 | 0.09 | 0.46 | 0.32 | 0.06 | 0.08 |
| 41F00 | 91 | 3 | 0.05 | 0.60 | 0.20 | 0.10 | 0.05 |
| 41G00 | 91 | 3 | 0.03 | 0.74 | 0.08 | 0.12 | 0.03 |
| 41H00 | 91 | 3 | 0.03 | 0.70 | 0.11 | 0.12 | 0.03 |
| 41K00 | 91 | 3 | 0.12 | 0.15 | 0.31 | 0.31 | 0.12 |

* NOTE: ALL TIMES ARE IN HOURS.

# MAINTENANCE TIMELINE DATA



# MDC MAINTENANCE TASK BREAKDOWN

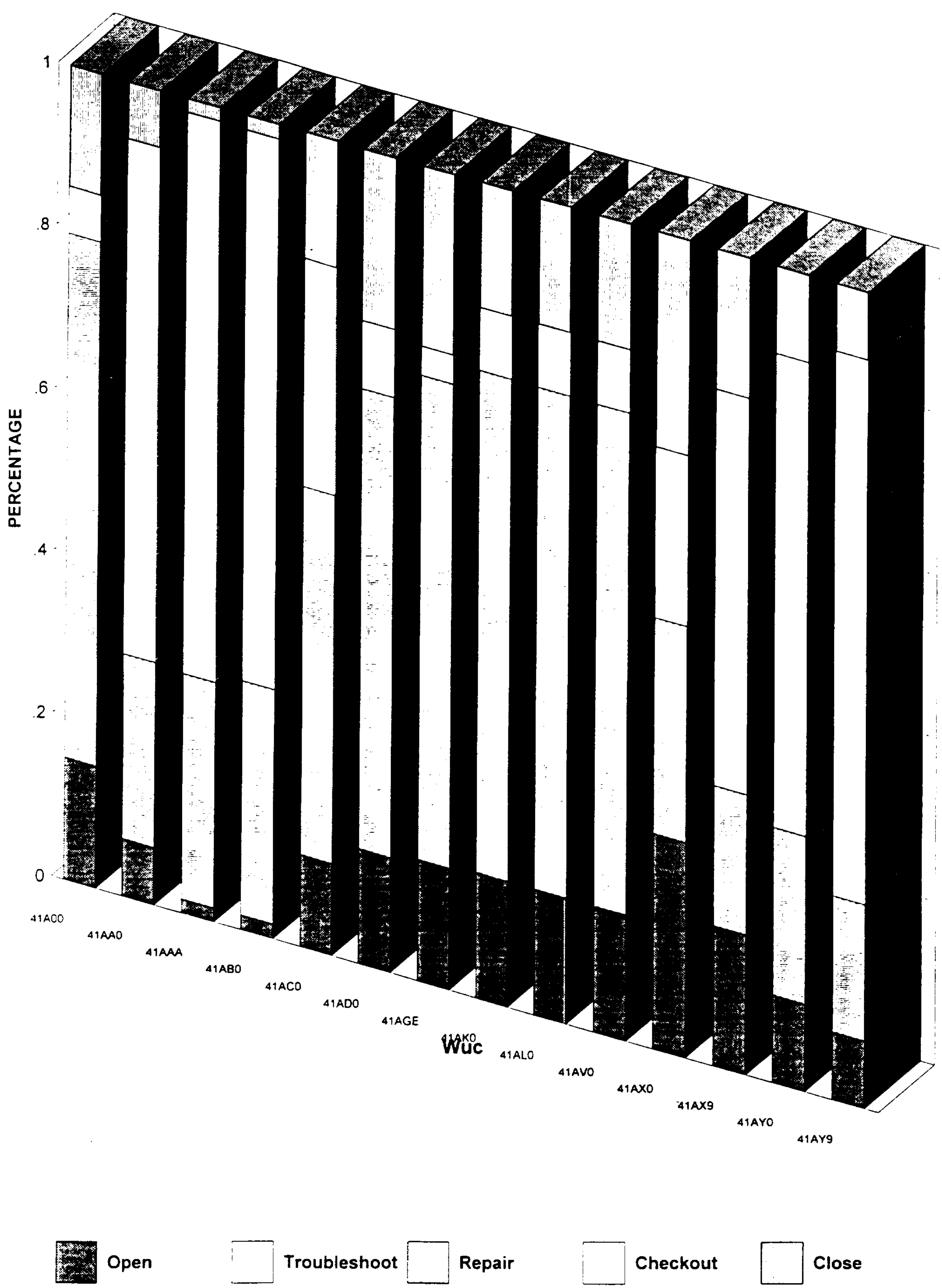
Today : 6/13/94

Aircraft : E003C Flight Hours : 3,476.00

| Wuc | Year | Ind | Open* | Troubleshoot | Repair | Checkout | Close |
|---|---|---|---|---|---|---|---|
| 41A00 | 91 | 5 | 0.15 | 0.00 | 0.64 | 0.06 | 0.15 |
| 41AA0 | 91 | 5 | 0.07 | 0.00 | 0.23 | 0.63 | 0.07 |
| 41AAA | 91 | 5 | 0.02 | 0.00 | 0.28 | 0.69 | 0.02 |
| 41AB0 | 91 | 5 | 0.02 | 0.00 | 0.29 | 0.68 | 0.02 |
| 41AC0 | 91 | 5 | 0.11 | 0.00 | 0.45 | 0.28 | 0.16 |
| 41AD0 | 91 | 5 | 0.14 | 0.00 | 0.57 | 0.08 | 0.21 |
| 41AGE | 91 | 5 | 0.15 | 0.00 | 0.59 | 0.03 | 0.22 |
| 41AK0 | 91 | 5 | 0.15 | 0.00 | 0.62 | 0.08 | 0.15 |
| 41AL0 | 91 | 5 | 0.15 | 0.00 | 0.62 | 0.08 | 0.15 |
| 41AV0 | 91 | 5 | 0.15 | 0.00 | 0.62 | 0.08 | 0.15 |
| 41AX0 | 91 | 5 | 0.26 | 0.00 | 0.26 | 0.21 | 0.26 |
| 41AX9 | 91 | 5 | 0.17 | 0.00 | 0.17 | 0.49 | 0.17 |
| 41AY0 | 91 | 5 | 0.11 | 0.00 | 0.21 | 0.58 | 0.11 |
| 41AY9 | 91 | 5 | 0.08 | 0.00 | 0.17 | 0.67 | 0.08 |

* NOTE: ALL TIMES ARE IN HOURS.

