Lockheed Martin
Missiles & Space- Huntsville

LMSC P038190

605 f.

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

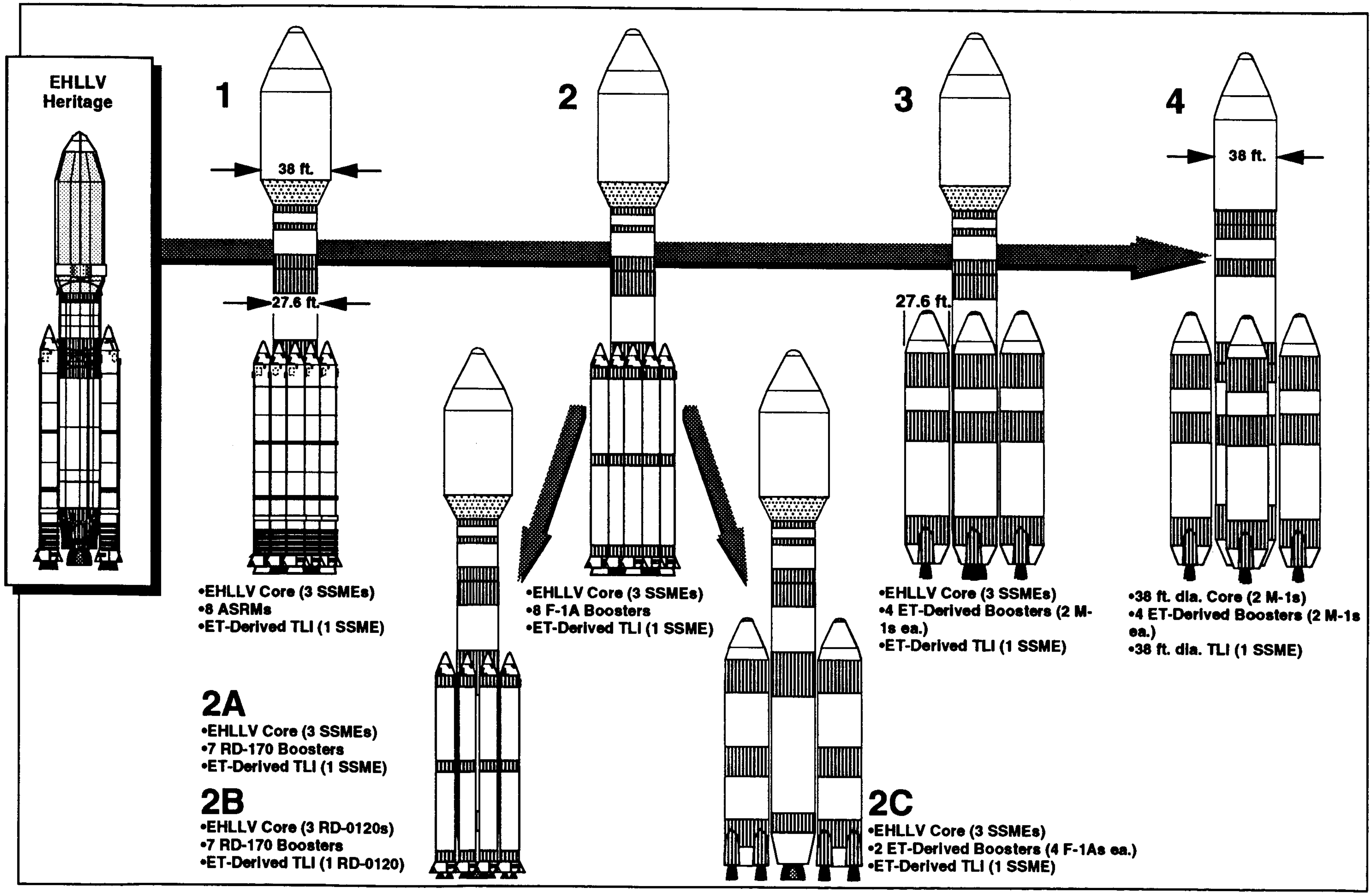
Table of Contents

Single Launch Lunar Launch Vehicle Configuration Options

Lockheed, Aerojet, ECON

Parallel-Burn Lunar Launch Vehicle Configuration Options

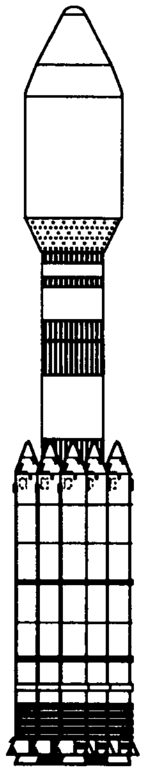
Lunar Mission, Parallel-Burn Configuration Options



J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options

1



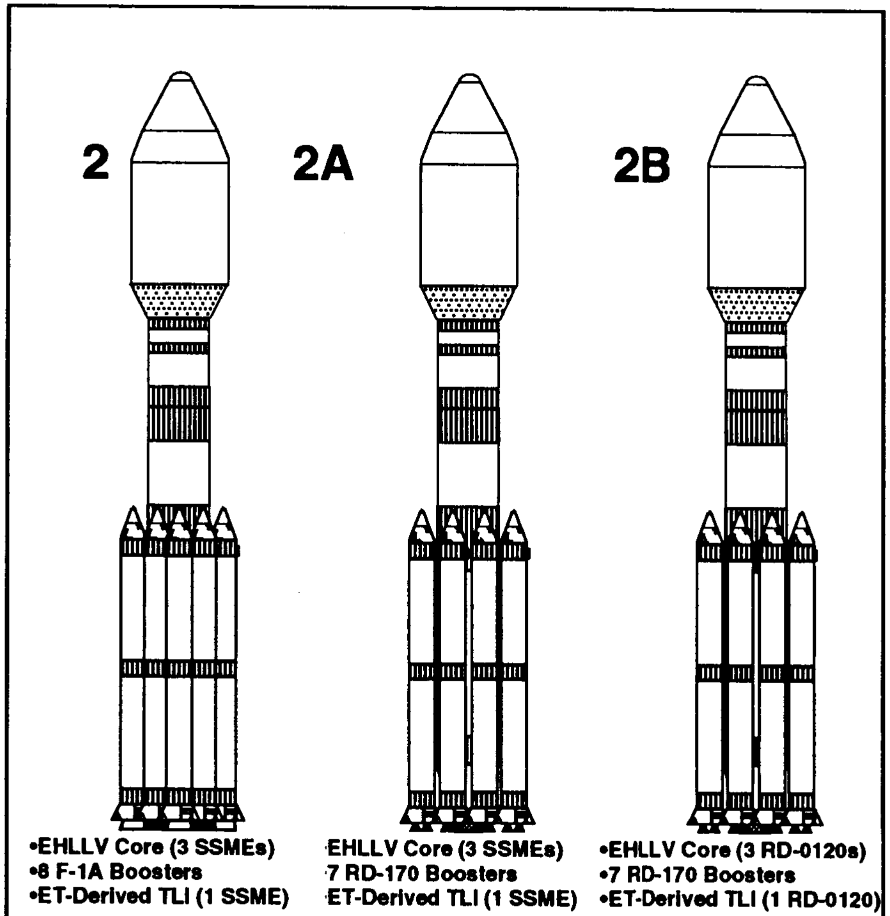
•EHLLV Core (3 SSMEs)
•8 ASRMs
•ET-Derived TLI (1 SSME)

Pros:
- No new engines
- Booster simplicity
- Low DDT&E

Cons:
- Cannot lift 93 MT
- Booster stacking complexity
- Environmental issues
- Acceleration loads vs. core vehicle beef-up
- Separation dynamics
- Vehicle & ground load path complexity
- Mars evolution questionable
- Hammerhead vs. inert wt. hit

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options

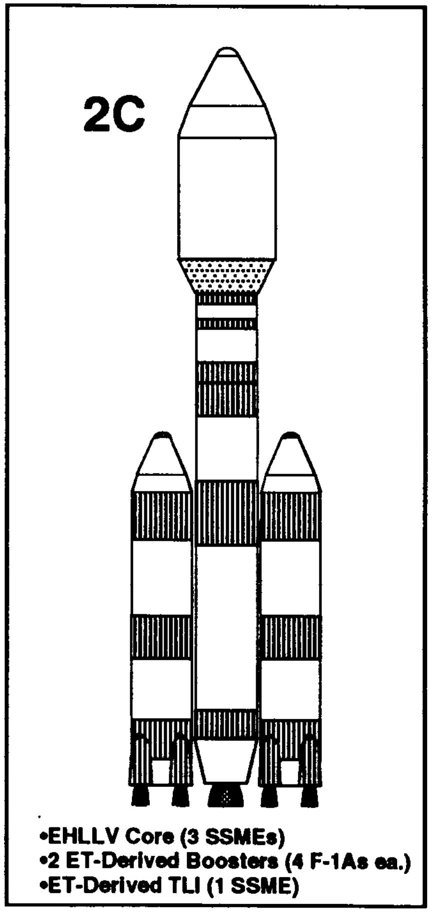


Pros:
- Core commonality
- Booster design simplicity
- Booster test & check-out simplicity
- Booster unit cost savings (larger lot buy)
- ELV family evolution from booster

Cons:
- New RP boosters engine
- Hammerhead vs. inert wt. hit
- Booster stacking
- Programmatic risk of CIS engines
- QA/QC uncertainties of CIS engines
- Mars evolution questionable
- Vehicle & ground load path complexity
- High FMEA/CIL count
- FRF feasibility questionable

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options

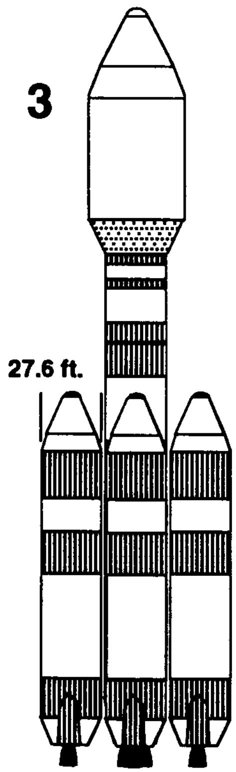


Pros:
- Booster commonality with core
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Simple vehicle & ground load paths
- Fewer vehicle FMEAs/CILs

Cons:
- Booster MPS feed complexity
- Potentially higher booster unit cost (smaller lot buy)
- Hammerhead vs. core inert wt. hit

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options



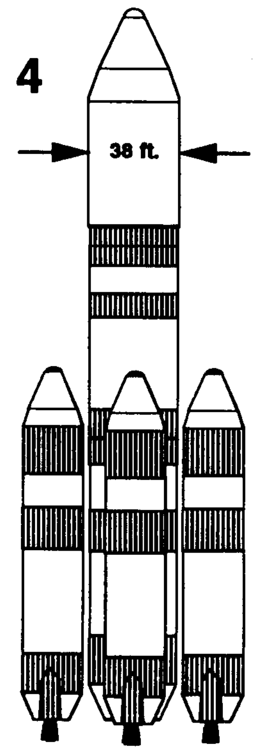
•EHLLV Core (3 SSMEs)
•4 ET-Derived Boosters (2 M-1s ea.)
•ET-Derived TLI (1 SSME)

Pros:
- Booster commonality with core
- Common booster/core propellants
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Fewer vehicle FMEAs/CILs
- No environmental impacts
- Potentially lower booster unit cost (larger lot buy)

Cons:
- New booster engine
- Booster leak potential
- Lower booster density impulse
- Hammerhead vs. core inert wt. hit
- Vehicle & ground load path complexity

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options



•38 ft. dia. Core (2 M-1s)
•4 ET-Derived Boosters (2 M-1s ea.)
•38 ft. dia. TLI (1 SSME)

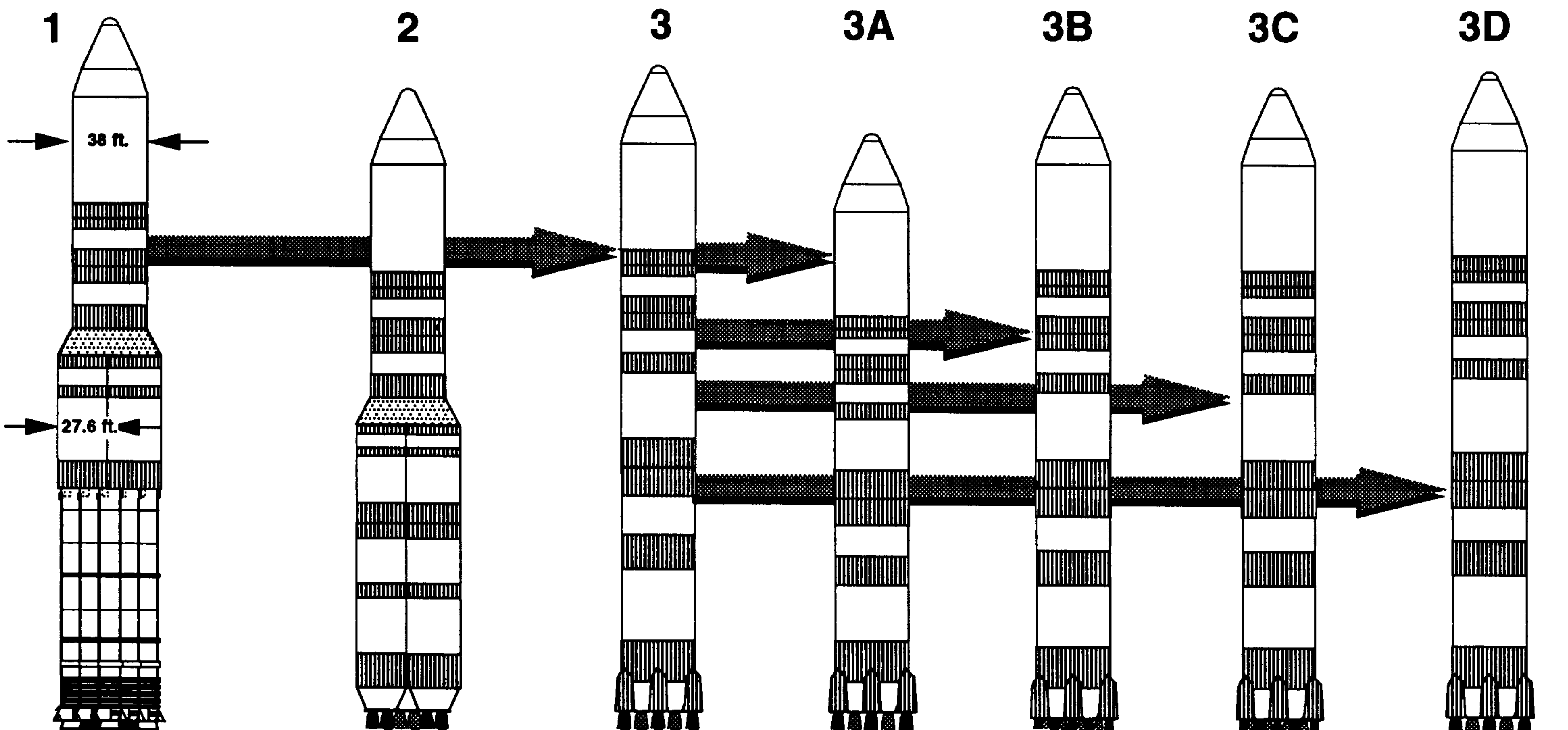
Pros:
- No hammerhead
- Common booster/core propellants
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Fewer vehicle FMEAs/CILs
- No environmental impacts
- Potentially lower booster unit cost (larger lot buy)

Cons:
- New core vs. booster commonality with ET
- New booster/core engine
- Booster leak potential
- Lower booster density impulse
- Vehicle & ground load path complexity

J. B. McCurry
205-722-4509

Series-Burn Lunar Launch Vehicle Configuration Options

Lunar Mission
Series-Burn Configuration Options

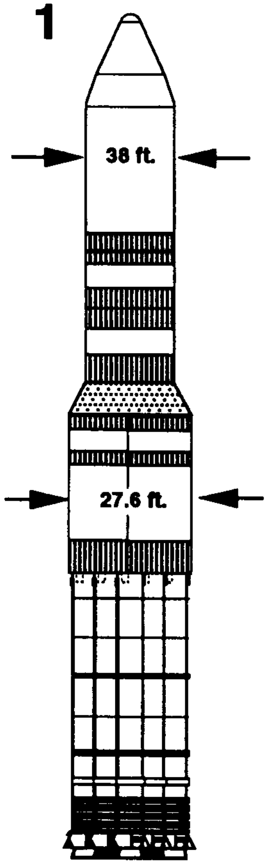


1	2	3	3A	3B	3C	3D
•TBD ASRMs First Stage •3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.) •38 ft. Dia. 3rd Stage (1 SSME)	•3 ET-Derived Tanks First Stage (3 F-1As ea.) •3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.) • 38 ft. Dia. 3rd Stage (1 SSME)	•38 ft. Dia. 1st Stage (7 F-1As) •38 ft. Dia. 2nd Stage (5 SSMEs) •38 ft. Dia. TLI Stage (1 SSME)	•38 ft. Dia. 1st Stage Multi-Cell (7 F-1As) •38 ft. Dia. 2nd Stage Multi-Cell (5 SSMEs) •38 ft. Dia. TLI Stage Multi-Cell (1 SSME)	•38 ft. Dia. 1st Stage (7 RD-170s) •38 ft. Dia. 2nd Stage (5 SSMEs) •38 ft. Dia. TLI Stage (1 SSME)	•38 ft. Dia. 1st Stage (7 RD-170s) •38 ft. Dia. 2nd Stage (5 RD-0120s) •38 ft. Dia. TLI Stage (1 RD-0120)	•38 ft. Dia. 1st Stage (TBD M-1s) •38 ft. Dia. 2nd Stage (TBD M-1s) •38 ft. Dia. TLI Stage (1 SSME)

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

1



- TBD ASRMs First Stage
- 3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
- 38 ft. Dia. 3rd Stage (1 SSME)

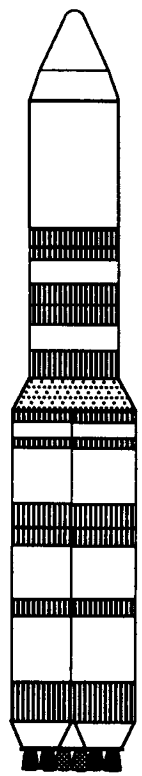
Pros:
- No hammerhead
- No new engines
- Booster simplicity
- Existing stage elements vs. attach hardware development

Cons:
- New TLI tank design
- Stage stacking complexity
- Interstage complexity
- TLI stage tank complexity (probably multiple tanks)
- Environmental issues
- Acceleration loads vs. core vehicle beef-up
- Vehicle & ground load path complexity
- Many SSMEs
- Mars evolution questionable

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

2



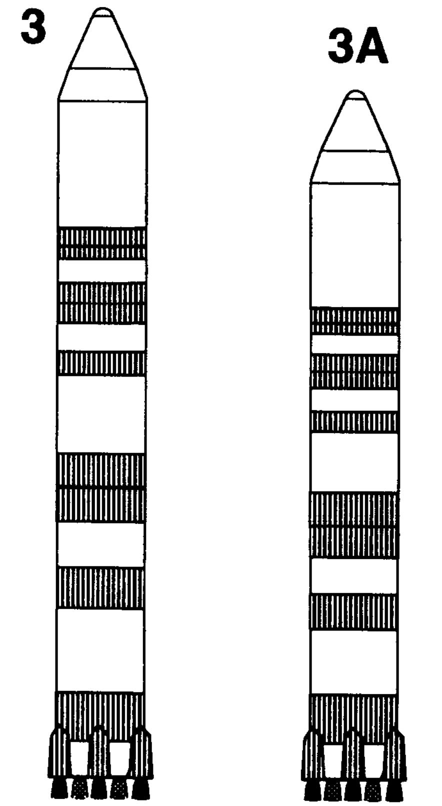
- 3 ET-Derived Tanks First Stage (3 F-1As ea.)
- 3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
- 38 ft. Dia. 3rd Stage (1 SSME)

Pros:
- No hammerhead
- First/Second stage tank commonality with ET
- Capable of testing each stage element independently
- Existing stage elements vs. attach hardware development
- Lessened environmental issues
- Mars evolution potential

Cons:
- New TLI tank design
- Stage stacking complexity
- Interstage complexity
- TLI stage tank complexity (probably multiple tanks)
- Vehicle load path complexity
- Many SSMEs

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

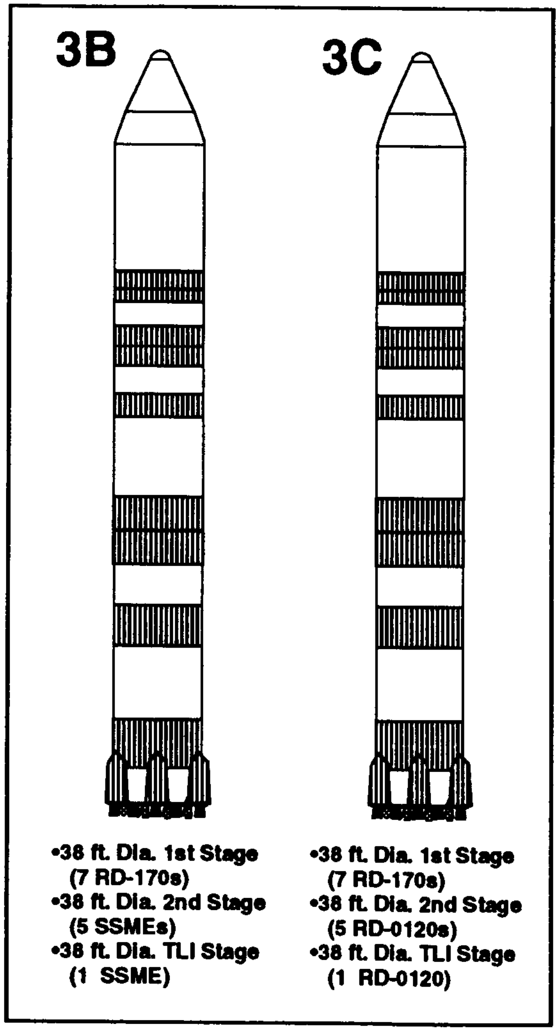


Pros:
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- Lessened environmental issues
- Mars evolution potential
- Less weld length for multi-cell
- Fewer stage pressurization subsystems
- No first stage leak issue

Cons:
- New stage design
- Many SSMEs
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)
- Many SSMEs
- Complicated first/second stage MPS feed subsystems (unless use multi-cell)

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options



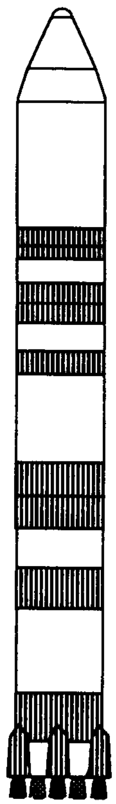
Pros:
- Shorter first stage design (better Isp)
- Potentially cheaper engines & no development cost
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- Lessened environmental issues
- Mars evolution potential
- Fewer stage pressurization subsystems
- No first stage leak issue

Cons:
- Foreign engines & RP source
- New stage design
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)
- Complicated first/second stage MPS feed subsystems

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

3D



•38 ft. Dia. 1st Stage (TBD M-1s)
•38 ft. Dia. 2nd Stage (TBD M-1s)
•38 ft. Dia. TLI Stage (1 SSME)

Pros:

- Fewer engines for first/second stage (higher thrust)
- Simpler MPS feed subsystems
- Propellant commonality for all stages
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- No environmental issues
- Mars evolution potential
- Fewer stage pressurization subsystems

Cons:

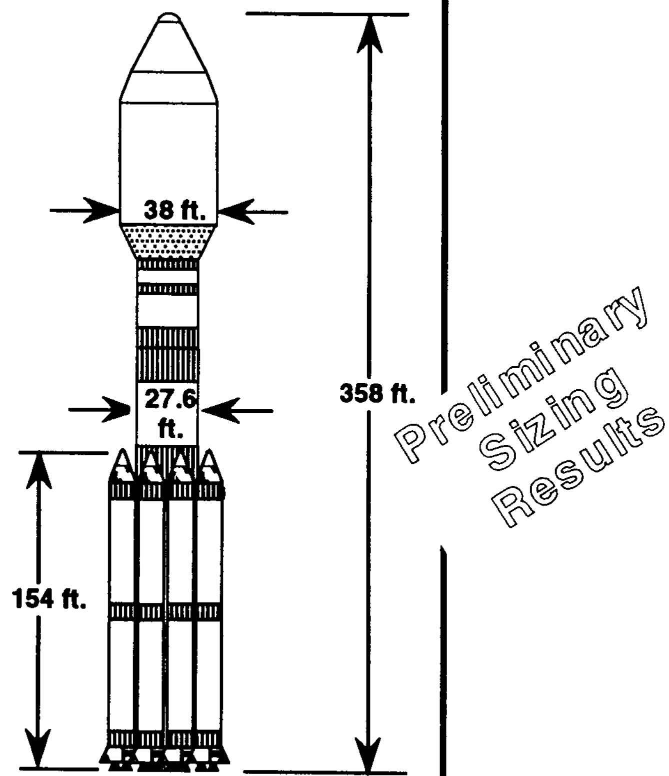
- Poor first stage density impulse (larger stage)
- Greater leak potential
- New stage design
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)

J. B. McCurry
205-722-4509

Current Reference Lunar Launch Vehicle Configurations

Preliminary Data

Parallel Burn FLO Vehicle 2



- EHLLV Core (3 SSMEs)
- 8 F-1A Boosters
- EHLLV-Derived TLI (1 SSME)

Payload:	205,000 lbm (93 MT)
Final Position:	TLI

GLOW: 11,094,236 lbm

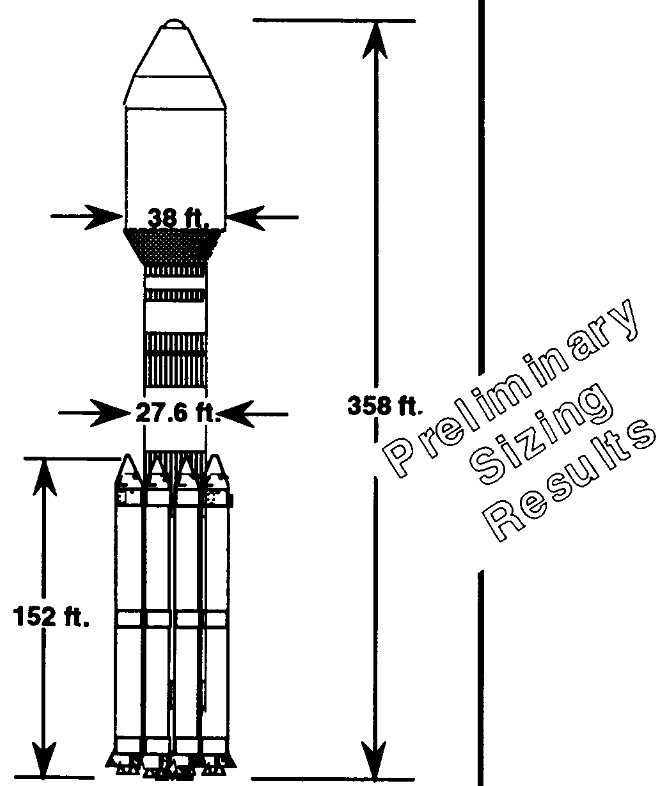
CORE:
- Inert Mass: 153,935 lbm
- Usable Propellant Mass: 1,678,840 lbm
- Propellant Type: LOX/LH2
- Engine Type/No.: SSME/3
- Diameter: 27.6 ft
- Shroud Jettison Mass: 28 K lbm

BOOSTER:
- Number/Type: 8 Single F-1A Boosters
- Inert Mass: 68,341 lbm
- Usable Propellant: 950,000 lbm
- Propellant Type: LOX/RP-1
- Engine Type/No.: F-1A/1
- Diameter: 15.0 ft

TLI Stage:
- Inert Mass: 61,157 lbm
- Usable Propellant Mass: 600,000 lbm
- Propellant Type: LOX/LH2
- Engine Type/No.: SSME/1
- Diameter: 27.6 ft

J. B. McCurry
(205) 722-4509

Preliminary Data Parallel Burn FLO Vehicle 2A



- EHLLV Core (3 SSMEs)
- 7 RD-170 Boosters
- EHLLV-Derived TLI (1 SSME)

Payload: Final	205,000 lbm (93MT)
Position:	TLI

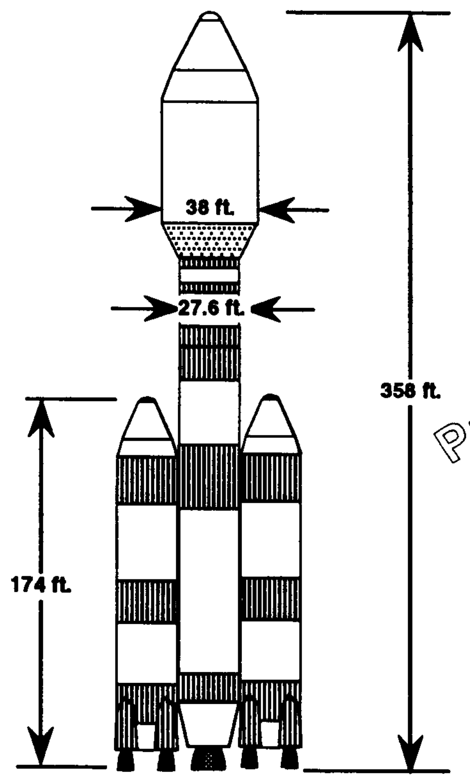
GLOW: 9,871,217 lbm
CORE:
Inert Mass: 153,335 lbm
Usable Propellant Mass: 1,678,840 lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/3
Diameter: 27.6 ft
Shroud Jettison Mass: 28 K lbm

BOOSTER:
Number/Type: 7 Single RD-170 Boosters
Inert Mass: 65,539 lbm
Usable Propellant: 800,000 lbm
Propellant Type: LOX/SYN10
Engine Type/No.: RD-170/1
Diameter: 14.0 ft

TLI Stage:
Inert Mass: 81,494 lbm
Usable Propellant Mass: 600,000 lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/1
Diameter: 27.6 ft

J. B. McCurry
(205) 722-4509

Preliminary Data
Parallel Burn FLO Vehicle 2C



•EHLLV Core (3 SSMEs)
•2 Boosters with 4 F-1 Each
•EHLLV-Derived TLI (1 SSME)

Preliminary Sizing Results

Payload:	205,000 lbm (93 MT)
Final Position:	TLI

GLOW: 11,056,511 lbm

CORE:
- Inert Mass: 153,935 lbm
- Usable Propellant Mass: 1,678,840 lbm
- Propellant Type: LOX/LH2
- Engine Type/No.: SSME/3
- Diameter: 27.6 ft
- Shroud Jettison Mass: 28 K lbm

BOOSTER:
- Number/Type: 2 Boosters with 4 F-1As Each
- Inert Mass: 254,501 lbm
- Usable Propellant: 3,800,000 lbm
- Propellant Type: LOX/RP-1
- Engine Type/No.: F-1A/4
- Diameter: 27.6 ft

TLI Stage:
- Inert Mass: 61,157 lbm
- Usable Propellant Mass: 600,000 lbm
- Propellant Type: LOX/LH2
- Engine Type/No.: SSME/1
- Diameter: 27.6 ft

J. B. McCurry
(205) 722-4509

Advanced Transportation System Studies
Technical Area 2
Heavy Lift Launch Vehicle Development

Contract Status Briefing

Given
at the
Marshall Space Flight Center
27 January 1993

Lockheed, Aerojet, ECON

Agenda

1. FY92/FY93 Accomplishments
2. General Sizing Philosophy
3. Previous Lunar Configurations
4. Propulsion Options
5. Structures Options
6. 50/80K Sizing
7. 50/80K Performance
8. Cost
9. Ground Operations
10. Vehicle Health Management
11. Technology Priorities

J. B. McCurry
205-722-4509

1. FY92/FY93 Accomplishments

TA-2 Accomplishments for FY92

- Parametric sizing of nine three-stage monolithic (constant stage diameter) vehicles using various propulsion options was completed
- A concept of clustering several Space Shuttle-derived External Tanks together for each stage of a candidate three-stage lunar HLLV was developed as a means to promote minimum design, development, test, and engineering costs
- A cost analysis benchmark was developed for a candidate monolithic HLLV concept as a calibration point to compare with MSFC in-house cost modeling
- An assessment of launch processing requirements was completed for candidate Shuttle-derived HLLV concepts that have been identified by the Space Station Assembly "super red team", in response to an action item from the red team
- A contract kick-off meeting was held with Aerojet personnel with a focus on the identification of alternative HLLV propulsion options; of prime interest being the LOX/LH2 M-1 engine that was of F-1 class thrust
- A two-day concurrent engineering brainstorming meeting was held to identify, categorize, and rank HLLV design goals that would enable the First Lunar Outpost mission requirements; manufacturing, structures, loads, thermodynamics, vehicle design, performance assessment, ground operations, and cost modeling were represented
- Design goals were identified and ranked for three primary HLLV design approaches: minimum DDT&E cost, minimum recurring cost, and minimum programmatic risk
- Shuttle Orbiter wing load issues regarding use of liquid rocket boosters of various diameters were identified and provided to the customer per a request from NASA Headquarters, with implication to use of SEI-derived boosters
- A concurrent engineering approach was drafted for the definition and analysis of vehicle health management concepts for candidate HLLV concepts
- An ascent trajectory simulation was developed of the Commonwealth of Independent States Energia HLLV and benchmarked against MSFC in-house assessments; Energia's capability to provide booster and core engine-out performance was parametrically ascertained and appeared to account for, in conjunction with engine start-up modeling, previously seen discrepancies in payload capability versus "advertised" values published in literature

James B. McCurry
(205) 722-4509

TA-2 Accomplishments for FY92 (Concluded)

- Six series-burn and six parallel-burn candidate HLLV concepts have been identified for detailed sizing, performance assessments, ground processing issues, and cost assessments; the configurations include both EHLLV-derived and "clean-sheet" monolithic concepts that stress a minimum DDT&E approach
- Launch site operations evaluations were performed for the Single and Dual-Launch FLO scenarios, per a request from the NASA Headquarters "super red team"; included in the assessment were vehicle processing timelines, processing scenarios, mixed-fleet schedules, and launch site issues/concerns
- Four mixed-fleet launch processing scenarios were assessed covering NLS, Space Shuttle, and Lunar program launch manifests
- A detailed review of the FLO Preliminary Operations Concept Document (ExPO-controlled document) was performed, with red-lines being provided to the cognizant ExPO representative
- A two-day question/answer session with Dr. Boris Gubanov, principal designer of Energia, and his sponsor, Jerry Thomson of Aerojet, was attended and a detailed list of CIS vehicle design, performance, and operations questions were provided to Dr. Gubanov; the meeting was sponsored by LSOC and Aerojet
- A Propulsion Synergy Group QFD session on launch vehicle propulsion requirements was attended with the TA-2 technical monitor; a status of candidate FLO HLLV concepts was given and propulsion requirements inputs were provided to the group
- Detailed sizing and performance assessments were performed on ET-derived parallel-burn HLLV configurations that used boosters having the same diameter as the Shuttle ET, per direction from G. Austin/PT-01
- A simulation was developed and analyses performed on HLLV tower drift requirements during initial vertical rise versus similar requirements defined for Saturn V; undispersed and Monte Carlo assessments of influences from atmospheric and performance dispersions were assessed (the largest dispersion effect being thrust misalignment) and a minimum distance from the vehicle to the mobile launch tower was defined
- The preliminary FLO HLLV Subteam Status Report was assembled and edited for the Subteam's lead, G. Austin/PT-01
- Four FLO HLLV Subteam technical interchange meetings were supported; one of which was hosted at the LSOC facility

James B. McCurry
(205) 722-4509

TA-2 FY93 Accomplishments Through 1/93

- EHLLV-derived lunar configuration performance assessments made with down-sized TLI stage propellant load (600K lbm) to improve staging T/W for 8 single-F-1 boosters and 7 RD-170 boosters
  - -- 93 mt achievable with reasonable ascent trajectory for 3-SSME core vehicle
  - -- Sizing equations indicated 800K TLI stage preferable but resulted in low staging T/W and high alpha profile
  - -- Work halted prior to final optimization of TLI stage, due to redirection on 50K vehicle

- Ground operations planning document had been mocked up by LSOC and should be completed soon
  - -- Document is configuration-independent

- Work is proceeding on identification of VHM requirements for manufacturing-through-ground operations
  - -- Presentation prepared on approach for VHM technology bridging (given to F. Huffaker/PT-01)
  - -- Complete flow identified for manufacturing-through-ground operations
  - -- LH2 prevalve VHM demo task identified with quantifiable operations savings

- First-cut sizing of two-stage 50K vehicles completed over range of diameters
  - -- Minimum dry weight and minimum GLOW solutions identified
    - •Rubber F-1 first stage/rubber SSME second stage
    - •Rubber STME first stage/rubber SSME second stage
    - •"Actual" F-1A first stage/actual SSME second stage (100 %RPL throttles throughout)
  - -- Conceptual hybrid first stage motors sized by Aerojet for ET-derived and 17 ft. diameters
    - •ET-derived motor has unacceptably low l/d ratio (literally a wafer-sized motor)
  - -- Resizing to be performed for best stage throttle profile for Qbar/G limiting

- ECON completing pitch on "what DDT&E is good DDT&E"

James B. McCurry
(205) 722-4509

2. General Sizing Philosophy

General Sizing Philosophy

- Minimize number of engines per stage while striving for "reasonable" stage thrust-to-weight ratio
  --Simplifies propellant feed subsystem design
  --Helps to minimize stage structural mass
  --Helps to minimize stage unit cost
  --Helps to maximize stage reliability and lower FMEA/CIL items

- Minimize stage dry mass
  --Helps to minimize stage unit cost

- Maximize stage-to-stage hardware design commonality
  --Helps to minimize stage unit cost and manufacturing nonrecurring cost

- Seek to have no more than one "new" engine design for a given vehicle configuration
  --Helps to minimize DDT& E costs and programmatic risk

- Consider technologies that would be available to support a 2005 first launch date

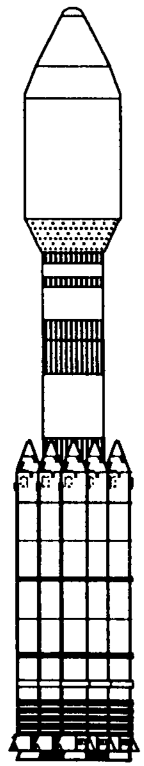
- Perform initial sizing sensitivity assessment of minimum-GLOW designs versus minimum-dry-weight solutions