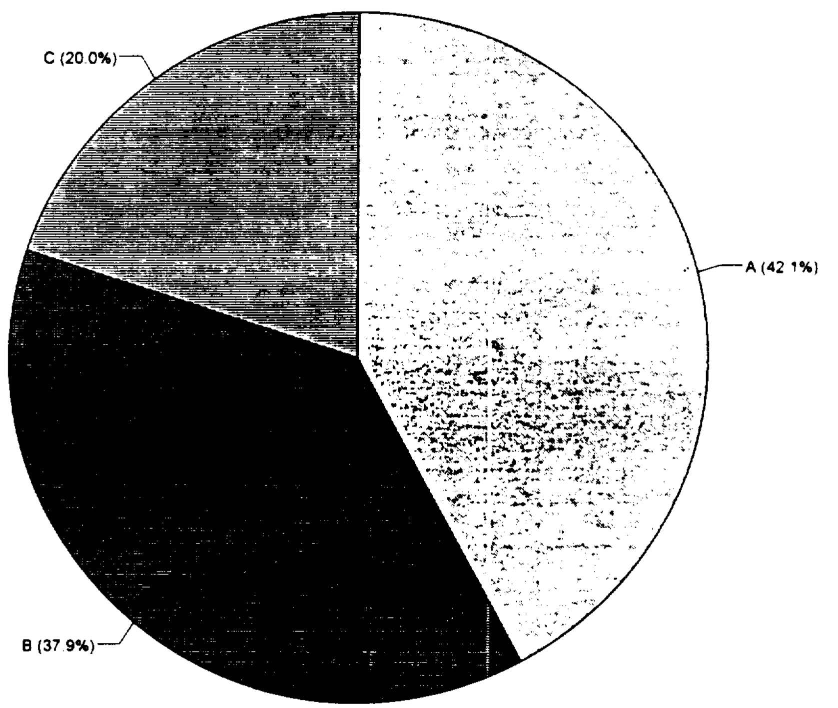
# MAINTENANCE TIMELINE DATA



# TYPE HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41**

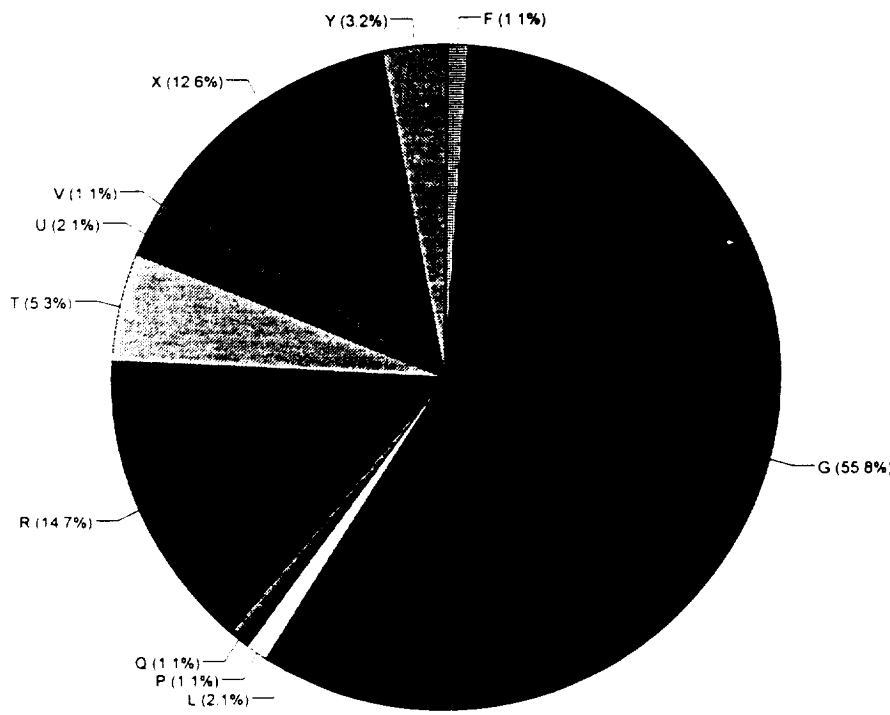



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# ACTION TAKEN BREAKDOWN

**Aircraft : E003C**

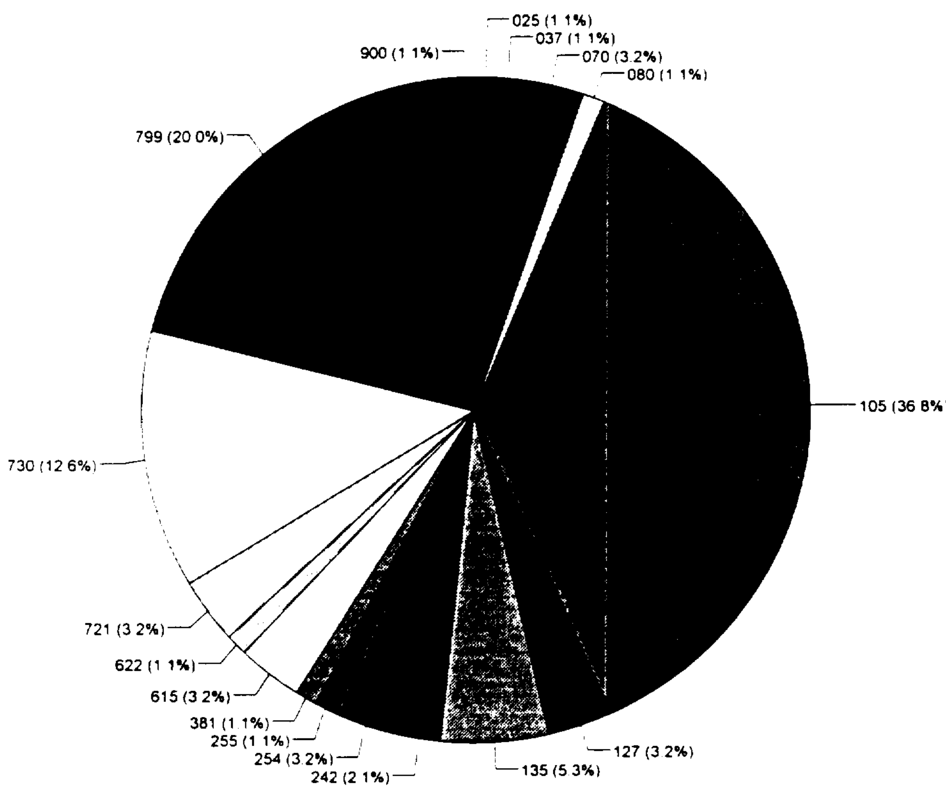
**Selection WUC : 41**



# HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41**

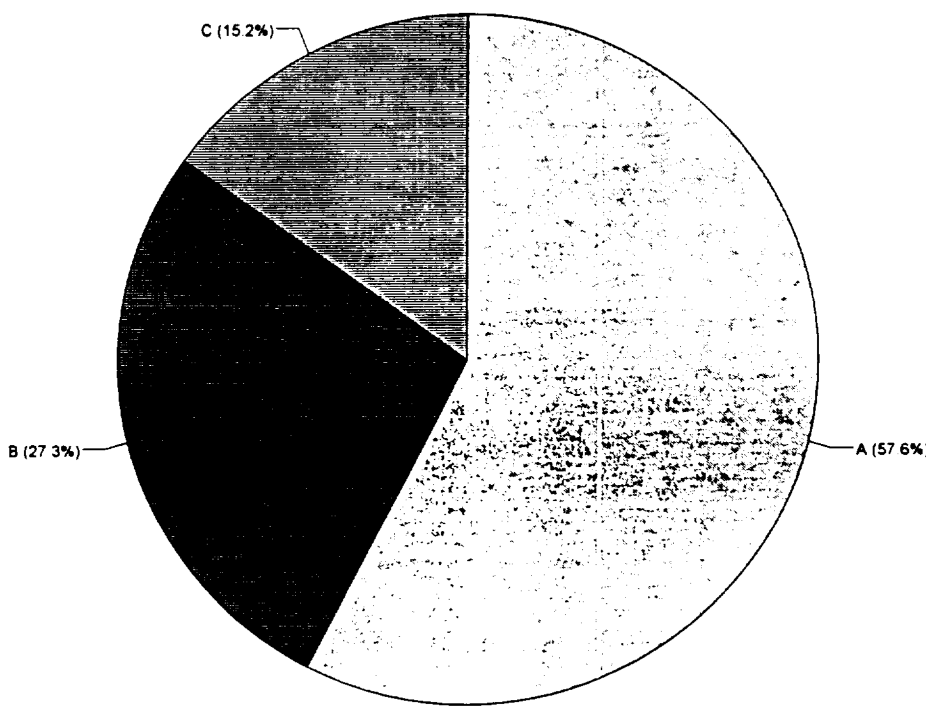



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# TYPE HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41A..**

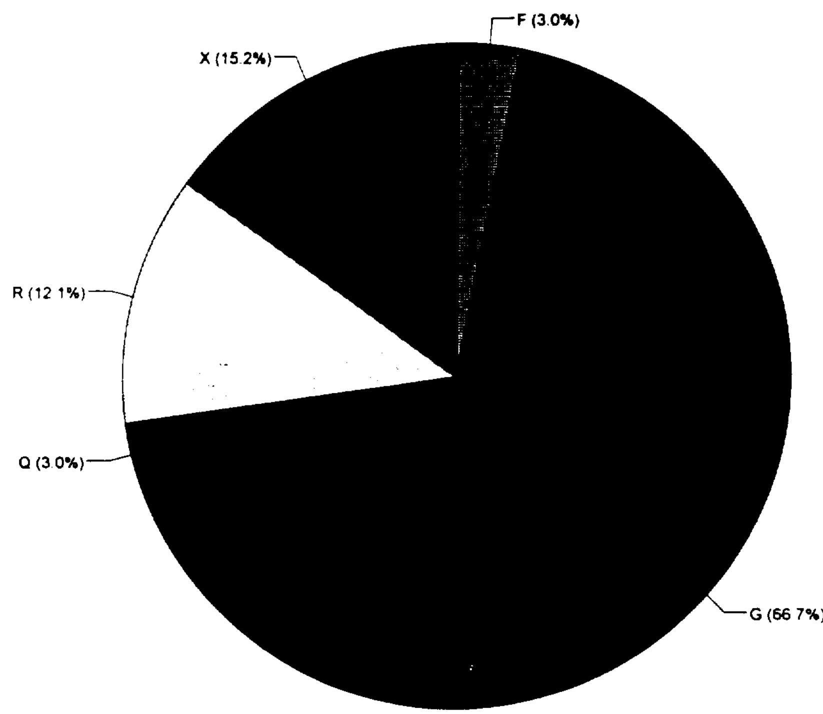



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# ACTION TAKEN BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41A..**