J. B. McCurry
205-722-4509

3. Previous Lunar Configurations

Lunar Mission
Parallel-Burn Configuration Options

1



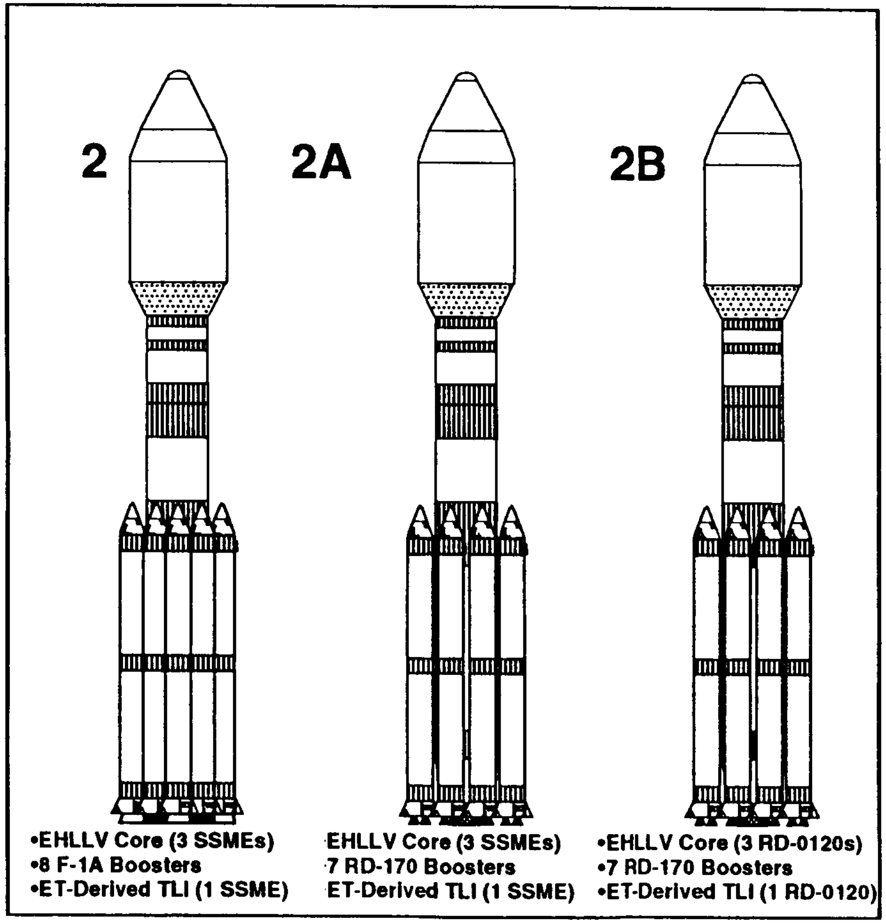
•EHLLV Core (3 SSMEs)
•8 ASRMs
•ET-Derived TLI (1 SSME)

Pros:
- No new engines
- Booster simplicity
- Low DDT&E

Cons:
- Cannot lift 93 MT
- Booster stacking complexity
- Environmental issues
- Acceleration loads vs. core vehicle beef-up
- Separation dynamics
- Vehicle & ground load path complexity
- Mars evolution questionable
- Hammerhead vs. inert wt. hit

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options

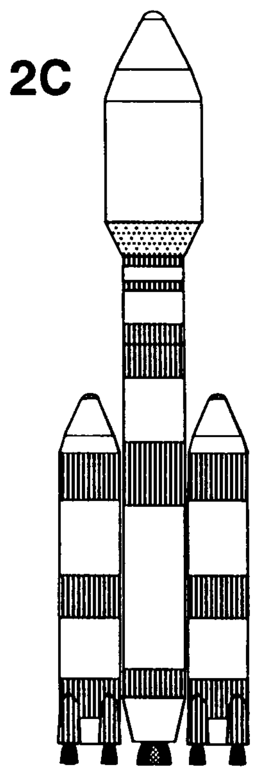


Pros:
- Core commonality
- Booster design simplicity
- Booster test & check-out simplicity
- Booster unit cost savings (larger lot buy)
- ELV family evolution from booster

Cons:
- New RP boosters engine
- Hammerhead vs. inert wt. hit
- Booster stacking
- Programmatic risk of CIS engines
- QA/QC uncertainties of CIS engines
- Mars evolution questionable
- Vehicle & ground load path complexity
- High FMEA/CIL count
- FRF feasibility questionable

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options



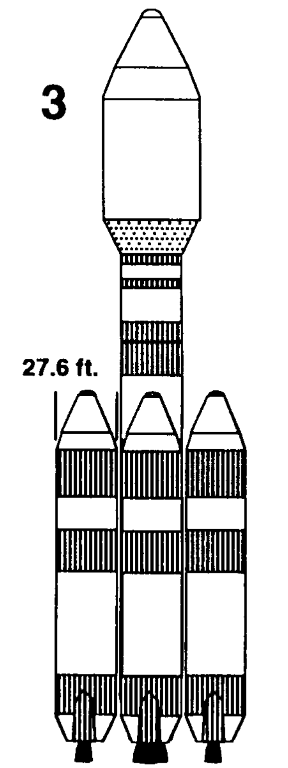
•EHLLV Core (3 SSMEs)
•2 ET-Derived Boosters (4 F-1As ea.)
•ET-Derived TLI (1 SSME)

Pros:
- Booster commonality with core
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Simple vehicle & ground load paths
- Fewer vehicle FMEAs/CILs

Cons:
- Booster MPS feed complexity
- Potentially higher booster unit cost (smaller lot buy)
- Hammerhead vs. core inert wt. hit

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options



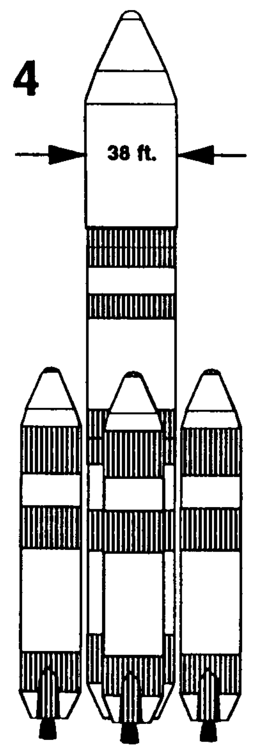
•EHLLV Core (3 SSMEs)
•4 ET-Derived Boosters (2 M-1s ea.)
•ET-Derived TLI (1 SSME)

Pros:
- Booster commonality with core
- Common booster/core propellants
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Fewer vehicle FMEAs/CILs
- No environmental impacts
- Potentially lower booster unit cost (larger lot buy)

Cons:
- New booster engine
- Booster leak potential
- Lower booster density impulse
- Hammerhead vs. core inert wt. hit
- Vehicle & ground load path complexity

J. B. McCurry
205-722-4509

Lunar Mission
Parallel-Burn Configuration Options



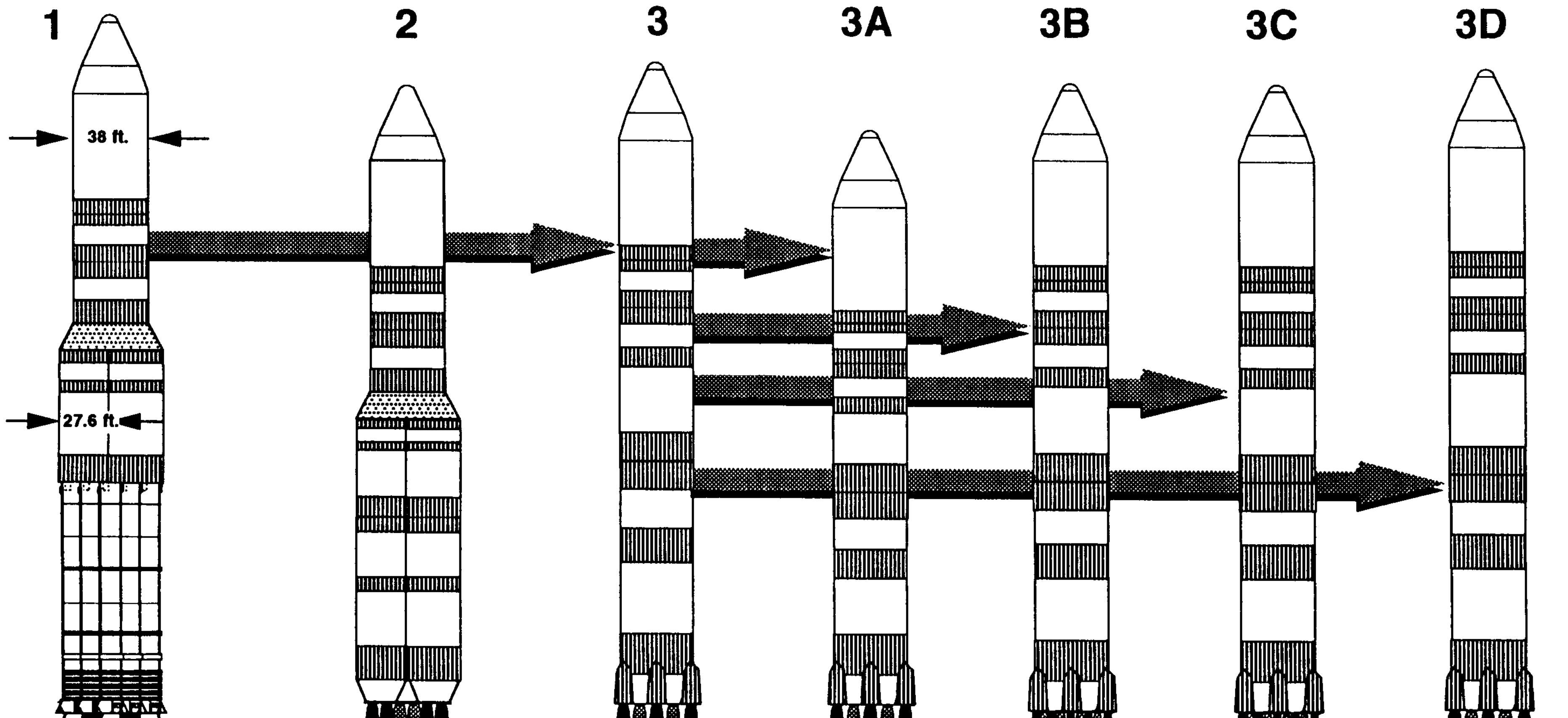
•38 ft. dia. Core (2 M-1s)
•4 ET-Derived Boosters (2 M-1s ea.)
•38 ft. dia. TLI (1 SSME)

Pros:
- No hammerhead
- Common booster/core propellants
- Booster stand-alone ELV
- Mars evolution potential
- Fewer element interfaces
- Fewer vehicle FMEAs/CILs
- No environmental impacts
- Potentially lower booster unit cost (larger lot buy)

Cons:
- New core vs. booster commonality with ET
- New booster/core engine
- Booster leak potential
- Lower booster density impulse
- Vehicle & ground load path complexity

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options



1
- TBD ASRMs First Stage
- 3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
- 38 ft. Dia. 3rd Stage (1 SSME)

2
- 3 ET-Derived Tanks First Stage (3 F-1As ea.)
- 3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
- 38 ft. Dia. 3rd Stage (1 SSME)

3
- 38 ft. Dia. 1st Stage (7 F-1As)
- 38 ft. Dia. 2nd Stage (5 SSMEs)
- 38 ft. Dia. TLI Stage (1 SSME)

3A
- 38 ft. Dia. 1st Stage Multi-Cell (7 F-1As)
- 38 ft. Dia. 2nd Stage Multi-Cell (5 SSMEs)
- 38 ft. Dia. TLI Stage Multi-Cell (1 SSME)

3B
- 38 ft. Dia. 1st Stage (7 RD-170s)
- 38 ft. Dia. 2nd Stage (5 SSMEs)
- 38 ft. Dia. TLI Stage (1 SSME)

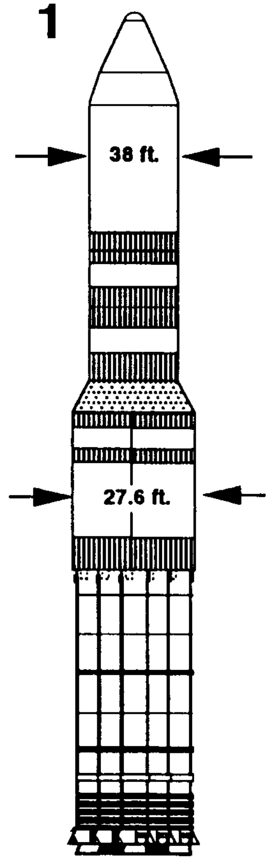
3C
- 38 ft. Dia. 1st Stage (7 RD-170s)
- 38 ft. Dia. 2nd Stage (5 RD-0120s)
- 38 ft. Dia. TLI Stage (1 RD-0120)

3D
- 38 ft. Dia. 1st Stage (TBD M-1s)
- 38 ft. Dia. 2nd Stage (TBD M-1s)
- 38 ft. Dia. TLI Stage (1 SSME)

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

1



•TBD ASRMs First Stage
•3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
•38 ft. Dia. 3rd Stage (1 SSME)

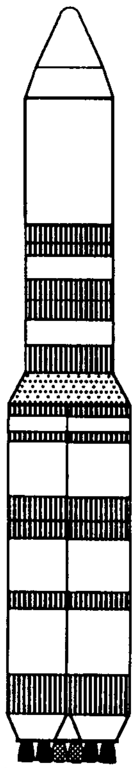
Pros:
- No hammerhead
- No new engines
- Booster simplicity
- Existing stage elements vs. attach hardware development

Cons:
- New TLI tank design
- Stage stacking complexity
- Interstage complexity
- TLI stage tank complexity (probably multiple tanks)
- Environmental issues
- Acceleration loads vs. core vehicle beef-up
- Vehicle & ground load path complexity
- Many SSMEs
- Mars evolution questionable

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

2



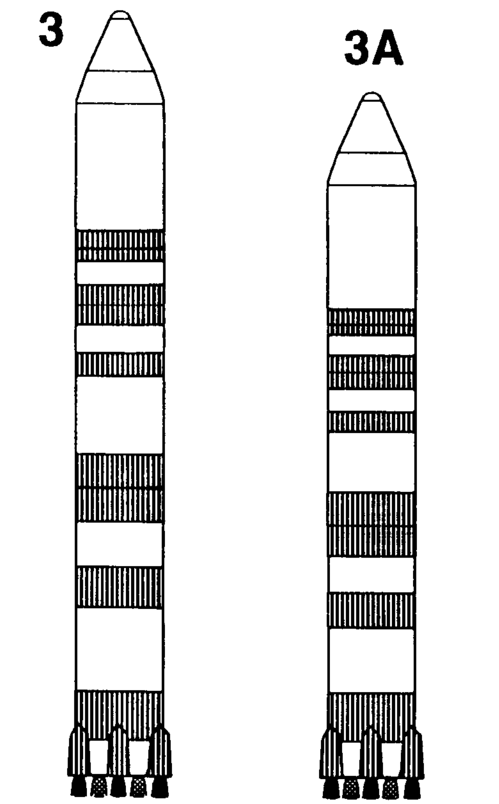
- 3 ET-Derived Tanks First Stage (3 F-1As ea.)
- 3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
- 38 ft. Dia. 3rd Stage (1 SSME)

<u>Pros</u>:
- No hammerhead
- First/Second stage tank commonality with ET
- Capable of testing each stage element independently
- Existing stage elements vs. attach hardware development
- Lessened environmental issues
- Mars evolution potential

<u>Cons</u>:
- New TLI tank design
- Stage stacking complexity
- Interstage complexity
- TLI stage tank complexity (probably multiple tanks)
- Vehicle load path complexity
- Many SSMEs

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

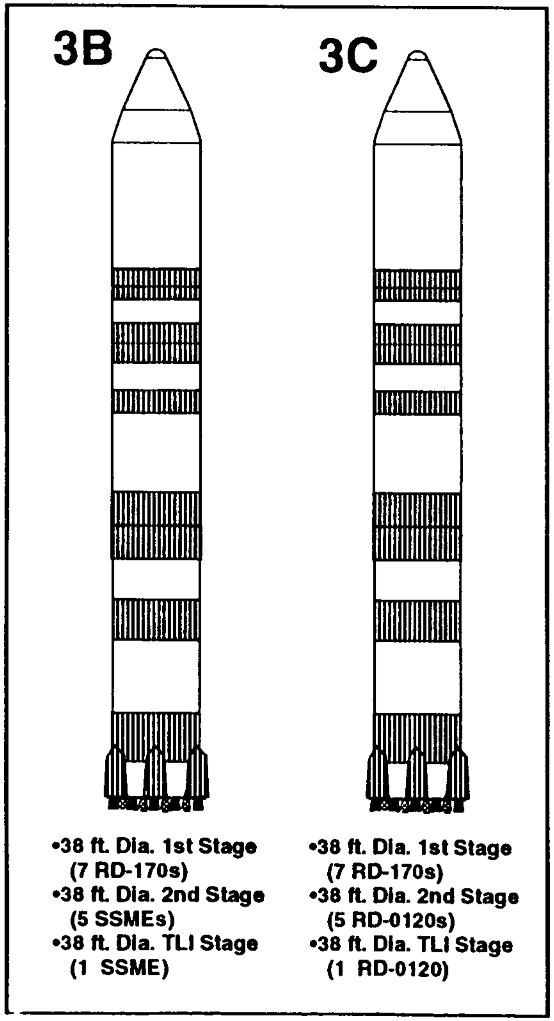


Pros:
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- Lessened environmental issues
- Mars evolution potential
- Less weld length for multi-cell
- Fewer stage pressurization subsystems
- No first stage leak issue

Cons:
- New stage design
- Many SSMEs
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)
- Many SSMEs
- Complicated first/second stage MPS feed subsystems (unless use multi-cell)

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options



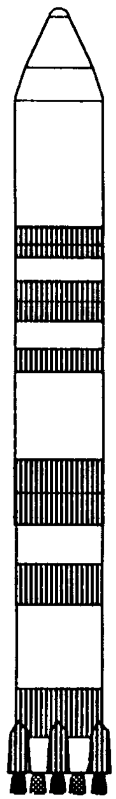
Pros:
- Shorter first stage design (better Isp)
- Potentially cheaper engines & no development cost
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- Lessened environmental issues
- Mars evolution potential
- Fewer stage pressurization subsystems
- No first stage leak issue

Cons:
- Foreign engines & RP source
- New stage design
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)
- Complicated first/second stage MPS feed subsystems

J. B. McCurry
205-722-4509

Lunar Mission
Series-Burn Configuration Options

3D



•38 ft. Dia. 1st Stage (TBD M-1s)
•38 ft. Dia. 2nd Stage (TBD M-1s)
•38 ft. Dia. TLI Stage (1 SSME)

Pros:
- Fewer engines for first/second stage (higher thrust)
- Simpler MPS feed subsystems
- Propellant commonality for all stages
- No hammerhead
- Common stage diameter & tank design
- Fewer stage sub-elements & FMEAs/CILs
- Fewer engines per stage with simpler MPS
- Capable of testing each stage element independently
- Simplified load path
- No environmental issues
- Mars evolution potential
- Fewer stage pressurization subsystems

Cons:
- Poor first stage density impulse (larger stage)
- Greater leak potential
- New stage design
- Long total vehicle length (unless use multi-cell)
- TLI stage tank complexity (probably multiple tanks)

J. B. McCurry
205-722-4509

4. Propulsion Options

Introduction

- Six engines were identified for HLLV studies

    260 in. diameter motor
    ASRM
    RD-170
    RD-0120
    STME
    M1
    M1A

- Performance characteristics of selected engines were compiled
- Emerging technologies are being identified for possible integration
- Comparison of selected engine combinations will be made
- Trades will be conducted to determine optimum configuration

J. B. McCurry
205-722-4509

Summary Of Large LOX/Hydrocarbon And LOX/LH2 Rocket Engines* (200K and Larger)

Engine	Country	Fuel	Oxidizer	Thrust, Klbf**		Specific Impulse, Sec.**	
				Sea Level	Vacuum	Sea Level	Vacuum
F-1	US	RP-1	LOX	1522	1746	265	304
MA-5A***	US	RP-1	LOX	423.5/50.5	473.4/85.0	264/220	295/309
RS-27A	US	RP-1	LOX	200	237	255	302
H-1	US	RP-1	LOX	205	230	264	296
LR-87	US	RP-1	LOX	300	344.4	252	289
XLR-109	US	RP-1	LOX	500	---	265	---
J-2	US	LH2	LOX	161.4	230	293.7	422.7
J-2S	US	LH2	LOX	201	265	330	435
M-1	US	LH2	LOX	---	1500	---	428
M-1A	US	LH2	LOX	1300	---	344.5	---
SSME	US	LH2	LOX	373.5	468.4	362	454

NOTES:
* Data Source: Ed Bair,Aerojet
** 100 percent rated power level
***MA-5A Data: Booster(2 Thrust Chambers)/Sustainer (1 Thrust Chamber)

J. B. McCurry
205-722-4509

Summary Of Large LOX/Hydrocarbon And LOX/LH2 Rocket Engines* (200K and Larger)

				Thrust, Klbf**		Specific Impulse, Sec.**	
Engine	Country	Fuel	Oxidizer	Sea Level	Vacuum	Sea Level	Vacuum
RD-107	CIS	Kerosene	LOX	184.6	224.8	257	314
RD-108	CIS	Kerosene	LOX	167.5	211.5	248	315
RD-170	CIS	Kerosene	LOX	1631	1777	309	337
RD-120	CIS	Kerosene	LOX	---	181.5	---	350
NK-33	CIS	Kerosene	LOX	339	378	297	331
NK-43	CIS	Kerosene	LOX	---	395	---	346
RD-0120	CIS	LH2	LOX	326	441	354	452.5

NOTES:
* Data Source: Ed Bair, Aerojet
** 100 percent rated power level

J. B. McCurry
205-722-4509

M1 Altitude Engine Summary

Thrust at 200K ft, lbf	1,500,000
I_{sp}, sec	428
MR (O/F)	5.0:1
Exit Area Ratio	40:1
Chamber Pressure (psia)	1000
Engine Wt	
Dry	20,000
Wet	22,000
Burnout	21,086
Propellants	
Fuel	H_2
OX	O_2
Cycle	GG
Status	Dev

J. B. McCurry
205-722-4509

M1 Engine Development Status

Components Built

Gas Generator	11
Injector	4
LO_2 Pump	1
LH_2 Pump	1
Thrust Chamber (Uncooled)	2

- One set of each component was successfully tested
- Some hardware was sent to the Smithsonian; the remainder was scrapped
- Prints, documentation and specifications are available

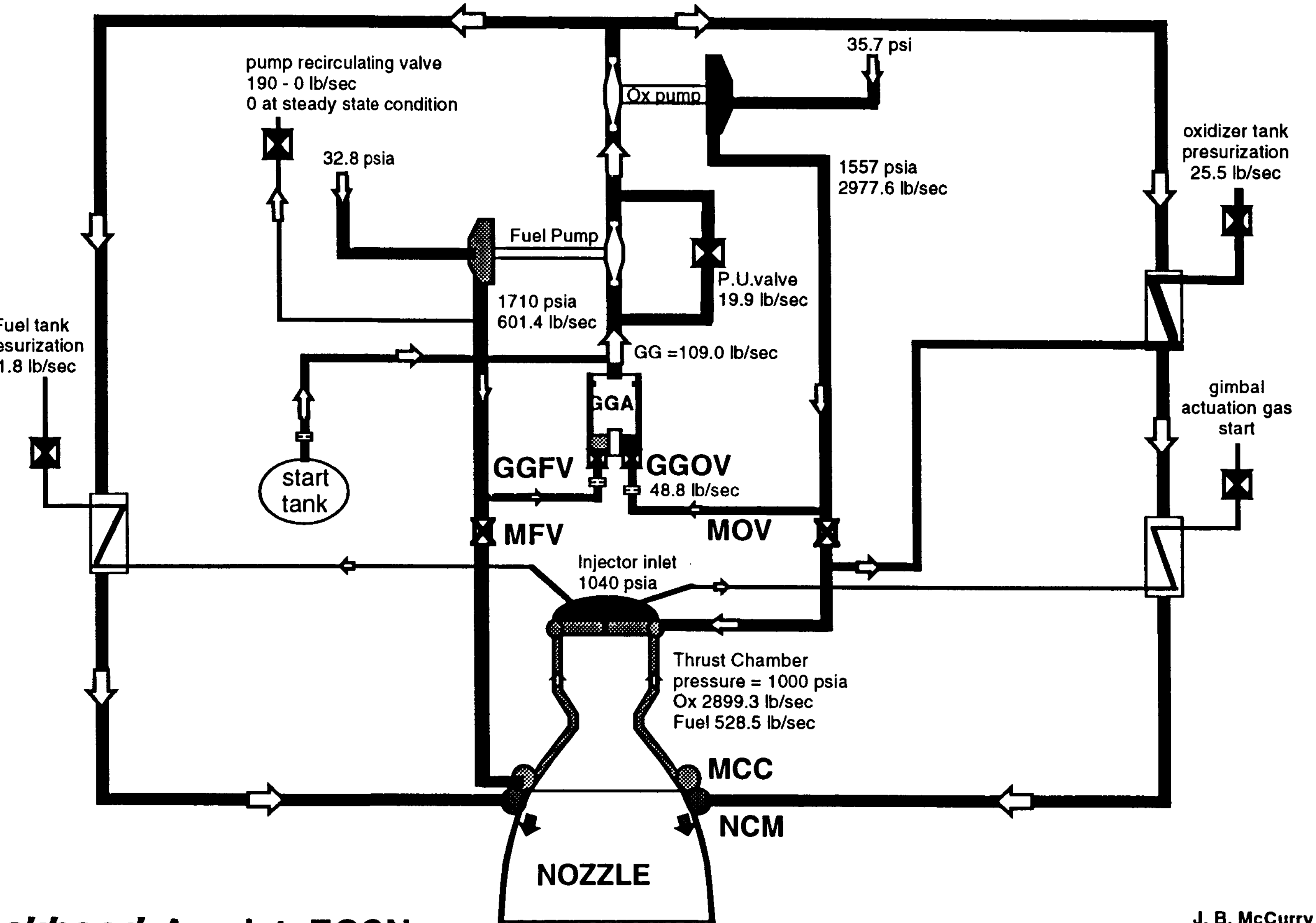
J. B. McCurry
205-722-4509

M1A Sea Level Engine Summary

Sea Level Thrust, lbf	1,300,000
Sea Level I_{sp}, sec	344.5
MR (O/F)	5.0:1
Exit Area Ratio	20:1
Chamber Pressure, psia	1000
Engine Weight, lbf	
Dry	19,100
Wet	21,100
Burnout	20,186
Propellants	
Fuel	H_2
OX	O_2
Cycle	GG
Status	Design

J. B. McCurry
205-722-4509

M-1 Power Balance Diagram



Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

M-1 Performance Characteristics

Nominal Operation (feet)*

Propellants:	
Liquid Oxygen-Density	70.67 lb/cu ft
Liquid Hydrogen-Density	4.334 lb/cu ft
Engine:	
Ambient Pressure	.0002 psia
Thrust	1,500,000 lb
Specific Impulse	424 sec
Mixture Ratio	5.00
Thrust Chamber Expansion Ratio	40.0:1
Thrust Chamber Throat Area	803.35 sq in.
Thrust Chamber:	
Thrust	1,471,000 lb
Specific Impulse	429.3 sec
Mixture Ratio	5.49
Flow Rate:	
Oxidizer	2899.3 lb/sec
Fuel	528.5 lb/sec
Fuel Pump:	
Head Rise	52,684 ft
Capacity	62,281 GPM
Efficiency	.777
Horsepower (Shaft)	74,138 BHP
Speed	12,961 RPM

Oxidizer Pump:	
Head Rise	3102 ft
Capacity	18,927 GPM
Efficiency	.681
Horsepower (Shaft)	24,655 BHP
Speed	3530 RPM
Fuel Turbine:	
Inlet Total Pressure	904 psia
Pressure Ratio	3.87
Inlet Total Temperature	1000 deg F
Efficiency	.65
Horsepower	74,138 BHP
Oxidizer Turbine:	
Inlet Total Pressure	194 psia
Pressure Ratio	1.62
Efficiency	.530
Horsepower	24,655 BHP
Inlet total Temperature	763 deg F
Exit Total Temperature	684 deg F
Start Tank:	
Initial Pressure	3000 psia
Volume	15 cu ft
Initial Temperature	250 deg R

J. B. McCurry
205-722-4509

Recommended Baseline for Minimum Number of DDT&E Engine Tests

Number of Engines

Development	6
Certification	4
Flight Acceptance	4
Total	14

Tests

Development	190
Certification	90
Flight	4
Total	284

J. B. McCurry
205-722-4509

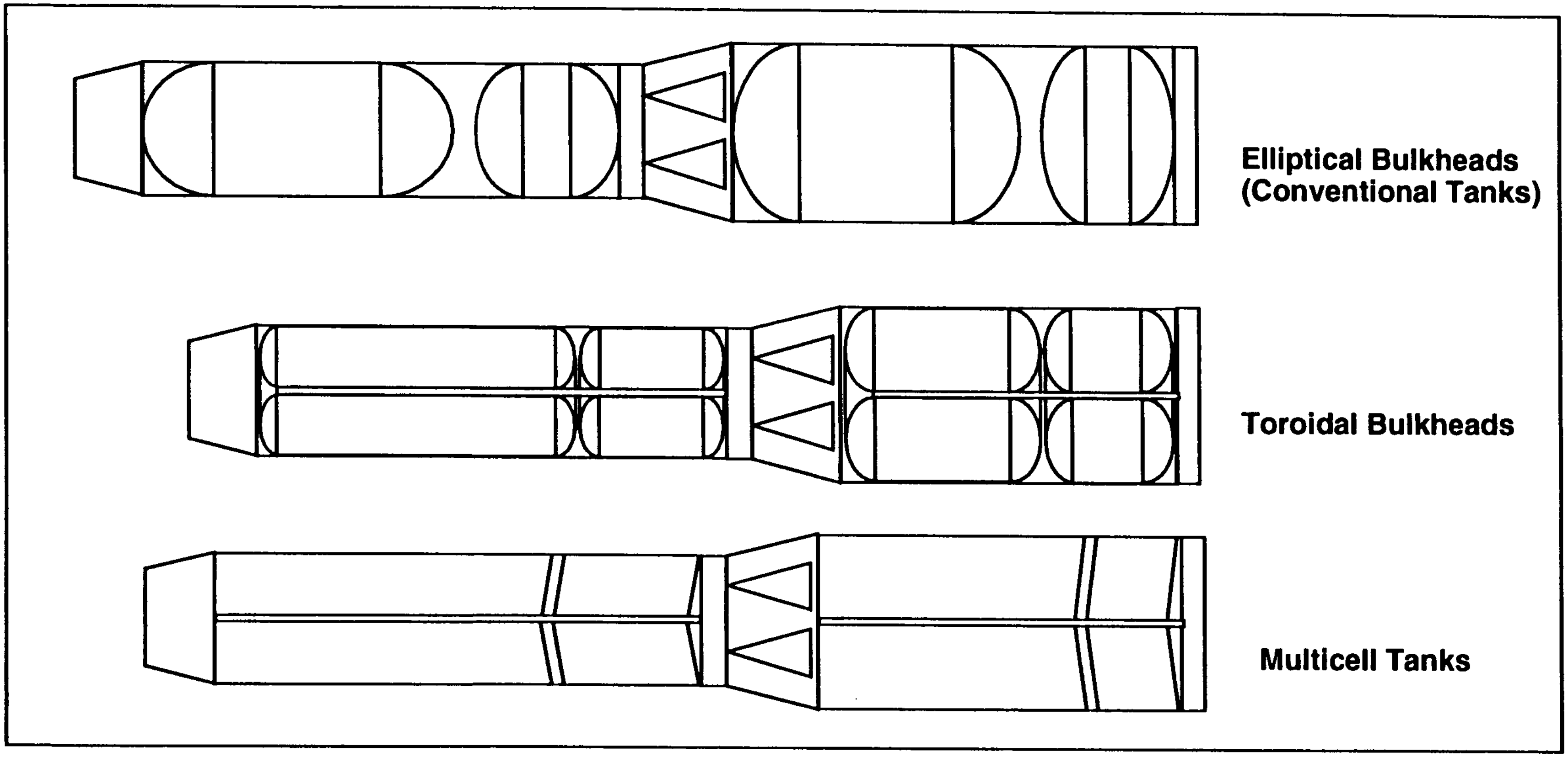
Emerging Technologies Are Being Identified for Possible Integration

- Unconventional nozzles allow excellant packaging for high thrust/Isp engines
  - Plug/nozzles technology is being developed under SSRT Program
  - Force deflection nozzle technology is being developed under IR&D and Mist Program
- Low Cost/Low Hazard (A3L) solid propellants are being developed under IR&D Programs
  - Insensitive to chance ignition
  - Simplified processing
  - Excellent performance
  - Environmentally advantageous (no HCl, no production waste, water clean-up)

J. B. McCurry
205-722-4509

5. Structures Options

Alternative Propellant Tank Designs



Vehicle Comparison

Tank Type	Structural Weight (lbm)	Length (ft)
Conventional	322,400	306
Toroidal	317,700	256
Multi-Cell	247,200	273

Keith Holden
(205) 722-4531

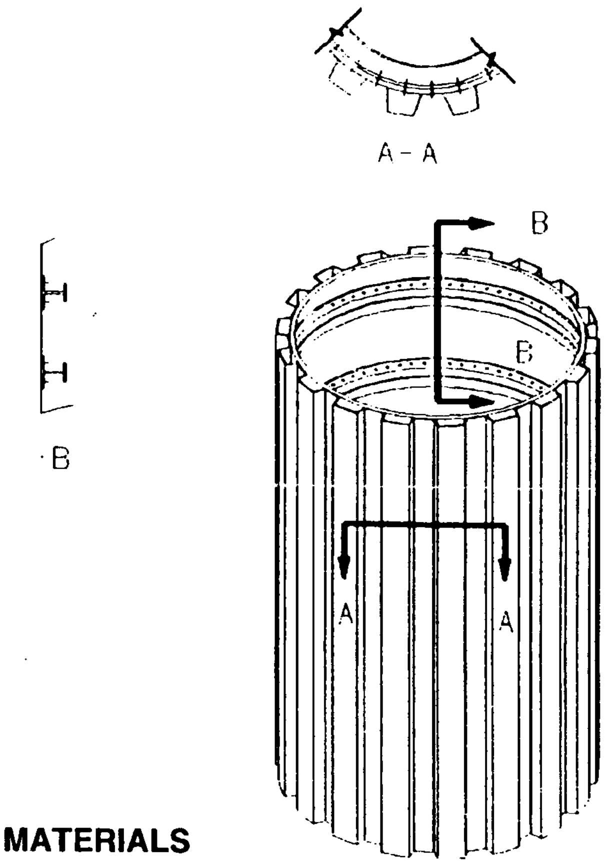
Weight (lbm) Comparison

Structure	Conventional	Semitoroidal	Multicell
Forward Skirt	29,000	19,000	15,000
Intertank Skirt	52,000	33,000	36,000
Fuel Tank	32,000	45,000	37,600
Lox Tank	67,400	75,700	63,600
Aft Skirt and Thrust Structure	142,000	145,000	95,000
Total	322,400	317,700	247,200

Keith Holden
(205) 722-4531

Structural Concept
Ring and Stringer Stiffened

- APPLICABLE STRUCTURES
  - INTERSTAGES
  - AFT & FWD SKIRTS
  - THRUST STRUCTURES
- CONSTRUCTION MATERIALS
  - AL7075 (BOLTED)
  - AL2219 (WELDED)
  - AL-LI (10-25% WEIGHT SAVING)
- FABRICATION METHODS
  - SKIN /STRINGER ATTACHMENT
    - AUTOMATIC RIVETING
    - WELD-BONDING
  - SKIN/RING ATTACHMENT
    - MECHANICAL FASTENERS



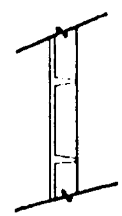
ADVANTAGES- EFFICIENT MATERIAL USAGE, MIXED USE OF MATERIALS POSSIBLE

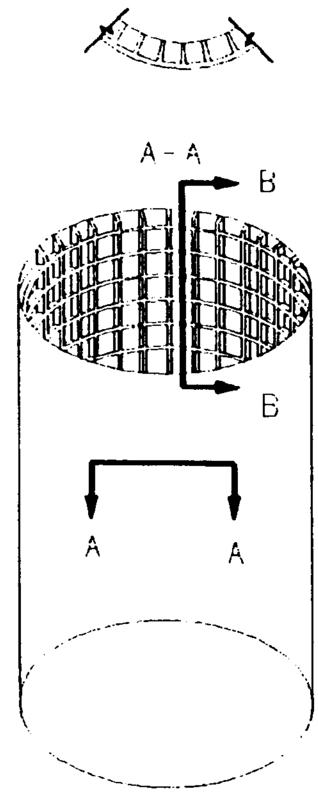
DISADVANTAGES - LABOR INTENSIVE, HIGHER WEIGHT ASSOCIATED WITH MECHANICALLY FASTENED STRUCTURAL SYSTEM, CORRUGATED EXTERNAL SURFACE COMPLICATE INSULATION PROCESS.