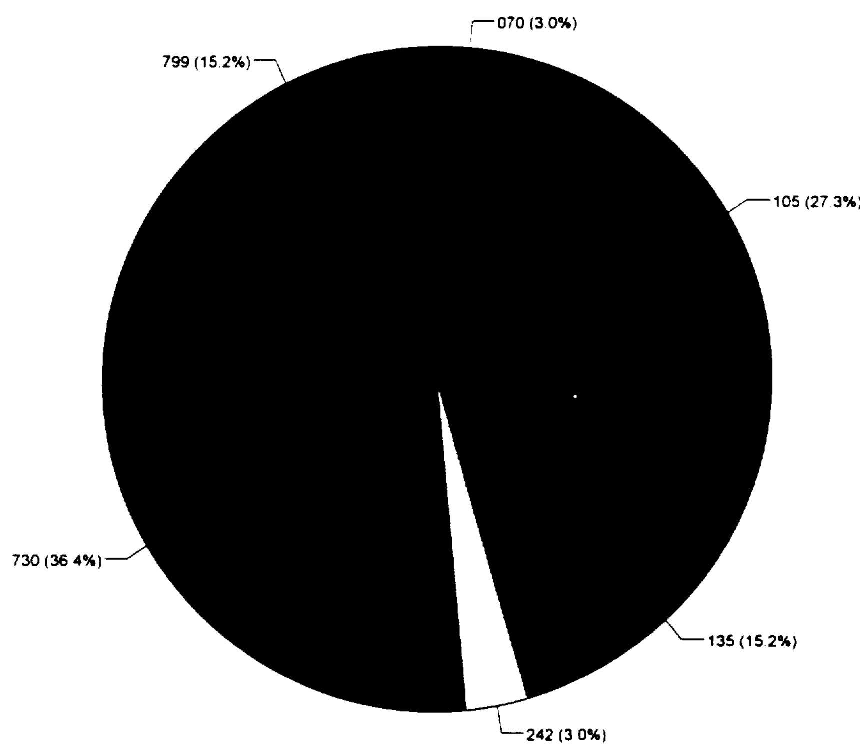



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41A..**

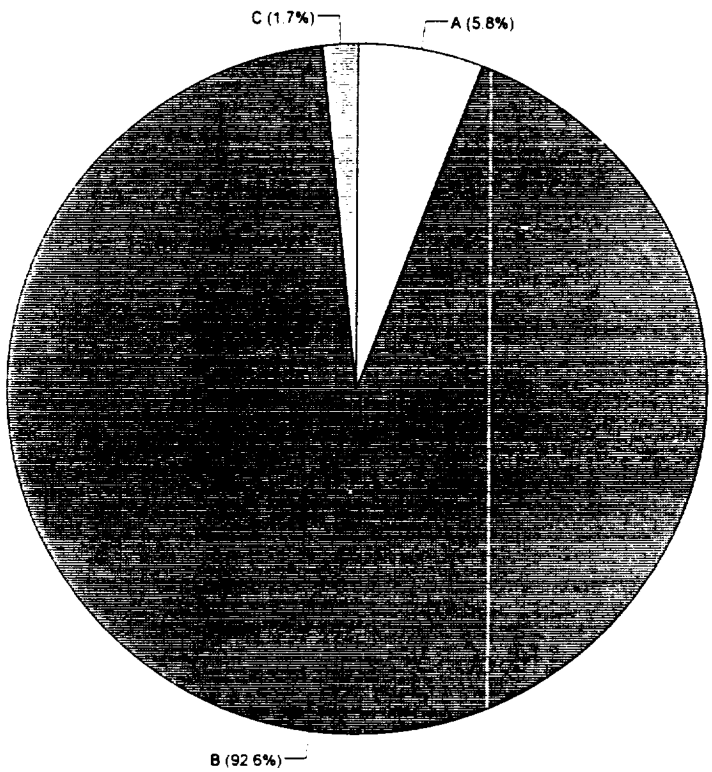



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# TYPE HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41AY0**

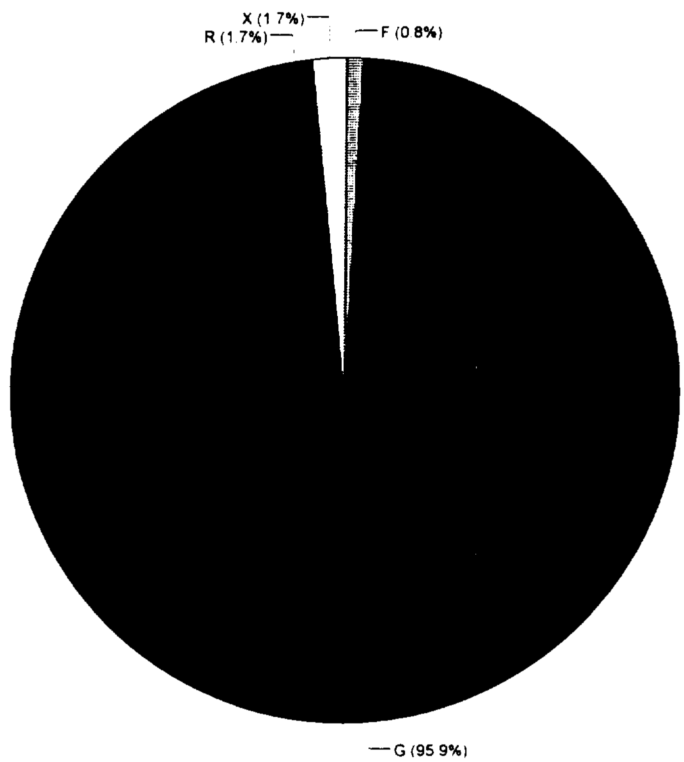



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# ACTION TAKEN BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41AY0**

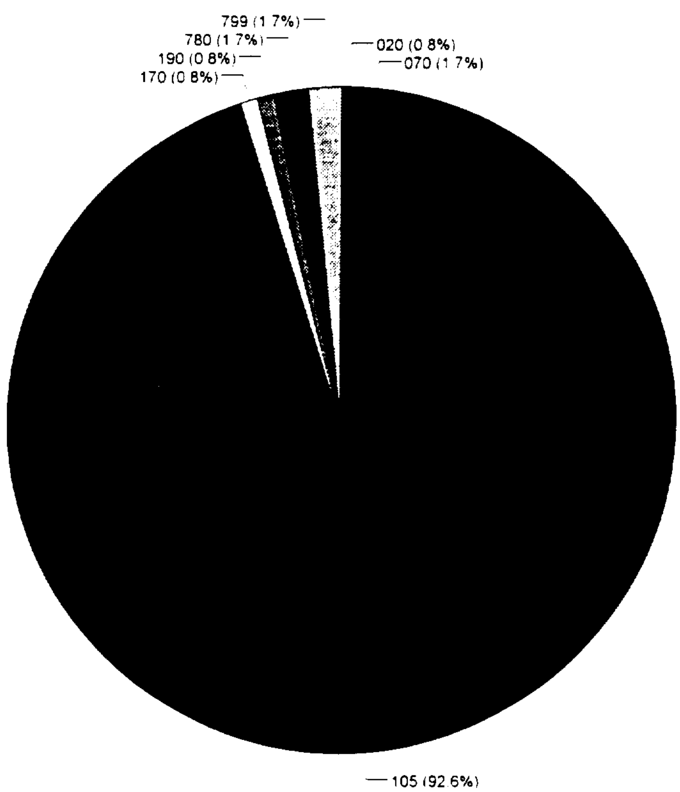



R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES

# HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41AY0**

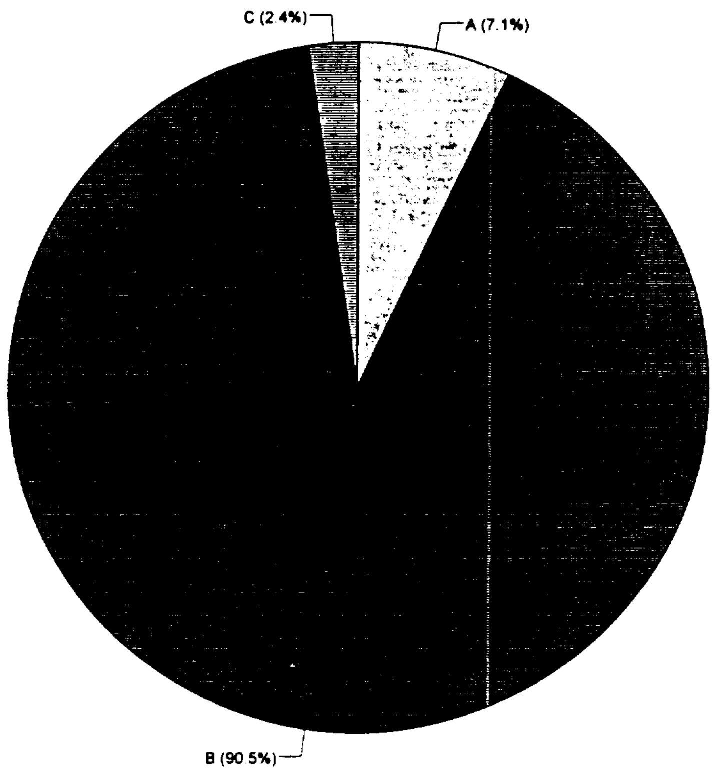



R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES

# TYPE HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41AX0**

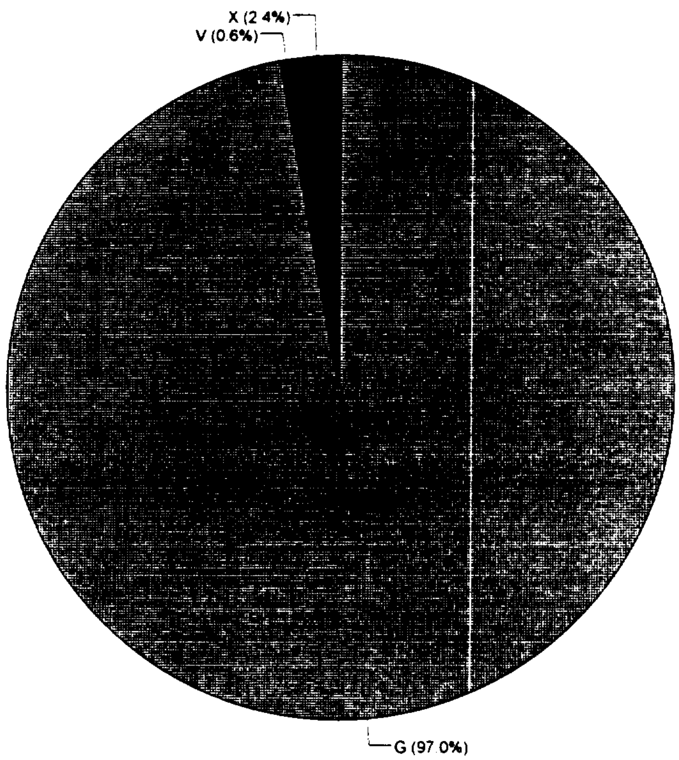



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# ACTION TAKEN BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41AX0**