J. B. McCurry
205-722-4509

Structural Concept Integral Stiffened

- APPLICABLE STRUCTURES
  - INTERSTAGES
  - AFT & FWD SKIRTS
  - THRUST STRUCTURES
- CONSTRUCTION MATERIALS
  - AL7075 (BOLTED)
  - AL2219 (WELDED)
  - AL-LI (10-25% WEIGHT SAVING)
- FABRICATION METHODS
  - SKIN PANELS
    - MACHINE GRIDS IN FLAT CONDITION, & BRAKE FORM OR AGE FORM IN DESIRED CONTOUR
  - ASSEMBLE METHODS
    - BOLTED AND/OR WELDED
- ADVANTAGES-
  - INTEGRAL RING/STRINGER SYSTEM (NO FASTENERS),
  - REDUCE LABOR COST,
  - REDUCED STRUCTURE WEIGHT (LESS FASTENERS)
- DISADVANTAGES -
  - INEFFICIENT MATERIAL USAGE (MACHINE HOGGED-OUT GRID SYSTEM
  - ISOGRID TAKES ~ 30% MORE MACHINING TIME THAN WAFFLE GRID
- IF AXIAL LOADS ARE DOMINANT, PREFER SKIN-STRINGER DESIGN; IF BIAXIAL LOADS ARE DOMINANT, PREFER ISOGRID



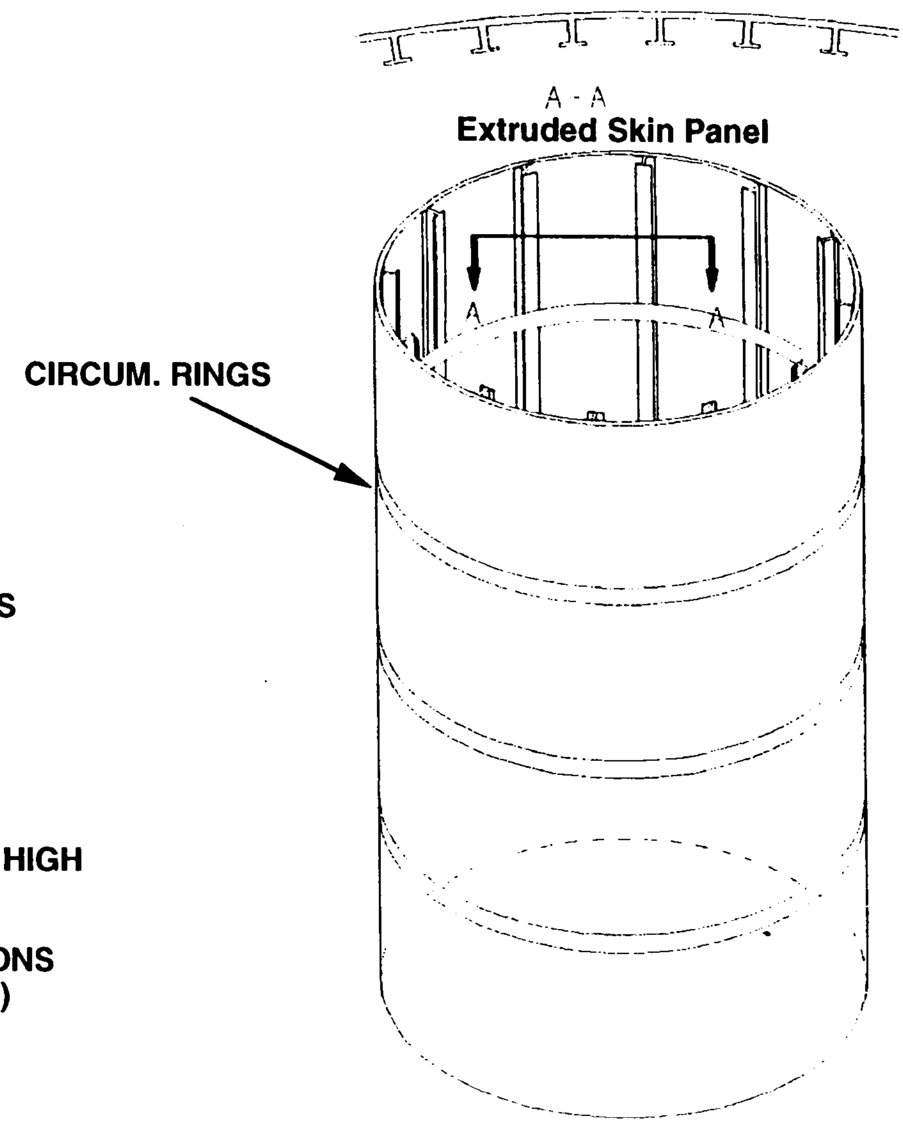


J. B. McCurry
205-722-4509

Structural Concept

EXTRUDED SKIN PANEL W/ INTEGRAL STIFFENERS

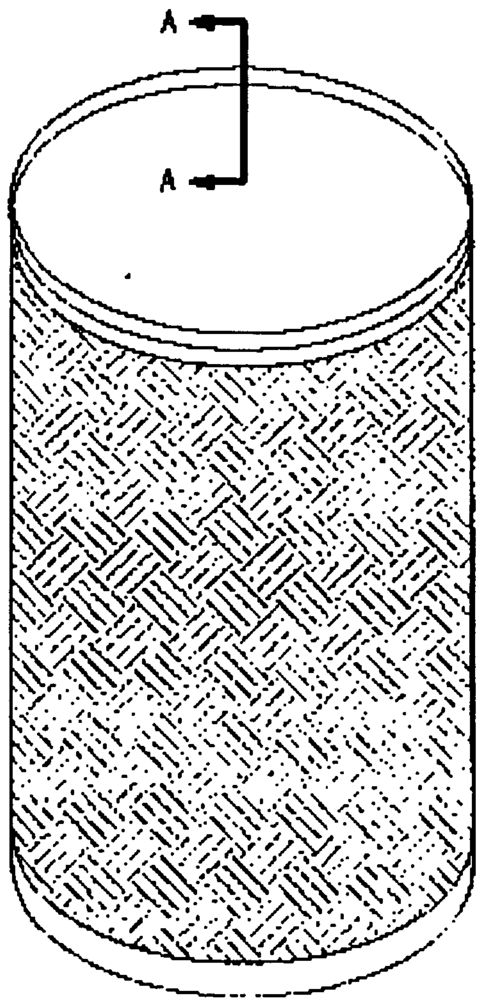
- APPLICABLE STRUCTURES
  - INTERSTAGES
  - AFT & FWD SKIRTS
- CONSTRUCTION MATERIALS
  - AL7075 (BOLTED)
  - AL2219 (WELDED)
  - AL-LI (10-25% WEIGHT SAVING)
- FABRICATION METHODS
  - SKIN PANELS
    - EXTRUSION
  - ASSEMBLE METHODS
    - BOLTED AND/OR WELDED
- ADVANTAGES-
  - INTEGRAL STRINGERS REDUCE FASTENERS
  - REDUCE MACHINING COST
  - EFFECTIVE MATERIAL USAGE
- DISADVANTAGES -
  - EXTRUDING LARGE PANELS REQUIRES DEVELOPMENT
  - GEOMETRIC STABILITY IS A CONCERN FOR HIGH ASPECT RATIO EXTRUSION
- HEAT-TREATING OF LARGE PANEL EXTRUSIONS IS A TECHNICAL ISSUE (FACILITY LIMITATION)

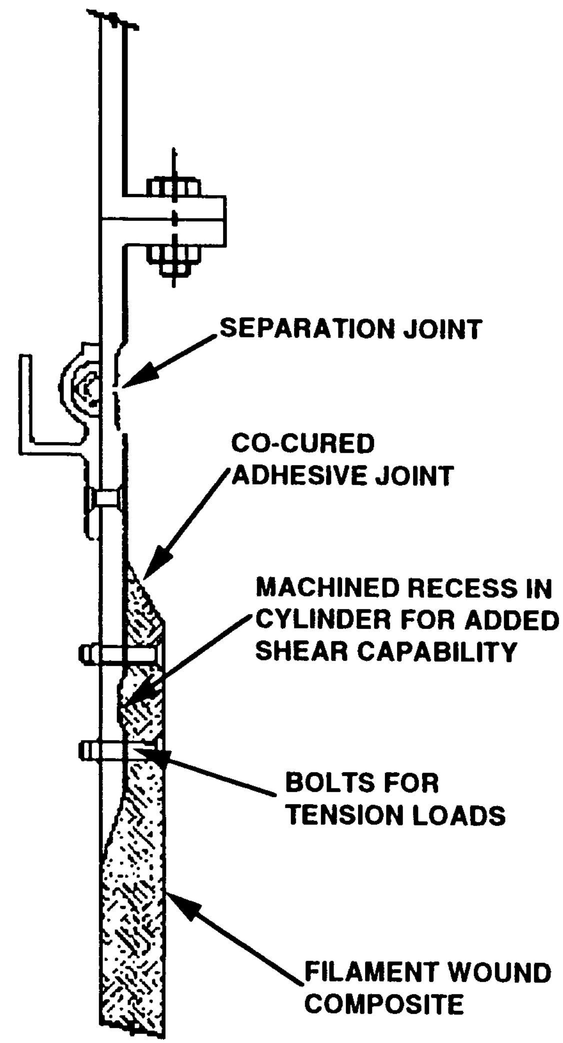


J. B. McCurry
205-722-4509

Structural Concept
Filament Wound Composite

- APPLICABLE STRUCTURES
  - INTERSTAGES
  - AFT & FWD SKIRTS
  - THRUST STRUCTURES
- FABRICATION METHODS
  - FILAMENT WOUND
- ADVANTAGES-
  - LOW RPC
  - REDUCED STRUCTURE WEIGHT
- DISADVANTAGES -
  - HIGHER DDT&E COST

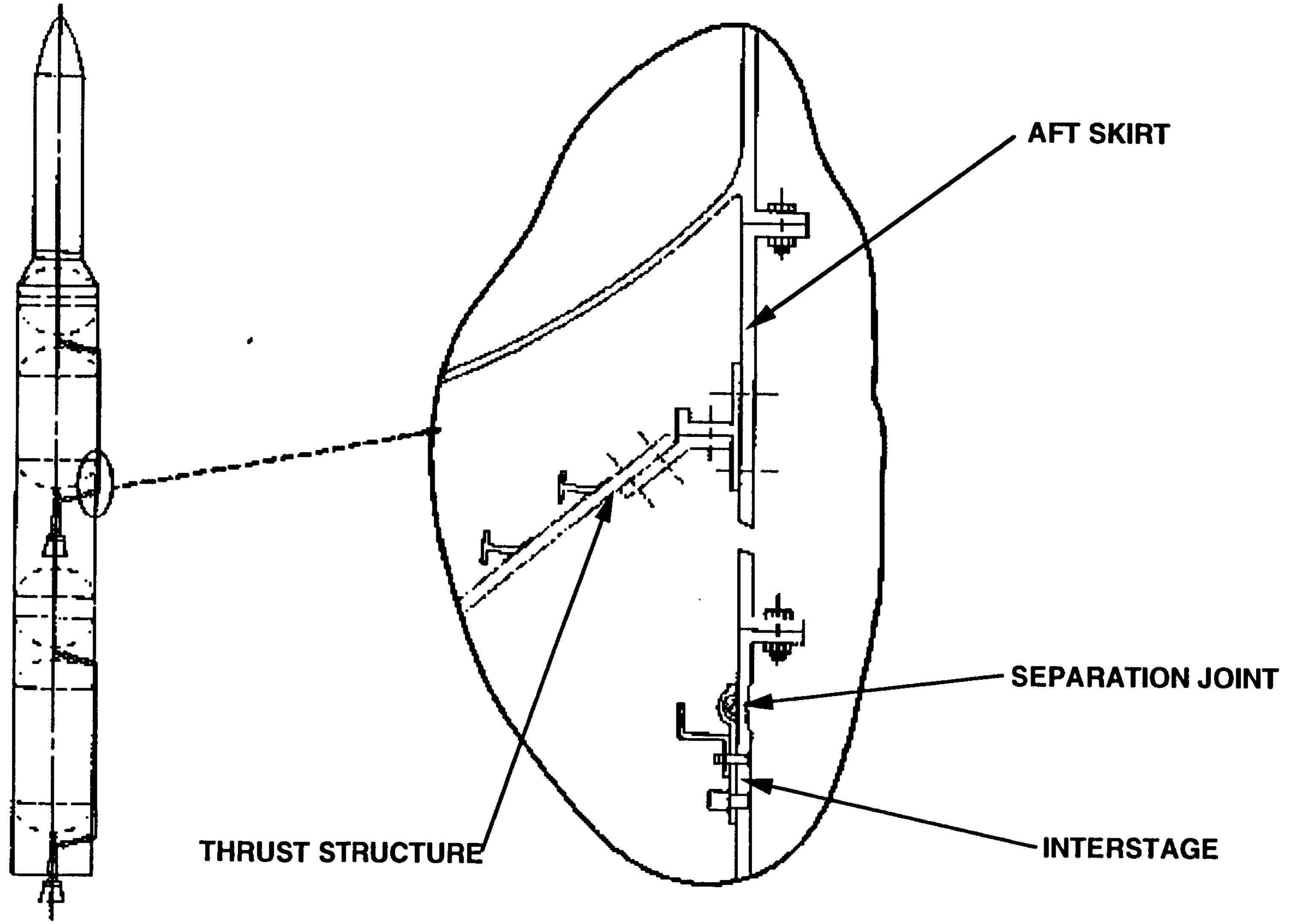




A-A

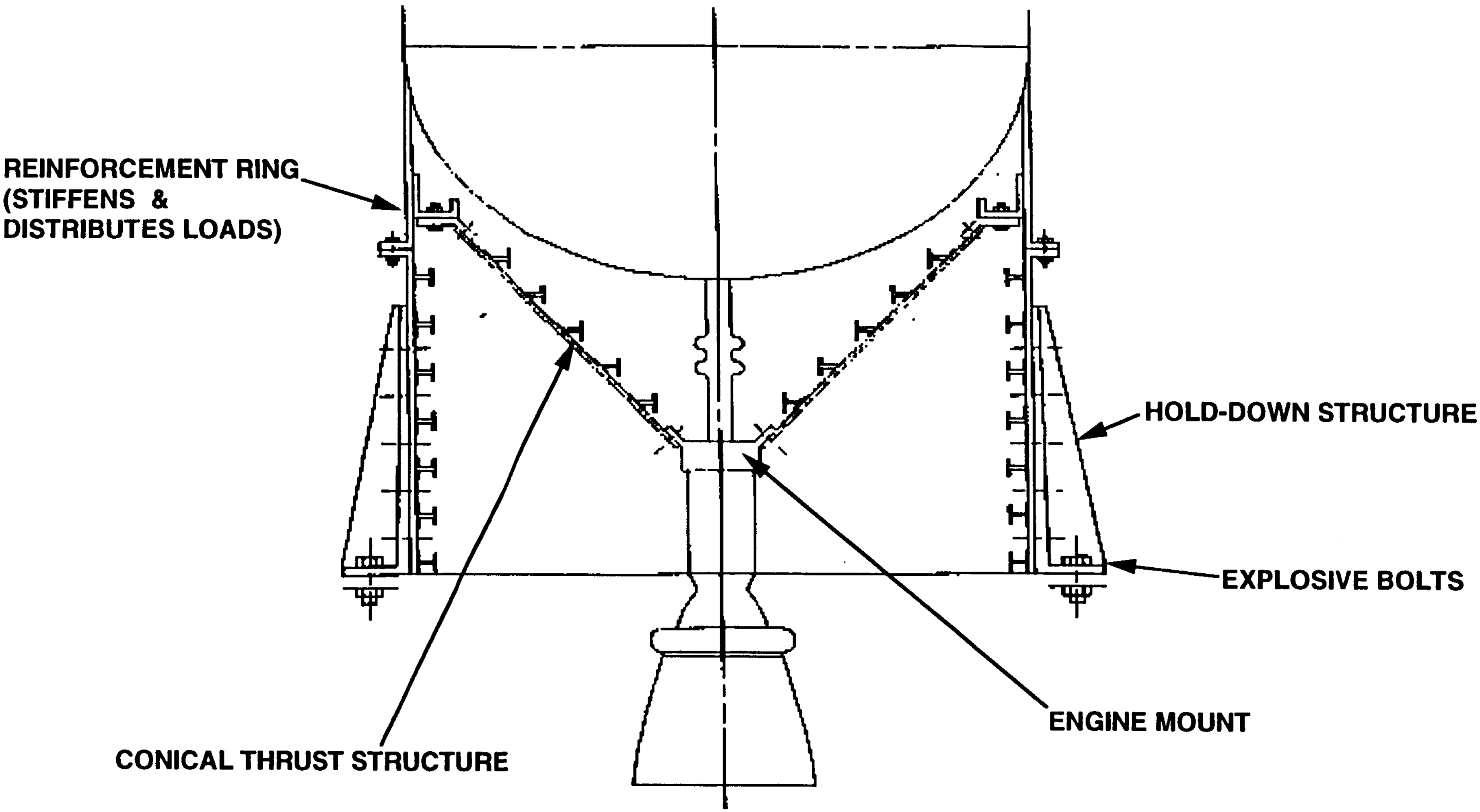
J. B. McCurry
205-722-4509

Separation Joint Concept



J. B. McCurry
205-722-4509

Thrust Structure Concept #1

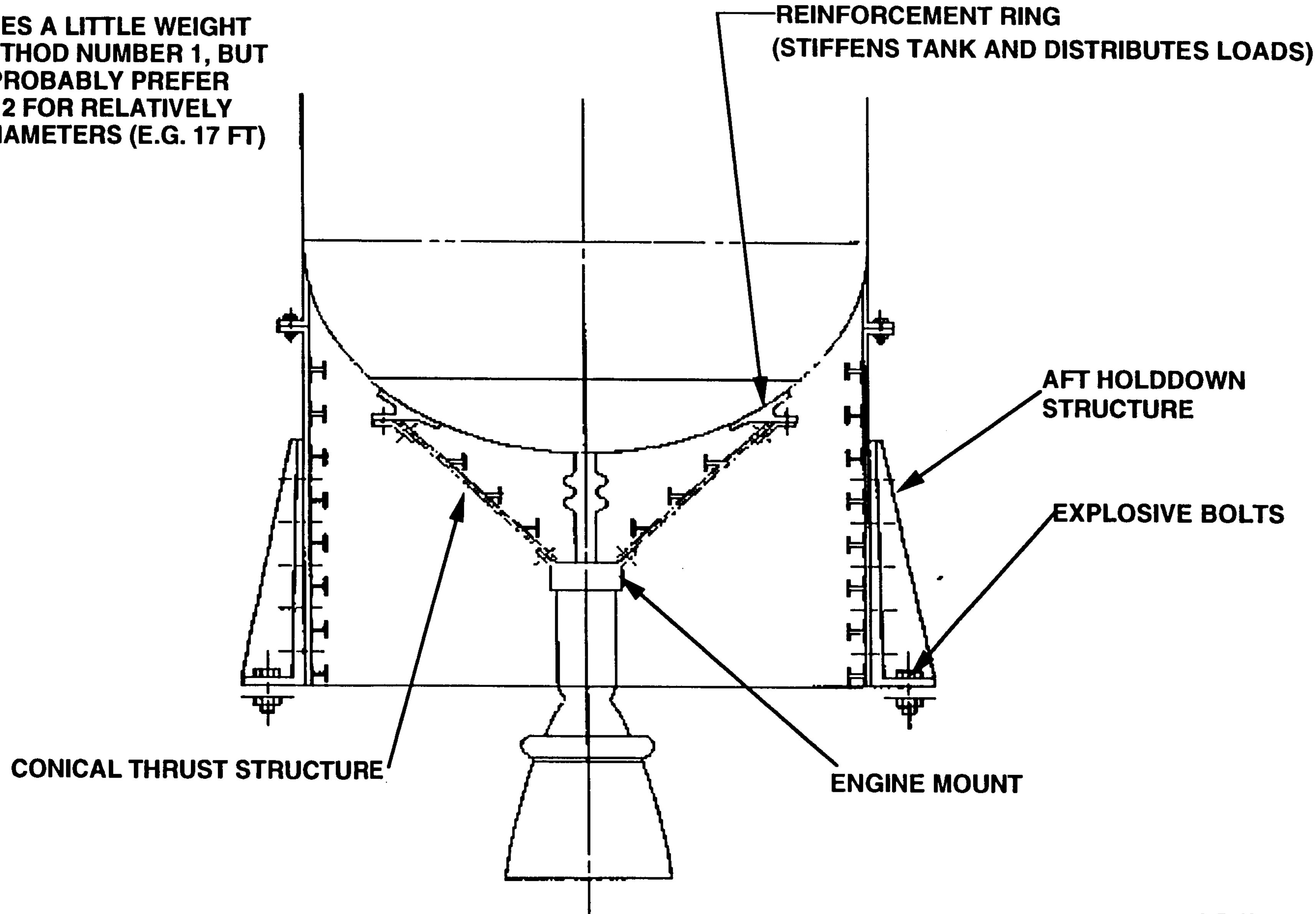


Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

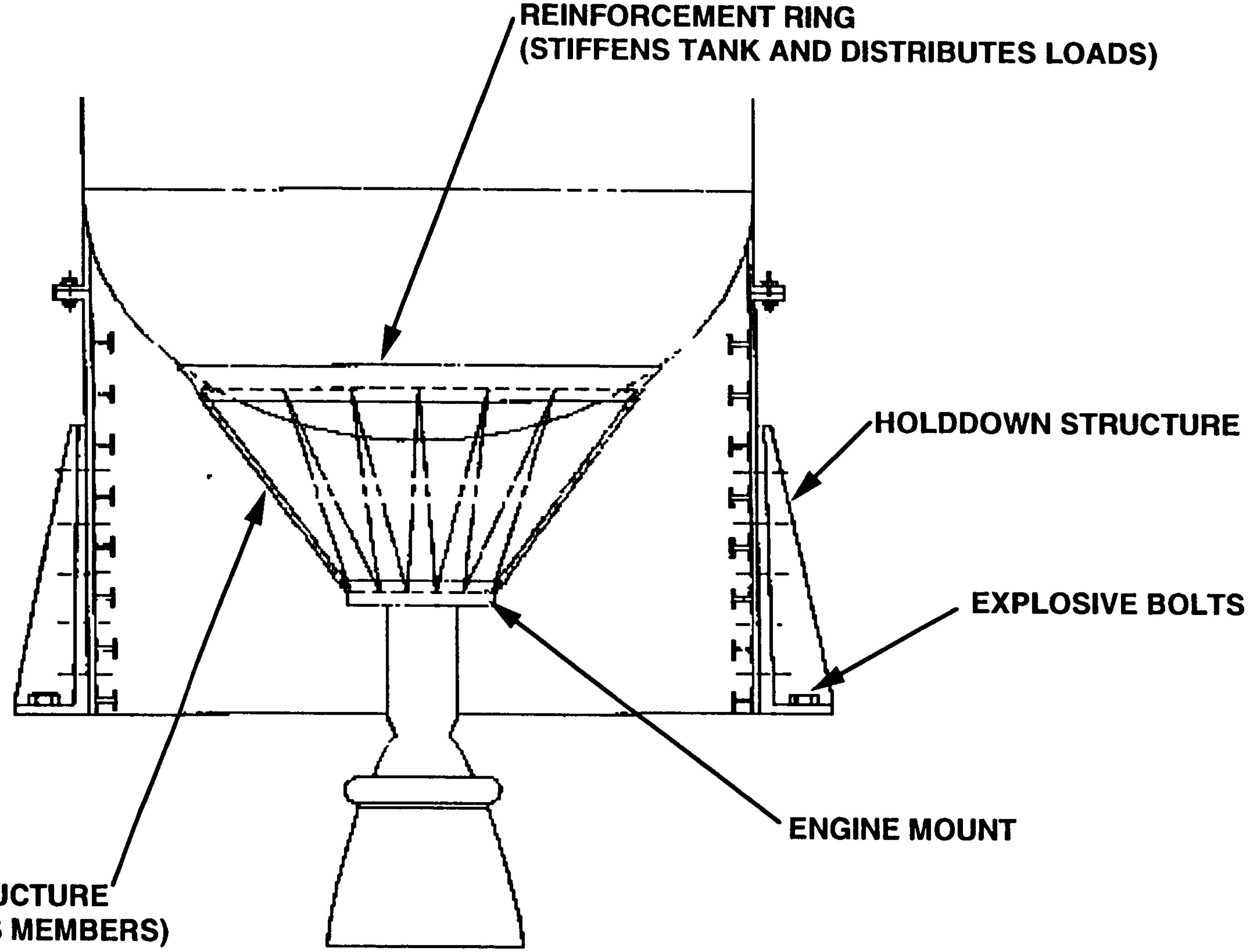
Thrust Structure Concept #2

- THIS SAVES A LITTLE WEIGHT OVER METHOD NUMBER 1, BUT WOULD PROBABLY PREFER NUMBER 2 FOR RELATIVELY SMALL DIAMETERS (E.G. 17 FT)



J. B. McCurry
205-722-4509

Thrust Structure Concept #3



J. B. McCurry
205-722-4509

6. 50/80K Sizing

Alternative Propellant Tank Designs

- Significant design analysis performed by MSFC in '60s on alternative tank designs for large (Saturn V growth) launch vehicles (Blumrich)

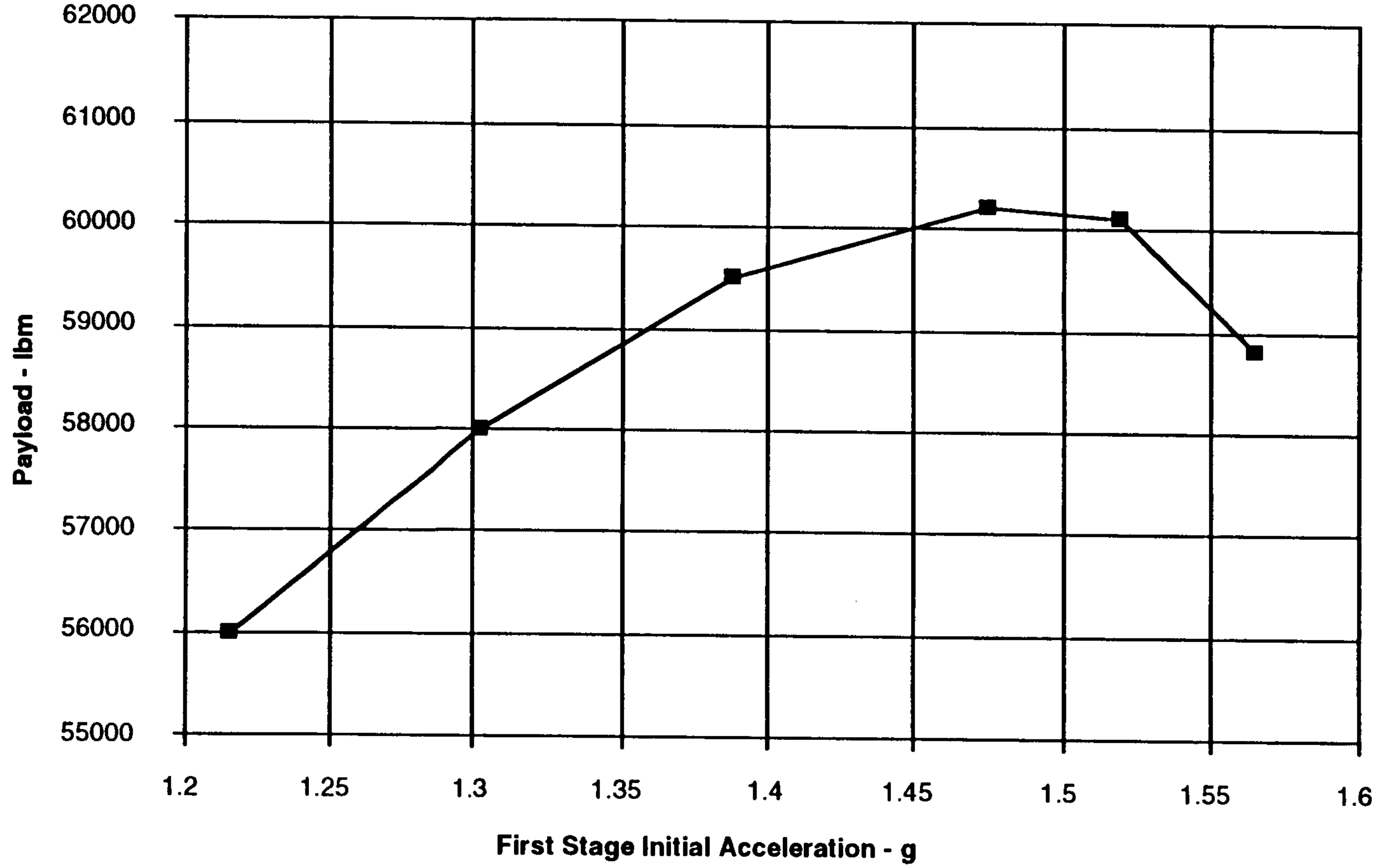
- Semi-Toroidal
  -- Produces large length reduction over conventional non-nested cylinders (up to 50 ft. for large vehicles)
  -- Can provide 2-5 % dry mass reduction
  -- Reduces the "stowed volume" within a given diameter than conventional design, giving higher volumetric efficiency
  -- Requires center post to allow thrust structure to help support the tanks for accelerational loads

- Multi-Cell
  -- Produces good length reduction over conventional designs
  -- Can provide ~10 % dry mass reduction for ET-sized diameters
  -- Can provide ~25 % dry mass reduction for Saturn V-sized diameters
  -- Slosh baffles become part of integral web stiffeners instead of purely parasitic dry mass
  -- If number of cells equals number of engines, can reduce feed line complexity and propellant residuals
  -- Total weld length is less than conventional designs and weld depth can be up to 1/3 less than required for conventional tanks

Keith Holden
(205) 722-4531

Payload as a Function of First Stage Initial Acceleration

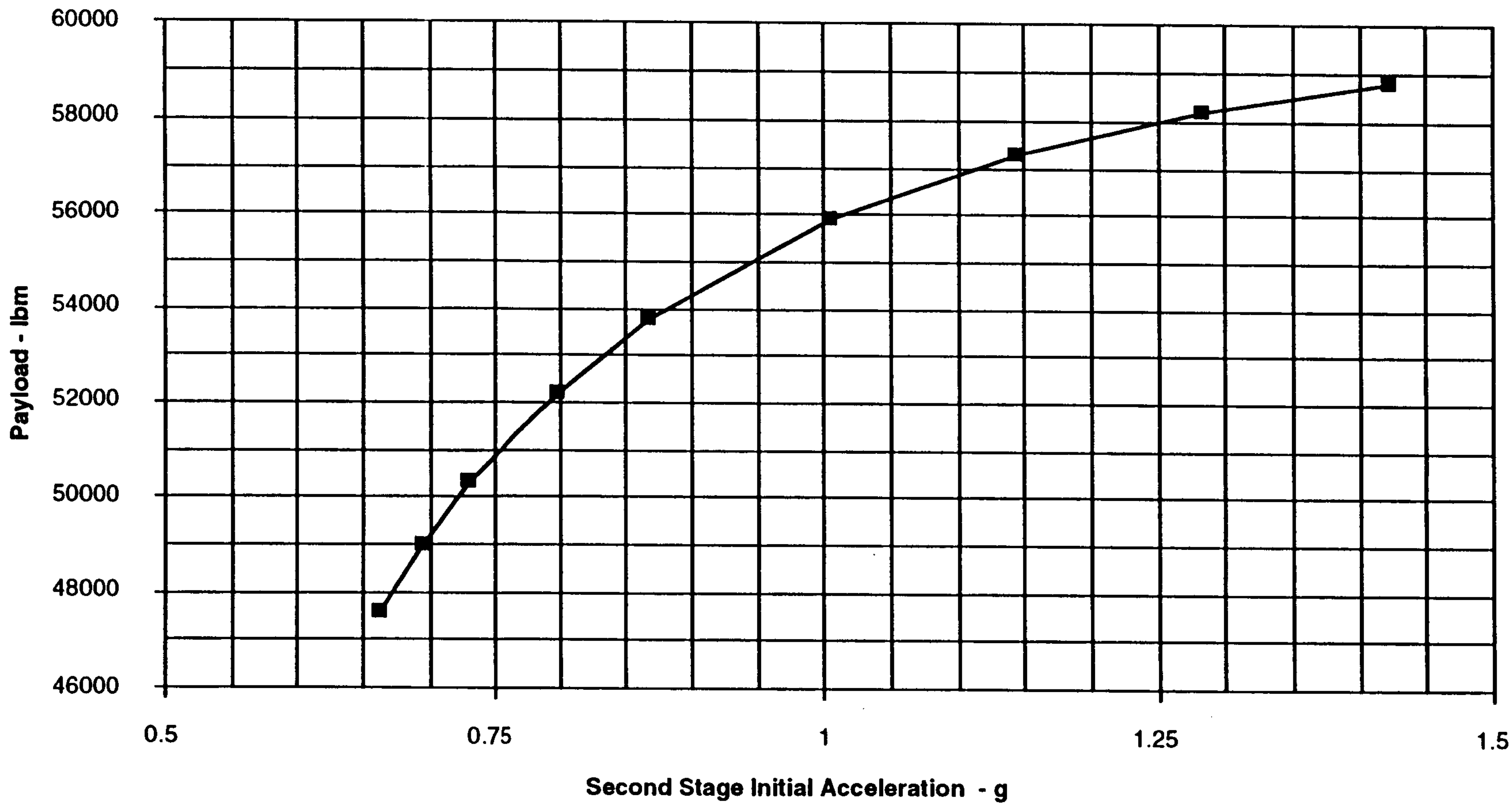
(F-1A/SSME, No1= 1.565 g, No2= 1.423 g)



K. A. Holden
205-722-4531

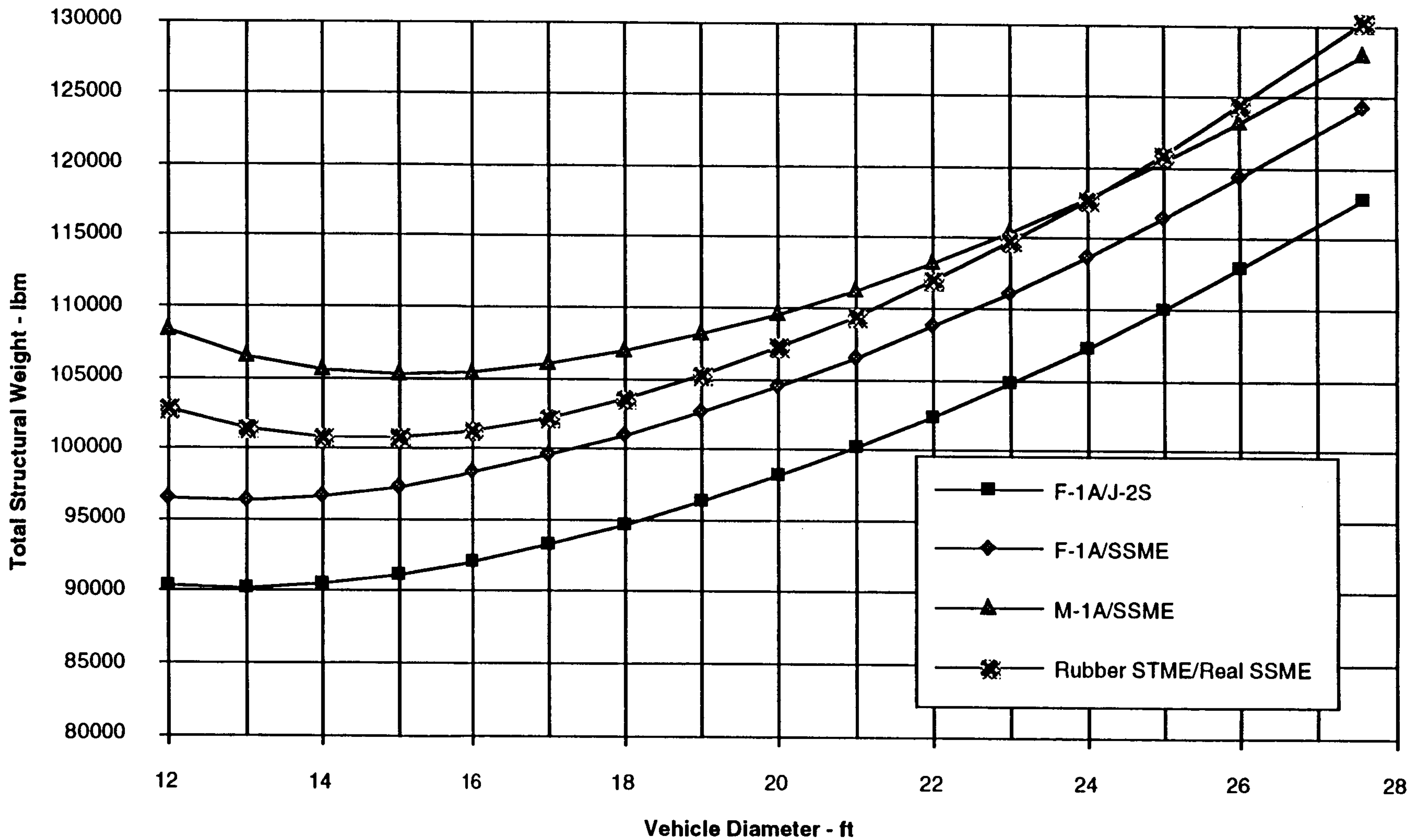
Payload as a Function of Second Stage Initial Acceleration

(F-1A/SSME, No1= 1.565g, No2= 1.423 g)