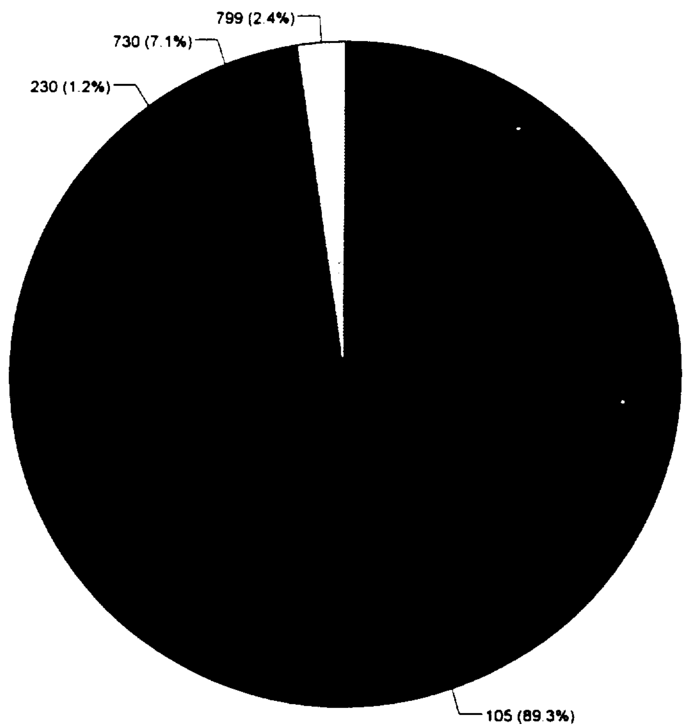



**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# HOW MAL BREAKDOWN

**Aircraft : E003C**

**Selection WUC : 41AX0**



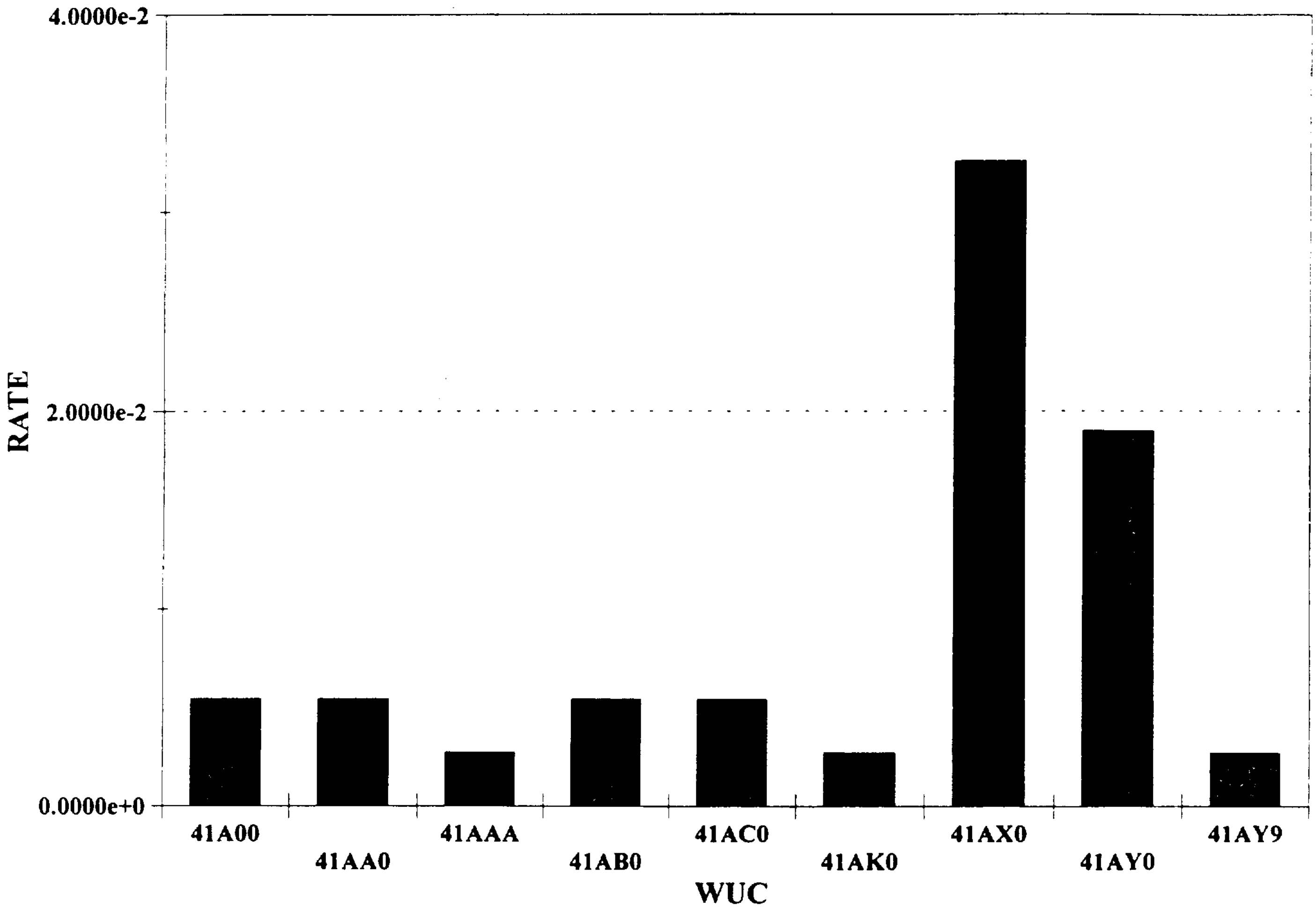

**R&M TOOLKIT - GENERAL PURPOSE GRAPH SERIES**

# COMPARATIVE BREAK RATES

**Aircraft : E003C**

**Today : 6/13/94**

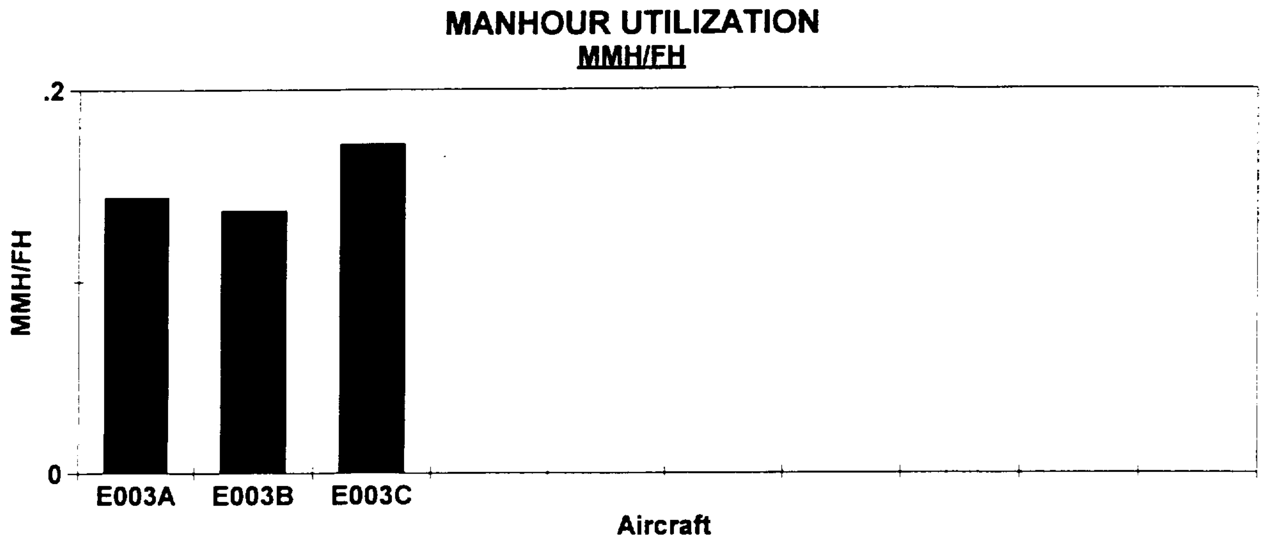
## BREAK RATES

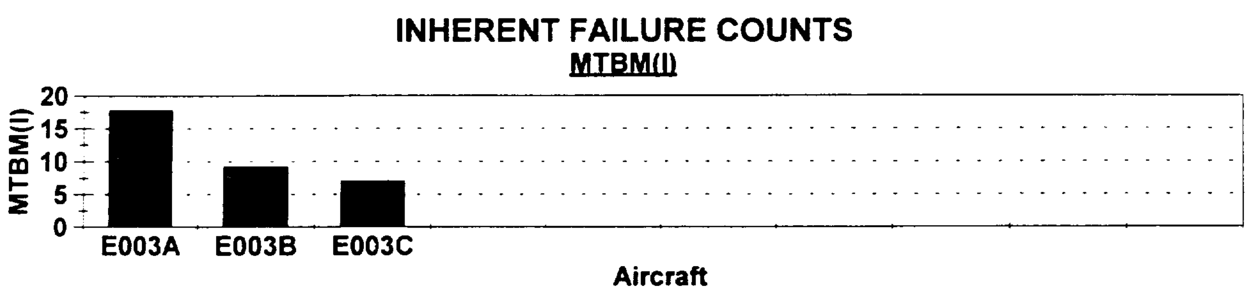


# MANHOURS AND FAILURES BY AIRCRAFT

Today : 6/13/94

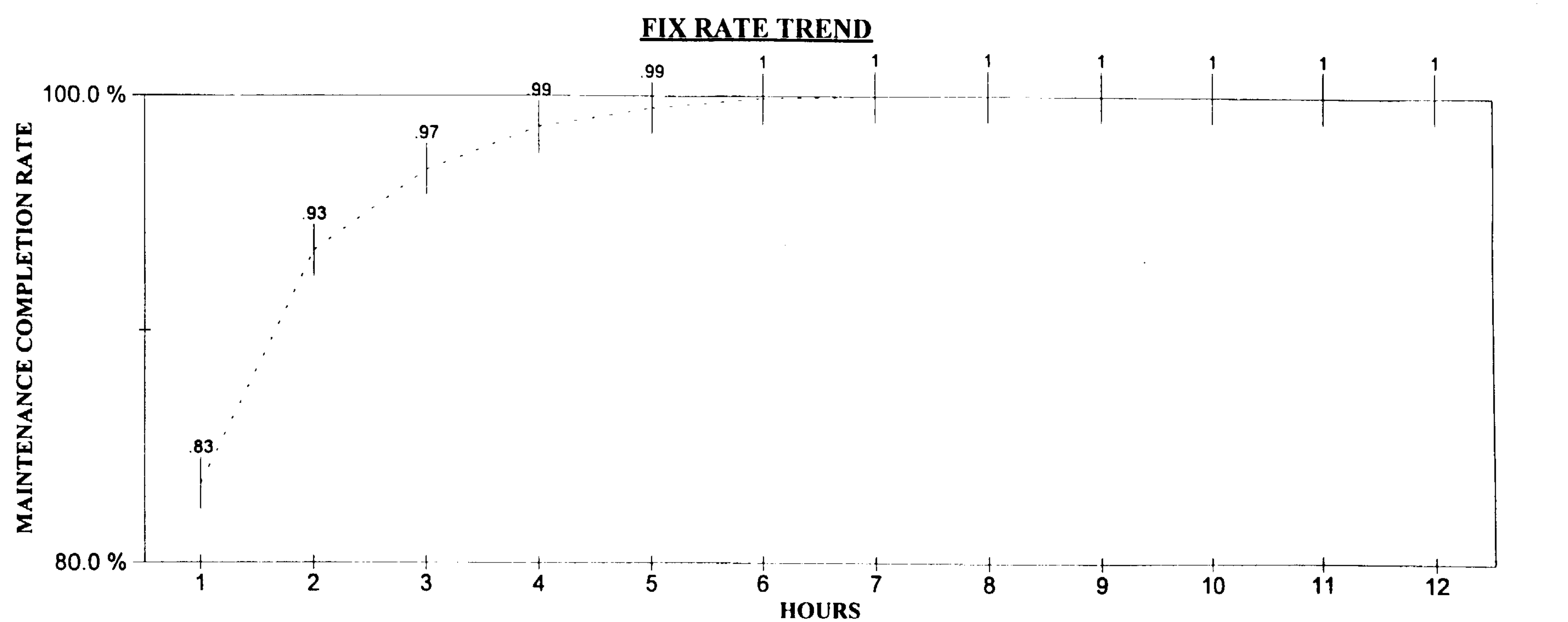
41000 AIR CONDITIONING AND PRESSURIZATION



# FIX RATE CHART

Aircraft : E003C   Wuc : 00000   Nomenclature : AIRCRAFT LEVEL

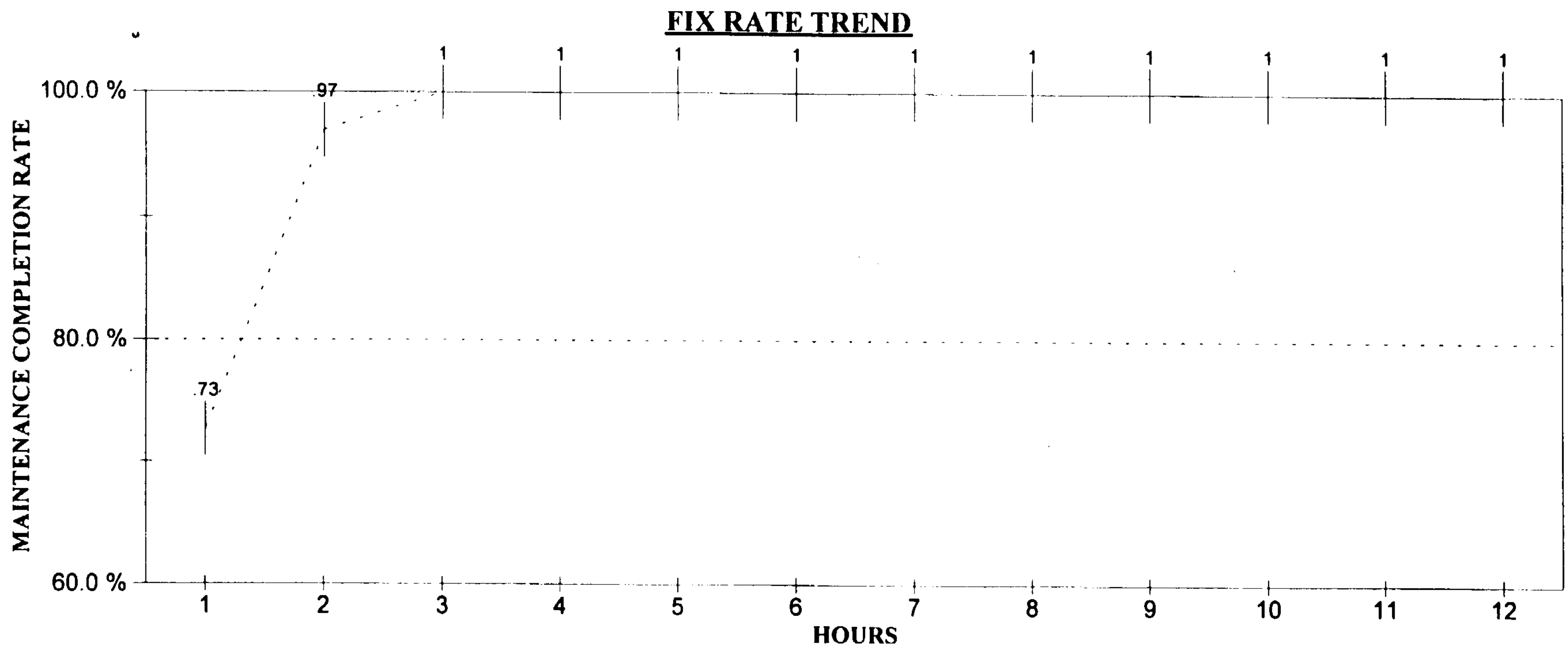



## CURRENT DATA

| Hour | Percent | Events |
|---|---|---|
| 1.00 | 0.83 | 824.00 |
| 2.00 | 0.93 | 99.00 |
| 3.00 | 0.97 | 34.00 |
| 4.00 | 0.99 | 18.00 |
| 5.00 | 0.99 | 8.00 |
| 6.00 | 1.00 | 4.00 |
| 7.00 | 1.00 | 1.00 |
| 8.00 | 1.00 | 0.00 |
| 9.00 | 1.00 | 0.00 |
| 10.00 | 1.00 | 0.00 |
| 11.00 | 1.00 | 0.00 |
| 12.00 | 1.00 | 0.00 |

# FIX RATE CHART

Aircraft : **E003C** Wuc : **41000** Nomenclature : **AIR COND PRESS**



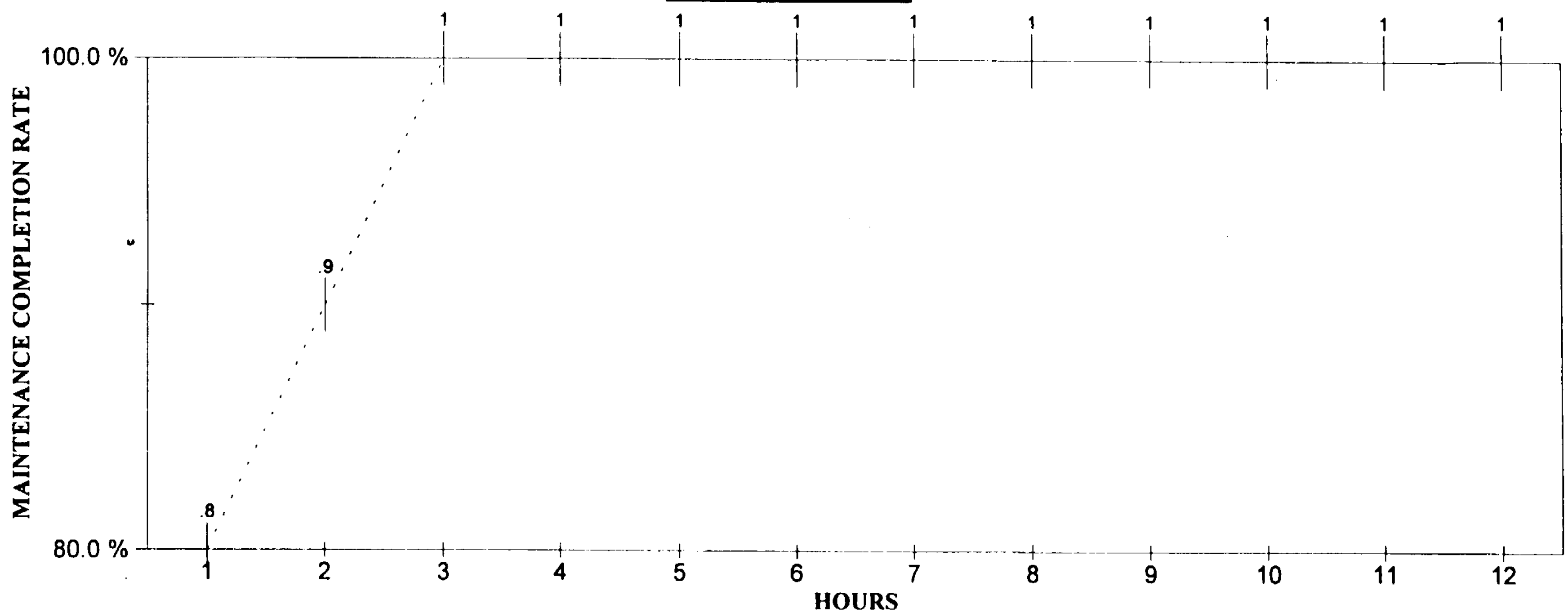

**CURRENT DATA**

| Hour | Percent | Events |
|---|---|---|
| 1.00 | 0.73 | 24.00 |
| 2.00 | 0.97 | 8.00 |
| 3.00 | 1.00 | 1.00 |
| 4.00 | 1.00 | 0.00 |
| 5.00 | 1.00 | 0.00 |
| 6.00 | 1.00 | 0.00 |
| 7.00 | 1.00 | 0.00 |
| 8.00 | 1.00 | 0.00 |
| 9.00 | 1.00 | 0.00 |
| 10.00 | 1.00 | 0.00 |
| 11.00 | 1.00 | 0.00 |
| 12.00 | 1.00 | 0.00 |