K. A. Holden
205-722-4531

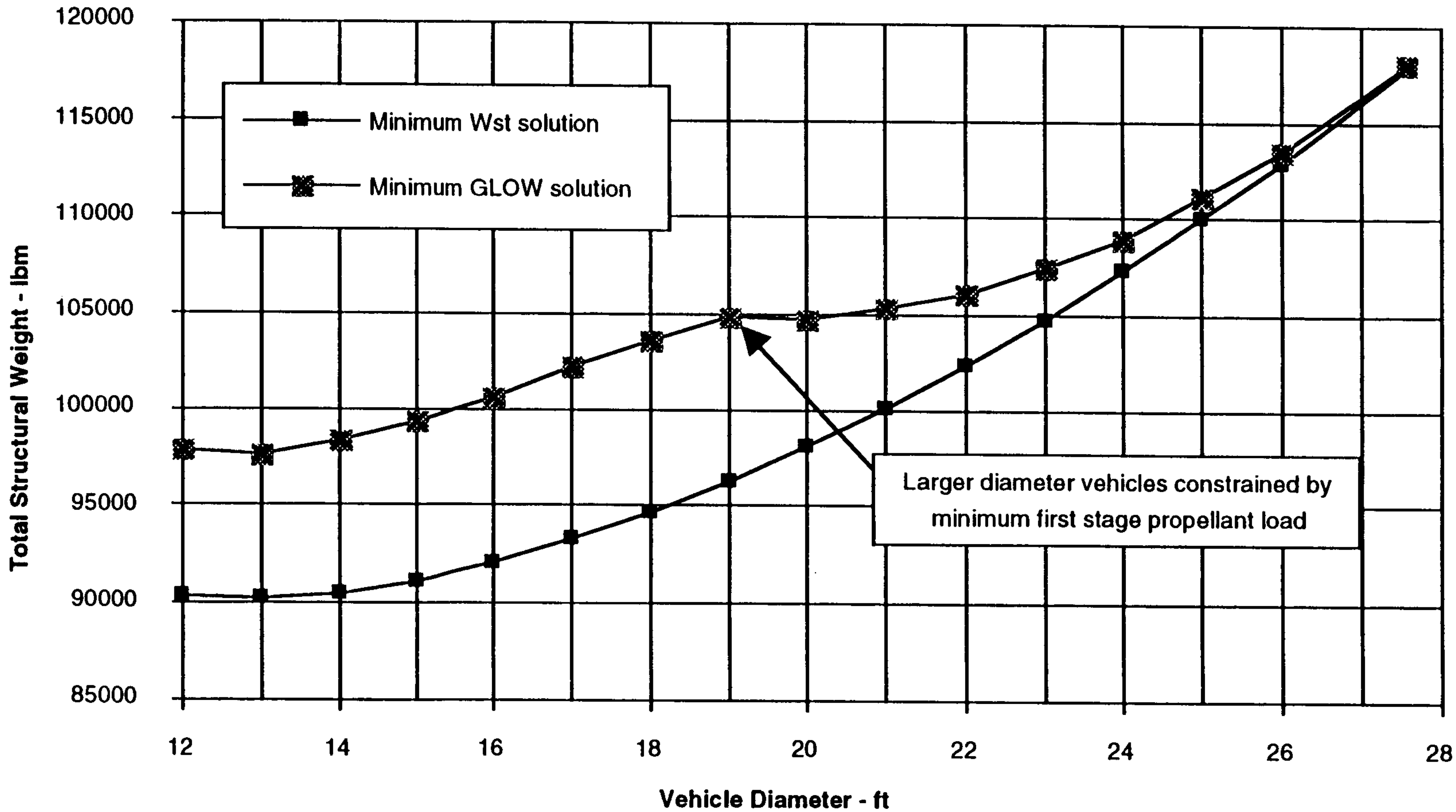
Vehicle Structural Weights



K. A. Holden
205-722-4531

Vehicle Structural Weight as a Function of Diameter

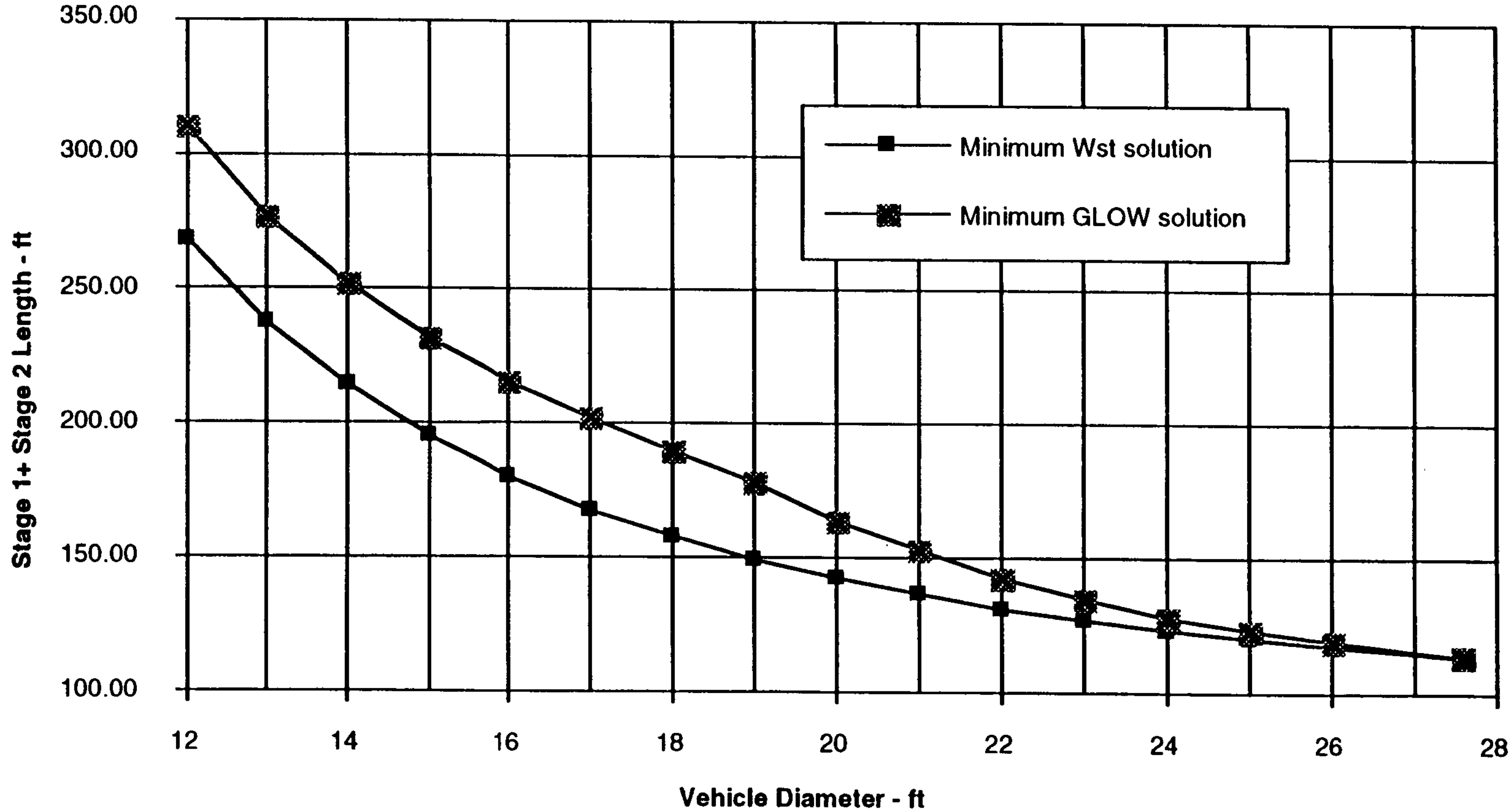
(50 k F-1A/J-2S)



K. A. Holden
205-722-4531

Vehicle Length as a Function of Diameter - ft

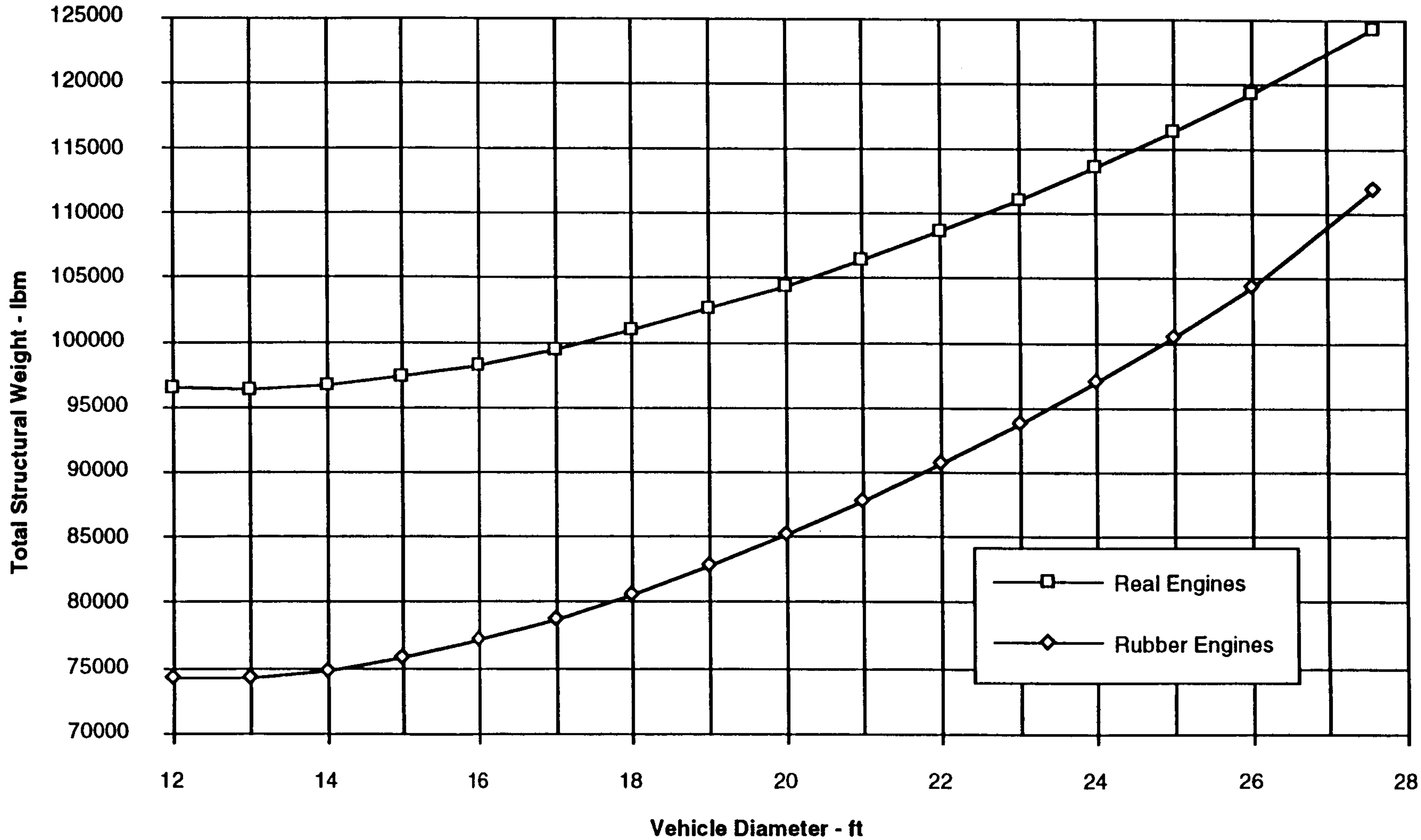
(50 k F-1A/J-2S)



K. A. Holden
205-722-4531

Vehicle Structural Weight as a Function of Diameter

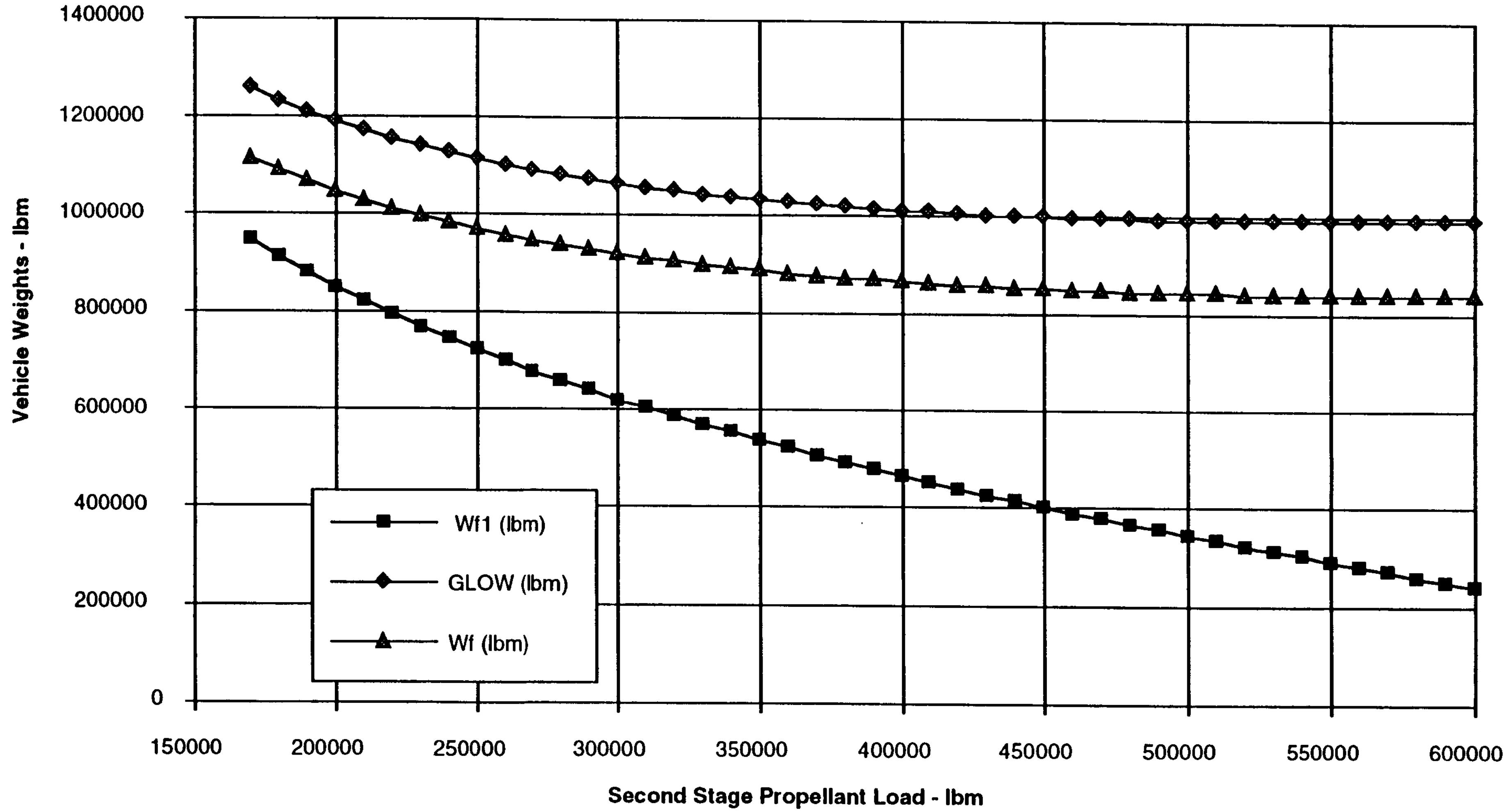
(50 k F-1A/SSME)



K. A. Holden
205-722-4531

Vehicle Weights as a Function of Second Stage Propellant Load

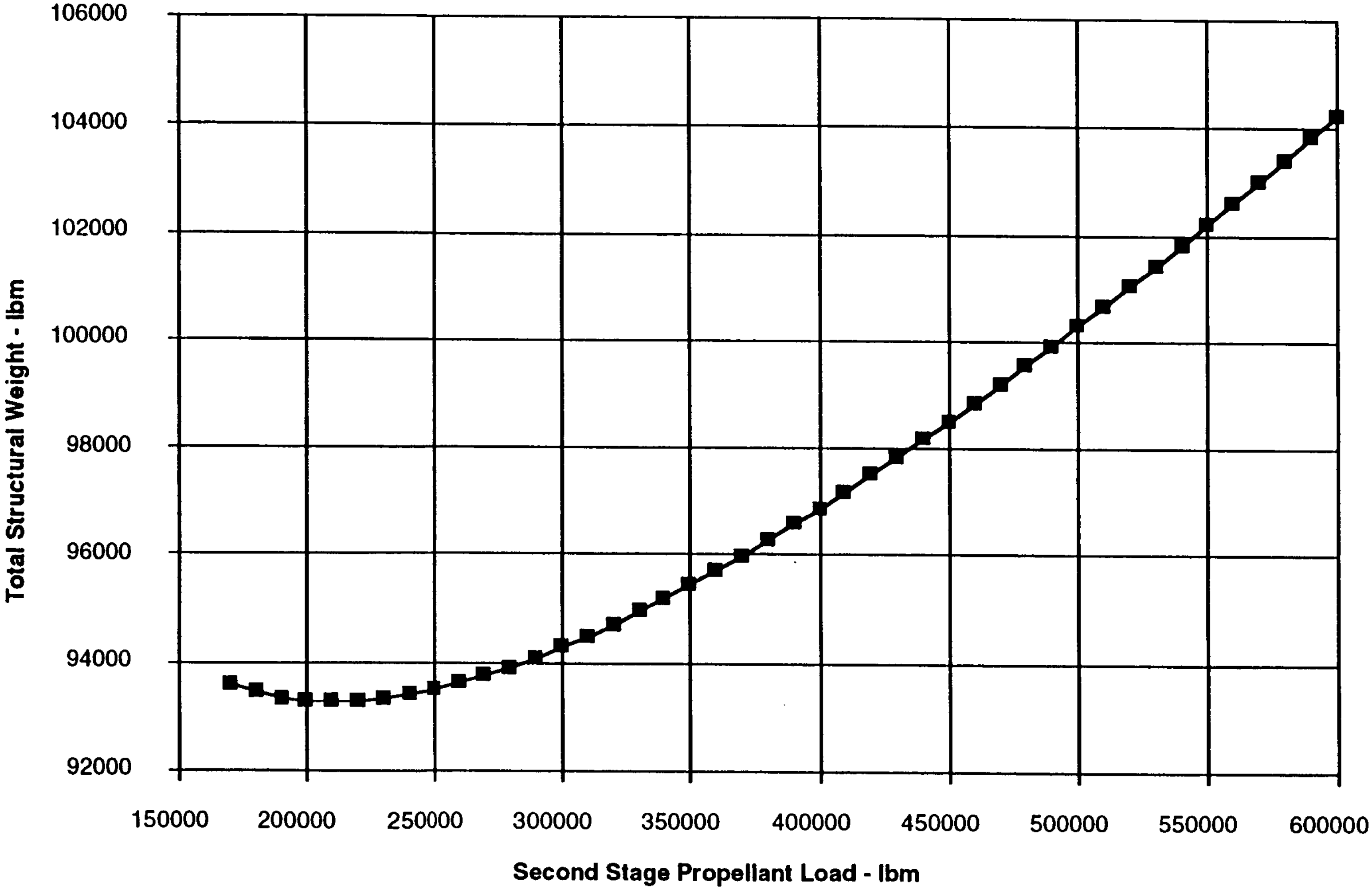
(17 foot diameter vehicle with F-1A and J-2S engines)



K. A. Holden
205-722-4531

Structural Weight as a Function of Second Stage Propellant Load

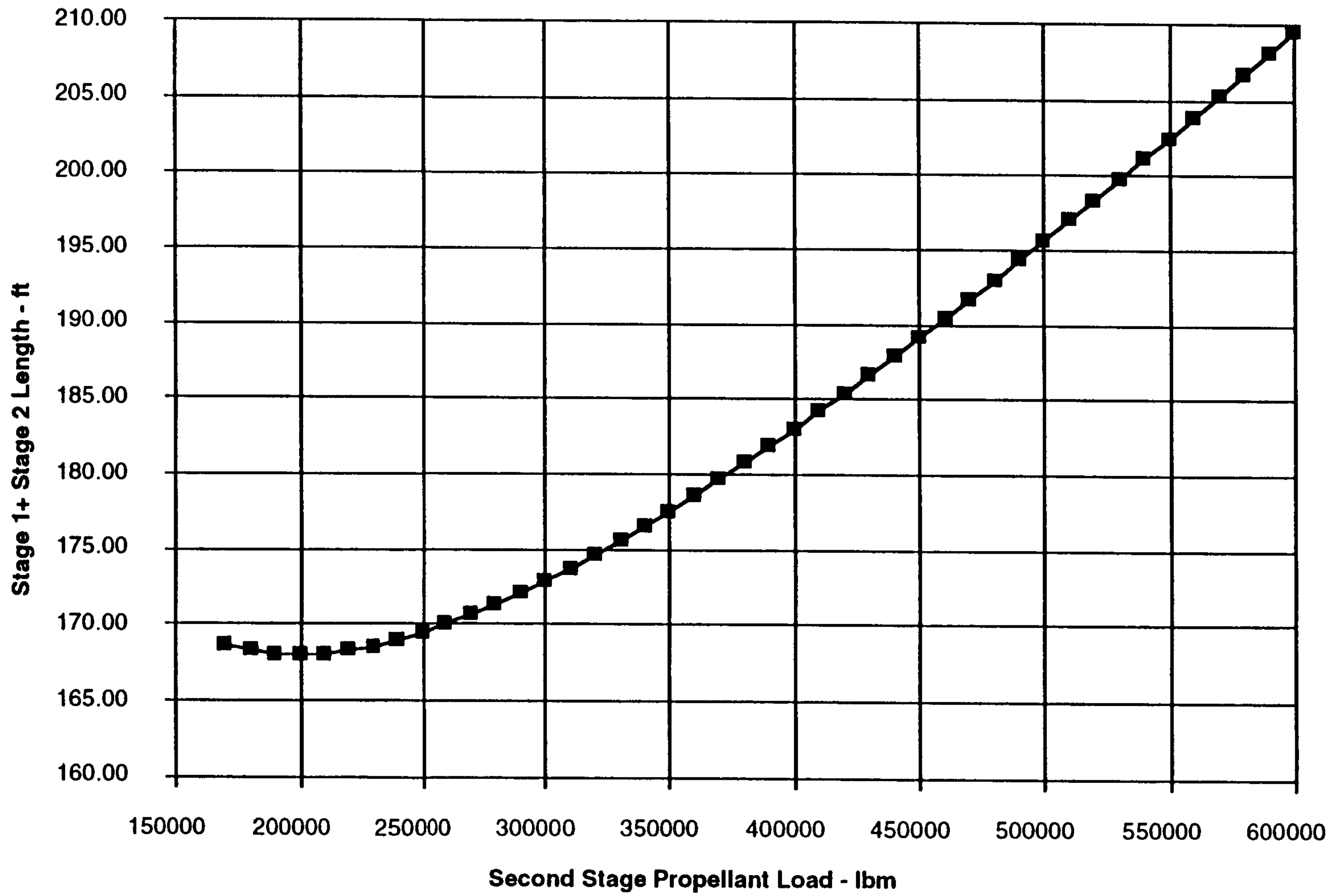
(17 foot diameter vehicle with F-1A and J-2S engines)