# FIX RATE CHART

Aircraft : **E003C** Wuc : **41A00** Nomenclature : **BLEED AIR SYST**

## FIX RATE TREND




## CURRENT DATA

| Hour | Percent | Events |
|---|---|---|
| 1.00 | 0.80 | 8.00 |
| 2.00 | 0.90 | 1.00 |
| 3.00 | 1.00 | 1.00 |
| 4.00 | 1.00 | 0.00 |
| 5.00 | 1.00 | 0.00 |
| 6.00 | 1.00 | 0.00 |
| 7.00 | 1.00 | 0.00 |
| 8.00 | 1.00 | 0.00 |
| 9.00 | 1.00 | 0.00 |
| 10.00 | 1.00 | 0.00 |
| 11.00 | 1.00 | 0.00 |
| 12.00 | 1.00 | 0.00 |

Monday, June 13, 1994

# GENERAL R&M PARAMETER REPORT



Aircraft : E003C Flight Hours : 3,476.00 Sorties : 368.00

| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 00000 | AIRCRAFT LEVEL | 1,135,500.5 | 113.55 | 0.70 | 9.32 | 2.55 | 0.88 | 2.62 |

Monday, June 13, 1994

# GENERAL R&M PARAMETER REPORT



Aircraft : E003C Flight Hours : 3,476.00 Sorties : 368.00

| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 41000 | AIR COND PRESS | 141,829.69 | 14.18 | 0.72 | 204.47 | 43.45 | 7.05 | 60.98 |

# GENERAL R&M PARAMETER REPORT



Aircraft : E003C Flight Hours : 3,476.00 Sorties : 368.00

| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 41A00 | BLEED AIR SYST | 94,936.71 | 9.49 | 0.65 | 695.20 | 119.86 | 10.53 | 115.87 |
| 41B00 | CABIN AIR COND SYS | 6,904.49 | 0.69 | 0.59 | 3,476.00 | 695.20 | 144.83 | 1,738.00 |
| 41C00 | CABIN PRESSURE SYS | 2,876.87 | 0.29 | 0.74 | 1,158.67 | 496.57 | 347.60 | 869.00 |
| 41D00 | AVION F A C SYS FW | 3,164.56 | 0.32 | 1.47 | 1,738.00 | 695.20 | 316.00 | 869.00 |
| 41E00 | AVION F A COOL SYS | 18,411.97 | 1.84 | 0.98 | 869.00 | 434.50 | 54.31 | 695.20 |
| 41F00 | AVION D T COOL SYS | 2,013.81 | 0.20 | 0.98 | | 1,158.67 | 496.57 | 3,476.00 |
| 41G00 | ANT PED COOL SYSTE | 3,452.24 | 0.35 | 0.54 | 3,476.00 | 869.00 | 289.67 | 1,738.00 |
| 41H00 | RADAR L C S | 4,027.62 | 0.40 | 0.59 | 3,476.00 | 434.50 | 248.29 | 695.20 |
| 41K00 | NESA WNDOW ANTI IC | 575.37 | 0.06 | 0.42 | | 1,738.00 | 1,738.00 | 3,476.00 |
| 41K00 | NESA WNDOW ANTI IC | 575.37 | 0.06 | 0.42 | | 1,738.00 | 1,738.00 | 3,476.00 |

Monday, June 13, 1994

# GENERAL R&M PARAMETER REPORT



Aircraft : E003C Flight Hours : 3,476.00 Sorties : 368.00

| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 41A00 | BLEED AIR SYST | 2,013.81 | 0.20 | 1.02 | 3,476.00 | 579.33 | 496.57 | 1,738.00 |
| 41AA0 | VALVE PRESSURE/FLO | 2,013.81 | 0.20 | 0.77 | 3,476.00 | 695.20 | 496.57 | 1,158.67 |
| 41AB0 | EXCHANGER HEAT | 2,301.50 | 0.23 | 1.20 | | | 434.50 | 1,738.00 |
| 41AC0 | VALVE SHUTOFF | 2,013.81 | 0.20 | 1.26 | 1,738.00 | 1,158.67 | 496.57 | 1,738.00 |
| 41AL0 | ENG-APU NEG PRESSV | 287.69 | 0.03 | 0.50 | | | 3,476.00 | |
| 41AV0 | VALVE ISOLATION | 1,150.75 | 0.12 | 1.31 | | | 869.00 | |
| 41AX0 | INSTR BLD AIR SYST | 48,906.79 | 4.89 | 0.62 | | 267.38 | 20.45 | 289.67 |
| 41AY0 | DUCTING | 35,097.81 | 3.51 | 0.72 | | 3,476.00 | 28.49 | 434.50 |
| 41B00 | CABIN AIR COND SYS | 863.06 | 0.09 | 0.58 | | 3,476.00 | 1,158.67 | |
| 41BA0 | VALVE FLOW CONT | 2,013.81 | 0.20 | 0.52 | 3,476.00 | 3,476.00 | 496.57 | 3,476.00 |
| 41BB0 | EXCHANGER HEAT | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41BC0 | MACHINE AIR CYCLE | 863.06 | 0.09 | 0.67 | | 3,476.00 | 1,158.67 | |
| 41BD0 | SEPARATOR MOISTURE | 2,301.50 | 0.23 | 0.91 | | 3,476.00 | 434.50 | |
| 41BF0 | REGLTR ZONE TRM PR | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41BP0 | ACTUATOR RAM AIR | 287.69 | 0.03 | 0.67 | | 3,476.00 | 3,476.00 | 3,476.00 |
| 41C00 | CABIN PRESSURE SYS | 1,150.75 | 0.12 | 1.19 | 3,476.00 | 869.00 | 869.00 | 3,476.00 |
| 41CA0 | VALVE OUTFLOW | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41CC0 | CONTROL UNIT PRES | 863.06 | 0.09 | 0.50 | 1,738.00 | 1,158.67 | 1,158.67 | 1,738.00 |
| 41CD0 | PANEL SELECTOR | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41CG0 | WIRING | 287.69 | 0.03 | 1.00 | | | 3,476.00 | 3,476.00 |
| 41D00 | AVION F A C SYS FW | 575.37 | 0.06 | 1.25 | | 3,476.00 | 1,738.00 | |
| 41DB0 | FAN REC AIR FWD | 863.06 | 0.09 | 2.00 | 3,476.00 | 1,158.67 | 1,158.67 | 1,158.67 |
| 41DD0 | VALVE OVERBOARD | 287.69 | 0.03 | 0.25 | | | 3,476.00 | |
| 41DL0 | CONTROLLER TEMP | 287.69 | 0.03 | 0.50 | | | 3,476.00 | |
| 41DS0 | VALVE SHUTOFF 4 IN | 863.06 | 0.09 | 1.02 | 3,476.00 | 3,476.00 | 1,158.67 | 3,476.00 |
| 41E00 | AVION F A COOL SYS | 863.06 | 0.09 | 1.04 | | 1,158.67 | 1,158.67 | |
| 41EB0 | VALVE AFT RAM AIR | 8,342.92 | 0.83 | 0.56 | | 3,476.00 | 119.86 | |
| 41EF0 | FAN REC AIR AFT | 1,150.75 | 0.12 | 2.06 | 3,476.00 | 3,476.00 | 869.00 | 3,476.00 |
| 41ES0 | DUCTING | 6,329.11 | 0.63 | 0.64 | | | 158.00 | 3,476.00 |
| 41EY0 | ELEC LOAD CONT UNI | 1,726.12 | 0.17 | 1.01 | 1,158.67 | 1,158.67 | 579.33 | 1,158.67 |
| 41F00 | AVION D T COOL SYS | 575.37 | 0.06 | 0.50 | | 1,738.00 | 1,738.00 | |
| 41FA0 | FAN ASSY 2 SPEED | 1,150.75 | 0.12 | 1.02 | | 3,476.00 | 869.00 | 3,476.00 |
| 41FB0 | VALVE FLOW CONTROL | 287.69 | 0.03 | 0.33 | | | 3,476.00 | |
| 41G00 | ANT PED COOL SYSTE | 863.06 | 0.09 | 0.83 | 3,476.00 | 3,476.00 | 1,158.67 | 3,476.00 |
| 41GA0 | ACTUATOR DOOR | 575.37 | 0.06 | 0.58 | | 1,738.00 | 1,738.00 | 3,476.00 |
| 41GE0 | FAN AXIAL VANE | 1,726.12 | 0.17 | 0.88 | | | 579.33 | |
| 41GH0 | DUCTING | 287.69 | 0.03 | 1.00 | | 3,476.00 | 3,476.00 | |
| 41H00 | RADAR L C S | 287.69 | 0.03 | 1.00 | | 3,476.00 | 3,476.00 | |
| 41HA0 | E G W SYSTEM | 1,438.43 | 0.14 | 1.20 | | | 695.20 | |
| 41KA0 | CONTROL WINDOW HEA | 575.37 | 0.06 | 0.42 | | 1,738.00 | 1,738.00 | 3,476.00 |

Monday, June 13, 1994

# GENERAL R&M PARAMETER REPORT



| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 41KA0 | CONTROL WINDOW HEA | 575.37 | 0.06 | 0.42 | | 1,738.00 | 1,738.00 | 3,476.00 |

Monday, June 13, 1994

# GENERAL R&M PARAMETER REPORT



Aircraft : E003C Flight Hours : 3,476.00 Sorties : 368.00

| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 41A00 | BLEED AIR SYST | 2,013.81 | 0.20 | 1.02 | 3,476.00 | 579.33 | 496.57 | 1,738.00 |
| 41AA0 | VALVE PRESSURE/FLO | 1,726.12 | 0.17 | 0.69 | 3,476.00 | 869.00 | 579.33 | 1,738.00 |
| 41AAA | VALVE ASSY | 287.69 | 0.03 | 1.33 | | 3,476.00 | 3,476.00 | 3,476.00 |
| 41AB0 | EXCHANGER HEAT | 2,301.50 | 0.23 | 1.20 | | | 434.50 | 1,738.00 |
| 41AC0 | VALVE SHUTOFF | 2,013.81 | 0.20 | 1.26 | 1,738.00 | 1,158.67 | 496.57 | 1,738.00 |
| 41AL0 | ENG-APU NEG PRESSV | 287.69 | 0.03 | 0.50 | | | 3,476.00 | |
| 41AV0 | VALVE ISOLATION | 1,150.75 | 0.12 | 1.31 | | | 869.00 | |
| 41AX0 | INSTR BLD AIR SYST | 48,619.10 | 4.86 | 0.62 | | 289.67 | 20.57 | 289.67 |
| 41AX9 | NOC | 287.69 | 0.03 | 0.50 | | 3,476.00 | 3,476.00 | |
| 41AY0 | DUCTING | 34,810.13 | 3.48 | 0.74 | | 3,476.00 | 28.73 | 496.57 |
| 41AY9 | NOC | 287.69 | 0.03 | 0.50 | | | 3,476.00 | 3,476.00 |
| 41B00 | CABIN AIR COND SYS | 863.06 | 0.09 | 0.58 | | 3,476.00 | 1,158.67 | |
| 41BA0 | VALVE FLOW CONT | 2,013.81 | 0.20 | 0.52 | 3,476.00 | 3,476.00 | 496.57 | 3,476.00 |
| 41BB0 | EXCHANGER HEAT | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41BC0 | MACHINE AIR CYCLE | 863.06 | 0.09 | 0.67 | | 3,476.00 | 1,158.67 | |
| 41BD0 | SEPARATOR MOISTURE | 1,726.12 | 0.17 | 0.92 | | 3,476.00 | 579.33 | |
| 41BDA | COALESCER | 575.37 | 0.06 | 1.75 | | | 1,738.00 | |
| 41BF0 | REGLTR ZONE TRM PR | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41BP0 | ACTUATOR RAM AIR | 287.69 | 0.03 | 0.67 | | 3,476.00 | 3,476.00 | 3,476.00 |
| 41C00 | CABIN PRESSURE SYS | 1,150.75 | 0.12 | 1.19 | 3,476.00 | 869.00 | 869.00 | 3,476.00 |
| 41CA0 | VALVE OUTFLOW | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41CC0 | CONTROL UNIT PRES | 575.37 | 0.06 | 0.50 | 3,476.00 | 1,738.00 | 1,738.00 | 3,476.00 |
| 41CCD | NETWK AY RATE AMP | 287.69 | 0.03 | 0.50 | 3,476.00 | 3,476.00 | 3,476.00 | 3,476.00 |
| 41CD0 | PANEL SELECTOR | 287.69 | 0.03 | 1.00 | | | 3,476.00 | |
| 41D00 | AVION F A˘C SYS FW | 575.37 | 0.06 | 1.25 | | 3,476.00 | 1,738.00 | |
| 41DB0 | FAN REC AIR FWD | 863.06 | 0.09 | 2.00 | 3,476.00 | 1,158.67 | 1,158.67 | 1,158.67 |
| 41DD0 | VALVE OVERBOARD | 287.69 | 0.03 | 0.25 | | | 3,476.00 | |
| 41DL0 | CONTROLLER TEMP | 287.69 | 0.03 | 0.50 | | | 3,476.00 | |
| 41DS0 | VALVE SHUTOFF 4 IN | 863.06 | 0.09 | 1.02 | 3,476.00 | 3,476.00 | 1,158.67 | 3,476.00 |
| 41E00 | AVION F A COOL SYS | 863.06 | 0.09 | 1.04 | | 1,158.67 | 1,158.67 | |
| 41EB0 | VALVE AFT RAM AIR | 8,342.92 | 0.83 | 0.56 | | 3,476.00 | 119.86 | |
| 41EF0 | FAN REC AIR AFT | 1,150.75 | 0.12 | 2.06 | 3,476.00 | 3,476.00 | 869.00 | 3,476.00 |
| 41ES0 | DUCTING | 6,329.11 | 0.63 | 0.64 | | | 158.00 | 3,476.00 |
| 41EY0 | ELEC LOAD CONT UNI | 1,726.12 | 0.17 | 1.01 | 1,158.67 | 1,158.67 | 579.33 | 1,158.67 |
| 41F00 | AVION D T COOL SYS | 575.37 | 0.06 | 0.50 | | 1,738.00 | 1,738.00 | |
| 41FA0 | FAN ASSY 2 SPEED | 1,150.75 | 0.12 | 1.02 | | 3,476.00 | 869.00 | 3,476.00 |
| 41FB0 | VALVE FLOW CONTROL | 287.69 | 0.03 | 0.33 | | | 3,476.00 | |
| 41G00 | ANT PED COOL SYSTE | 863.06 | 0.09 | 0.83 | 3,476.00 | 3,476.00 | 1,158.67 | 3,476.00 |
| 41GA0 | ACTUATOR DOOR | 575.37 | 0.06 | 0.58 | | 1,738.00 | 1,738.00 | 3,476.00 |
| 41GE0 | FAN AXIAL VANE | 1,726.12 | 0.17 | 0.88 | | | 579.33 | |

# GENERAL R&M PARAMETER REPORT



| Wuc | Nomenclature | 10**6 | 10**2 | Mttr | Mtbur | Mtbm | Mtbma | Mtbf |
|---|---|---|---|---|---|---|---|---|
| 41GH0 | DUCTING | 287.69 | 0.03 | 1.00 | | 3,476.00 | 3,476.00 | |
| 41H00 | RADAR L C S | 287.69 | 0.03 | 1.00 | | 3,476.00 | 3,476.00 | |
| 41HA0 | E G W SYSTEM | 575.37 | 0.06 | 2.00 | | | 1,738.00 | |
| 41HA1 | CYL NITROGEN PRES | 287.69 | 0.03 | 0.50 | | | 3,476.00 | |
| 41HAA | PUMP UNIT CENTRIF | 575.37 | 0.06 | 0.75 | | | 1,738.00 | |
| 41HB3 | IND RESISTIVITY | 287.69 | 0.03 | 0.67 | | 3,476.00 | 3,476.00 | 3,476.00 |
| 41HBW | VALVE GATE 231295 | 287.69 | 0.03 | 1.00 | | 3,476.00 | 3,476.00 | 3,476.00 |
| 41HCB | COOLER GROUND | 287.69 | 0.03 | 0.38 | | 3,476.00 | 3,476.00 | |
| 41HDF | PNL ASSY LCS COUT | 287.69 | 0.03 | 0.67 | | 3,476.00 | 3,476.00 | 3,476.00 |
| 41KA0 | CONTROL WINDOW HEA | 575.37 | 0.06 | 0.42 | | 1,738.00 | 1,738.00 | 3,476.00 |
| 41KA0 | CONTROL WINDOW HEA | 575.37 | 0.06 | 0.42 | | 1,738.00 | 1,738.00 | 3,476.00 |

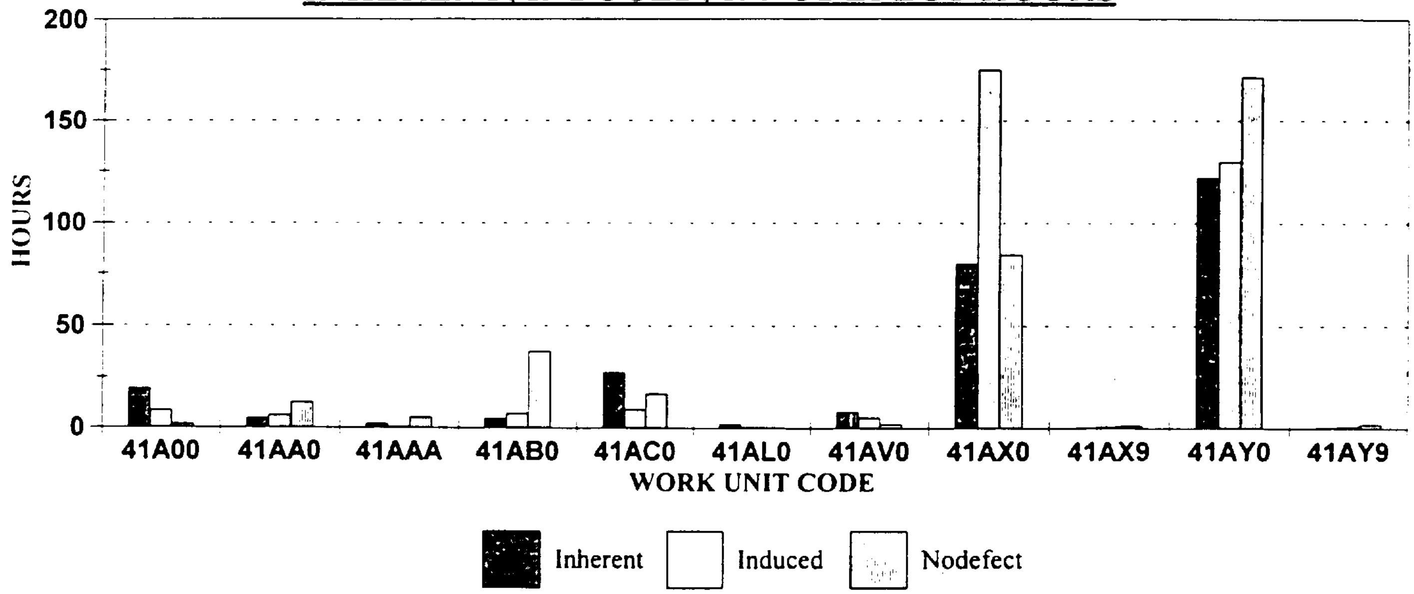
# SUMMARIZED MDC DATA

Today : 6/13/94

Aircraft : E003C Fh : 3,476.00 Sorties : 368.00

| Year | Ind | Wuc | Nomenclature | Inherent | Induced | Nodefect | Total_hrs |
|---|---|---|---|---|---|---|---|
| 91 | 5 | 41A00 | BLEED AIR SYST | 19.50 | 8.50 | 1.75 | 29.75 |
| 91 | 5 | 41AA0 | VALVE PRESSURE/FLO | 4.50 | 6.00 | 12.50 | 23.00 |
| 91 | 5 | 41AAA | VALVE ASSY | 2.00 | 0.50 | 5.00 | 7.50 |
| 91 | 5 | 41AB0 | EXCHANGER HEAT | 4.50 | 7.00 | 37.75 | 49.25 |
| 91 | 5 | 41AC0 | VALVE SHUTOFF | 27.50 | 9.17 | 17.00 | 53.67 |
| 91 | 5 | 41AL0 | ENG-APU NEG PRESSV | 2.00 | 0.50 | 0.50 | 3.00 |
| 91 | 5 | 41AV0 | VALVE ISOLATION | 8.00 | 5.25 | 2.00 | 15.25 |
| 91 | 5 | 41AX0 | INSTR BLD AIR SYST | 80.00 | 175.00 | 84.42 | 339.42 |
| 91 | 5 | 41AX9 | NOC | 0.50 | 1.00 | 1.42 | 2.92 |
| 91 | 5 | 41AY0 | DUCTING | 122.00 | 129.83 | 171.58 | 423.42 |
| 91 | 5 | 41AY9 | NOC | 0.50 | 1.00 | 2.00 | 3.50 |

## INHERENT, INDUCED, & NODEFECT HOURS



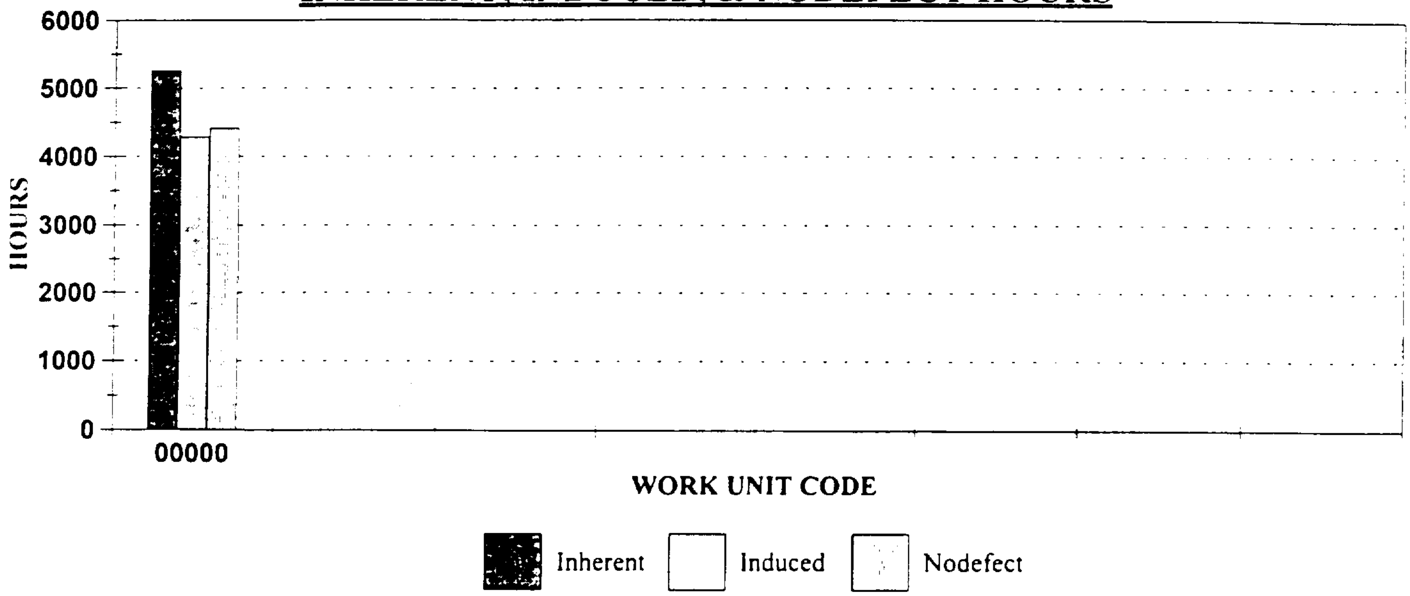
# SUMMARIZED MDC DATA

Today : 6/13/94

Aircraft : E003C Fh : 3,476.00 Sorties : 368.00

| Year | Ind | Wuc | Nomenclature | Inherent | Induced | Nodefect | Total_hrs |
|---|---|---|---|---|---|---|---|
| 91 | 1 | 00000 | AIRCRAFT LEVEL | 5,253.42 | 4,288.42 | 4,415.08 | 13,956.92 |

## INHERENT, INDUCED, & NODEFECT HOURS



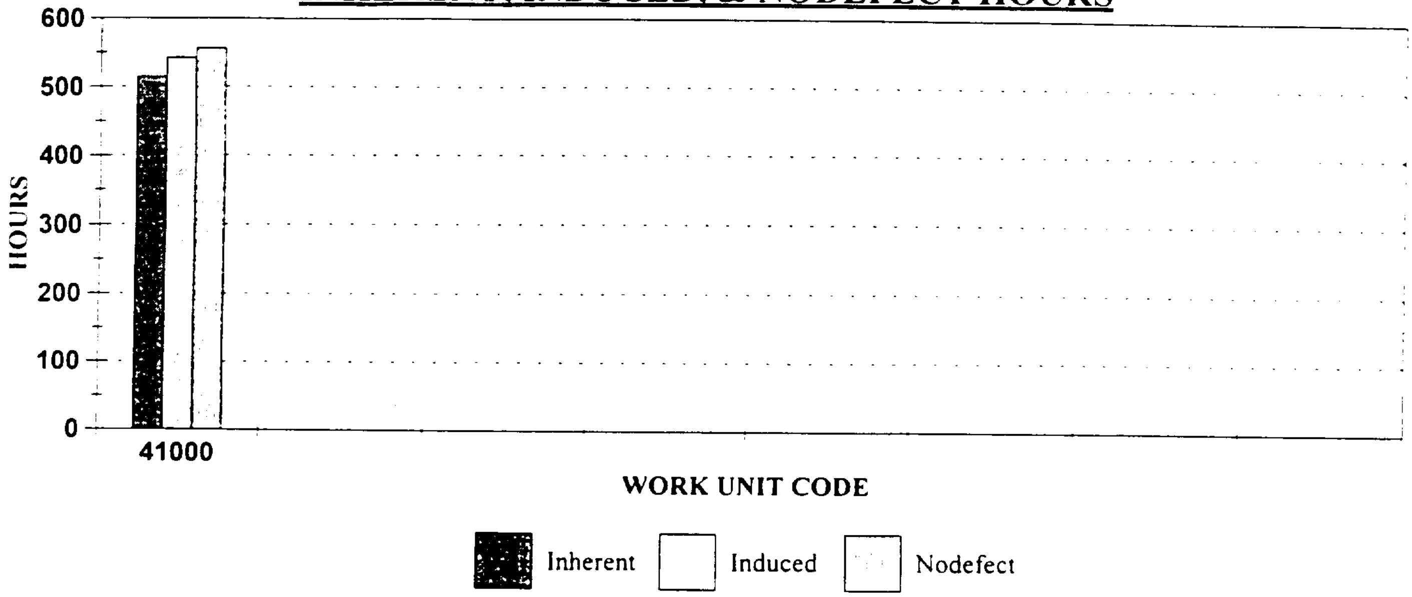
# SUMMARIZED MDC DATA

Today : 6/13/94

Aircraft : E003C    Fh : 3,476.00    Sorties : 368.00

| Year | Ind | Wuc | Nomenclature | Inherent | Induced | Nodefect | Total_hrs |
|---|---|---|---|---|---|---|---|
| 91 | 2 | 41000 | AIR COND PRESS | 514.75 | 542.67 | 556.17 | 1,613.58 |

## INHERENT, INDUCED, & NODEFECT HOURS



# SUMMARIZED MDC DATA

Today : 6/13/94

Aircraft : E003C    Fh : 3,476.00    Sorties : 368.00

| Year | Ind | Wuc | Nomenclature | Inherent | Induced | Nodefect | Total_hrs |
|---|---|---|---|---|---|---|---|
| 91 | 3 | 41A00 | BLEED AIR SYST | 285.75 | 347.75 | 338.17 | 971.67 |

## INHERENT, INDUCED, & NODEFECT HOURS