K. A. Holden
205-722-4531

Vehicle Length as a Function of Second Stage Propellant Load

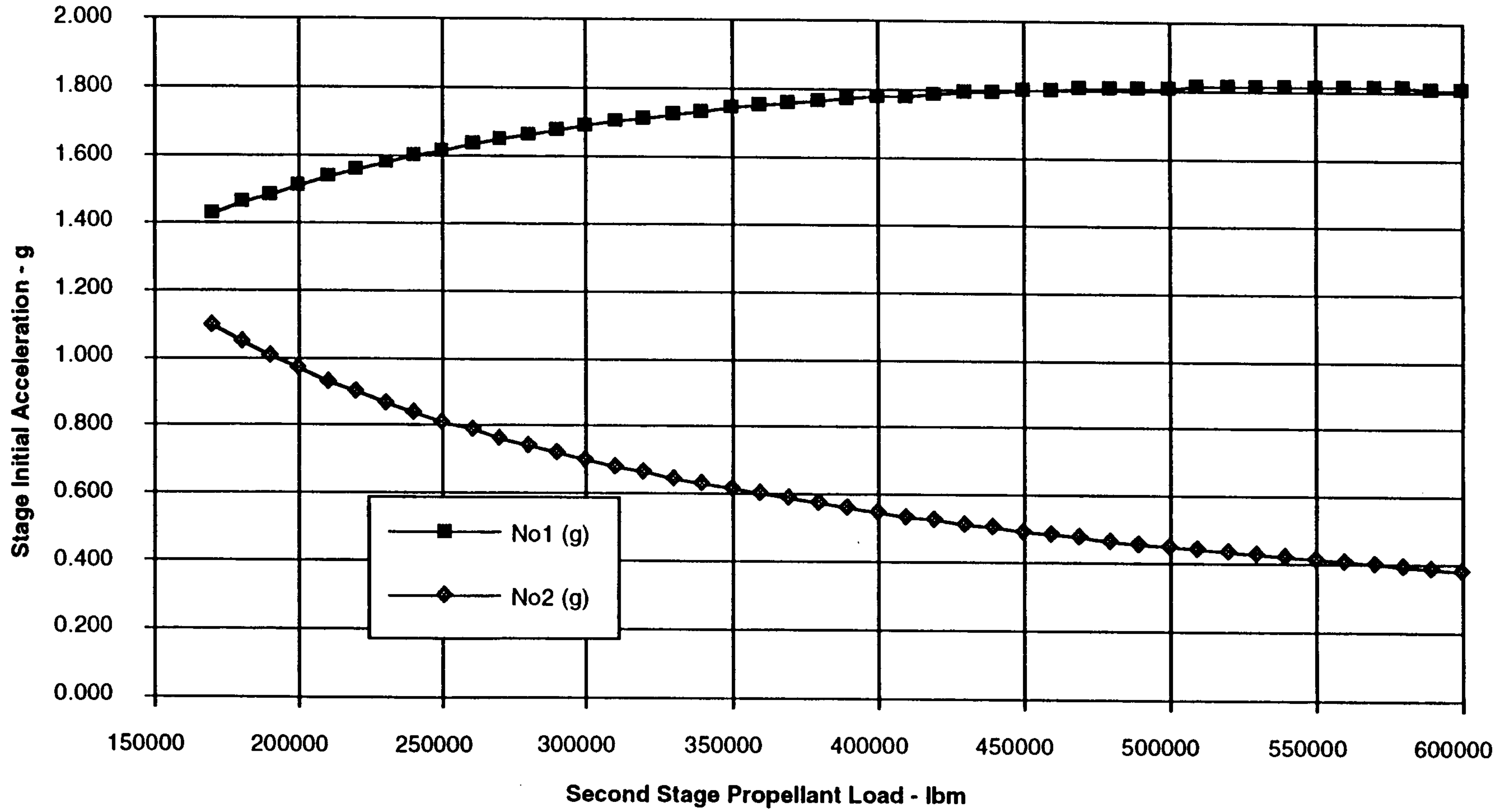
(17 foot diameter vehicle with F-1A and J-2S engines)



K. A. Holden
205-722-4531

Stage Initial Acceleration as a Function of Second Stage Propellant Load

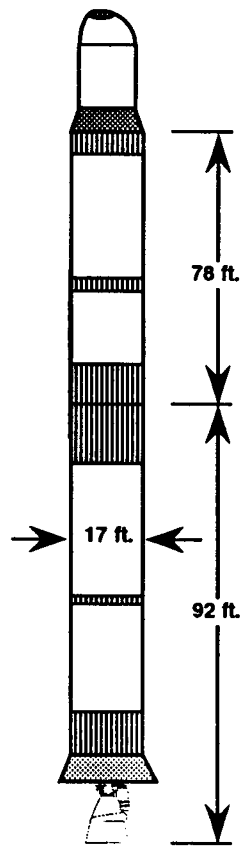
(17 ft diameter vehicle with F-1A and J-2S engines)



K. A. Holden
205-722-4531

7. 50/80K Performance

Preliminary Data
50K Vehicle, F-1A/SSME



Length/Diameter= 10.0

Payload:	58,800 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,166,286 lbm

First Stage:

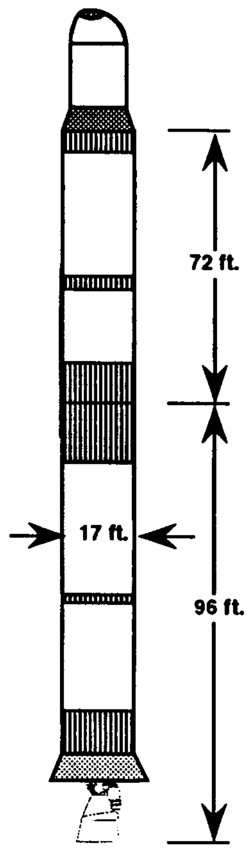
Inert Mass:	68,205 lbm
Usable Propellant:	761,978 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.522 g

Second Stage:

Inert Mass:	31,303 lbm
Usable Propellant:	236,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.423 g

K. A. Holden
205-722-4531

Preliminary Data
50K Vehicle, F-1A/ J-2S



Length/Diameter= 9.9

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,172,892 lbm

First Stage:

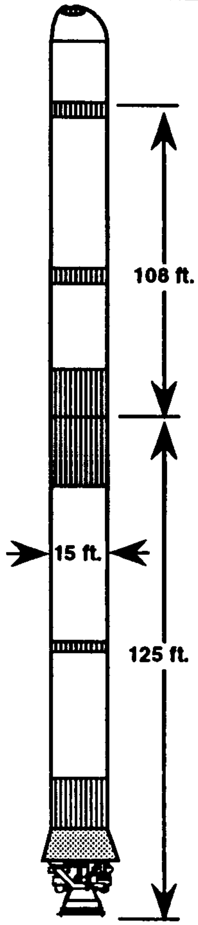
Inert Mass:	68,458 lbm
Usable Propellant:	819,635 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.535 g

Second Stage:

Inert Mass:	24,799 lbm
Usable Propellant:	210,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.930 g

K. A. Holden
205-722-4531

Preliminary Data, 50K Vehicle, Rubber STME/SSME



Length/Diameter= 15.5

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 828,442 lbm

First Stage:

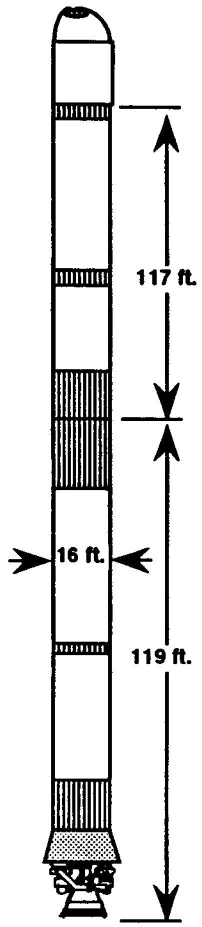
Inert Mass:	66,648 lbm
Usable Propellant:	377,669 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	15.0 ft
Thrust/Weight:	1.400 g
Sea Level Thrust:	1,159,819 lbf

Second Stage:

Inert Mass:	34,126 lbm
Usable Propellant:	300,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	15.0 ft
Thrust/Weight:	1.224 g

K. A. Holden
205-722-4531

Preliminary Data, 50K Vehicle, M-1A/SSME



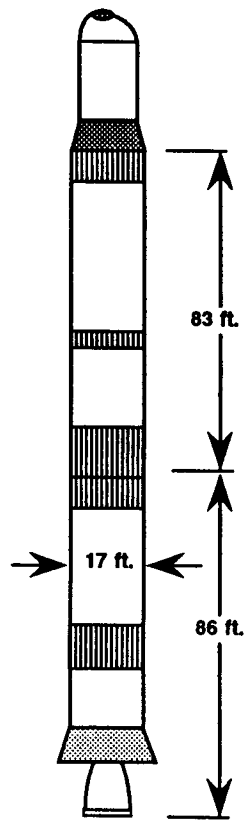
Length/Diameter= 14.8

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	854,128 lbm
First Stage:	
Inert Mass:	67,039 lbm
Usable Propellant:	318,712 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	M-1A/1
Diameter:	16.0 ft
Thrust/Weight:	1.522 g
Second Stage:	
Inert Mass:	38,377 lbm
Usable Propellant:	380,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	16.0 ft
Thrust/Weight:	1.003 g

K. A. Holden
205-722-4531

Preliminary Data, 50K Vehicle, Staged Combustion Hybrid/J-2S



Length/Diameter= 9.9

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,222,969 lbm

First Stage:

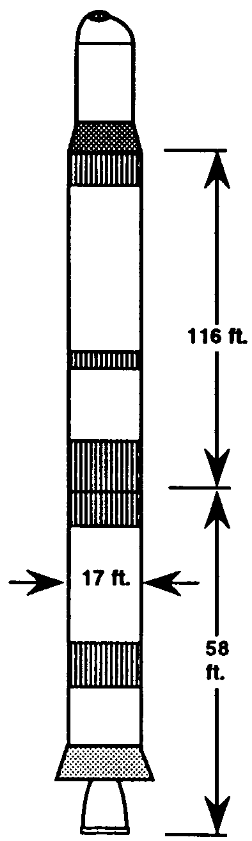
Inert Mass:	111,561 lbm
Usable Propellant:	774,091 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.472 g
Sea Level Thrust:	1,800,000 lbf

Second Stage:

Inert Mass:	27,317 lbm
Usable Propellant:	240,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.788 g

K. A. Holden
205-722-4531

Preliminary Data, 50K Vehicle, Staged Combustion Hybrid/SSME



Length/Diameter= 10.2

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 998,456 lbm

First Stage:

Inert Mass:	71,101 lbm
Usable Propellant:	417,571 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.402 g
Sea Level Thrust:	1,400,000 lbf

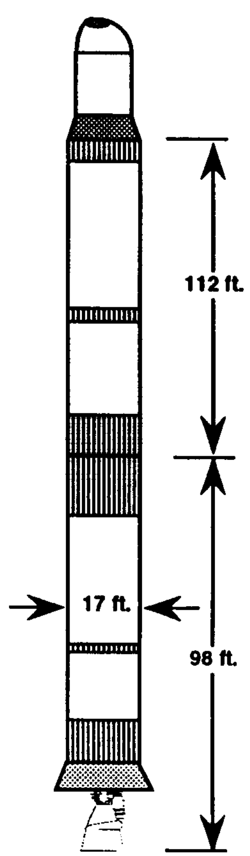
Second Stage:

Inert Mass:	40,436 lbm
Usable Propellant:	420,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.921 g

K. A. Holden
205-722-4531

Preliminary Data

80K Vehicle, F-1A/SSME



Length/Diameter= 12.4

Payload:	78,500 lbm (35.6 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,437,645 lbm

First Stage:

Inert Mass:	69,734 lbm
Usable Propellant:	850,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.252 g

Second Stage:

Inert Mass:	39,411 lbm
Usable Propellant:	400,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.943 g

K. A. Holden
205-722-4531

8. Cost

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Cost Assessment Status

- Historic vehicle cost trends being researched as function of diameter, propellant type, and engine type

- DDT& E and theoretical first unit (TFU) cost work breakdown structure developed at subsystem level for generic two-stage launch vehicle

- Preliminary cost sensitivity data generated for family of F-1A/SSME 50K vehicles
  --Stage diameter varied from 12 ft to 27.6 ft (1 ft increments)
  --Results currently being plotted

- Participated in brainstorming of candidate 50/80K configuration options

- Presentation being prepared on "what DDT&E is good DDT&E"

- Will participate with MSFC cost personnel in benchmarking DDT&E and TFU cost of current "reference" 50K in-house design (F-1A/J-2S)
  --Insures commonality of groundrules and assumptions for vehicle costing
  --Allows calibration of subsystem costing algorithms for future cost comparisons and independent costing of Lockheed's vehicle concepts

J. B. McCurry
205-722-4509

9. Ground Operations

Ground Operations Assessment Status

- FLO Dual Launch ground processing assessment performed
- Various mixed-fleet FLO Single Launch ground processing scenarios assessed as a function of type and number of processing facilities available
  --DDT&E, operational flexibility, and programmatic risk affected
- Alternative launch sites assessed for first-order figures of merit (cost, schedule performance, supportability, overflight hazards, security)
  --Short list of "desirable" sites derived for future detailed assessments
- Initiated ground operations assessment of MSFC's reference Early Heavy Lift Vehicle derived lunar vehicle configurations
  --8 single F-1A boosters around EHLLV (3 SSME) core
  --7 single RD-170 boosters around EHLLV (3 SSME) core
- Developed first draft of Integrated Logistics Support Plan document
  --Vehicle configuration independent
- Developed outline for generic Ground Operations Support Plan document

J. B. McCurry
205-722-4509

10. Vehicle Health Management

VHM Requirements Assessment Task

- Brainstorming of candidate areas of VHM requirements analysis, with respect to launch vehicle design, identified ground operations as highest leveraging/benefit arena
  - -- Considered opportunities for VHM from component manufacturing phase through flight, under Lockheed's end-to-end VHM philosophy being used on F-22 (ATF)
  - -- Leverages current experience and lessons-learned in Shuttle processing (LSOC)
  - -- Opportunities for vehicle-based VHM based on manufacturing experience will also be addressed
- Approach is to identify vehicle design attributes and requirements for VHM that will significantly improve ground operations recurring costs
  - -- VHM technology demonstration candidates also to be identified
- Major functions have been identified for a complete flow from manufacturing through ground operations
- Prepared an approach for VHM technology bridging and provided to F. Huffaker/PT01
- Identified six near-term candidates for VHM technology demonstration
  - -- Orbiter LH2 prevalve automated checkout appears to be best candidate
- Currently identifying launch vehicle VHM requirements for each of ground operations major functions
- Areas of VHM application to manufacturing being investigated based on Fleet Ballistic Missile lessons-learned
  - --Other manufacturing elements of Lockheed will also be contacted

J. B. McCurry
205-722-4509

11. Technology Priorities

Technology Development/Demonstration Priorities for Heavy Lift Launch Vehicles

Priority	Technology Development/Demonstration Area	Justification/Benefit
1	High-Reliability Propulsion Feed & Pressurization Subsystems	Reduced cost-of-failure, increased probability of mission success, decreased ground processing recurring costs, and decreased processing schedule risk
2	Low-Cost High-Thrust (>1 Million lbf) Engine	Reduced propellant feed complexity, increased integrated propulsion reliability and decreased cost, decreased ground processing recurring costs
3	Non-Intrusive Vehicle Health Management	Increased integrated stage reliability, decreased ground processing recurring costs
4	Operationally Efficient Modular Propellant Feed Subsystem	Increased integrated stage reliability, decreased integrated propulsion cost, decreased ground processing recurring costs, and decreased processing schedule risk
5	Alternative Construction Methods & Tooling for Very Large-Scale Launch Vehicle Stage Structure	Reduced stage size, propulsion sizing, stage recurring costs, facilities/tooling/infrastructure costs
6	Advanced Light-Weight/High-Strength Structural Materials	Reduced stage size, propulsion sizing, and stage recurring costs
7	Low-Cost Space-Rated Cryogenic Storage & Transfer Device	Reduced lunar lander sizing, and enabling for Mars missions and alternative space-based vehicle assembly/operations scenarios

J. B. McCurry
205-722-4509

Technology Development/Demonstration Priorities for Heavy Lift Launch Vehicles (Concluded)

Priority	Technology Development/Demonstration Area	Justification/Benefit
8	Automated Ground Processing and Vehicle Check-Out	Reduced cost-of-failure, increased probability of mission success, decreased ground processing recurring costs, and decreased schedule risk
9	Electrical Actuation	Reduced control subsystem complexity, increased reliability, decreased ground processing recurring costs, and decreased schedule risk
10	Integrated Autonomous Guidance and Navigation	Reduced pre-flight and mission operations recurring costs, and increased probability of mission success
11	Operable Low-Cost Open Architecture Avionics	Increased design robustness and obsolescence avoidance, decreased ground processing recurring costs, and decreased schedule risk
12	Advanced Non-Destructive Evaluation Analysis/Techniques	Increased stage reliability, reduced manufacturing and ground processing recurring costs, and decreased schedule risk
13	Laser-Initiated Pyrotechnics	Increased safety, increased reliability, decreased ground processing recurring costs, and decreased schedule risk

J. B. McCurry
205-722-4509

50 K Launch Vehicle Definition in Support of the Access to Space Panels

February 1993

Lockheed

J. B. McCurry
205-722-4509

Study Approaches for Analysis of New 50-80K Vehicles

Three Basic Mission Scenarios

1. Titan IV Replacement
   - 40K lbm payload mass
   - 80 x 150 nm 28.5 deg. ETR direct insertion (80 nm MECO)
   - Standard Titan IV shroud (85.8 ft x 16.7 ft) with jettison @ 400K ft geodetic

2. PLS/CTRV
   - 50-100 K lbm payload mass
   - 31 x 220 nm direct insertion (57 nm MECO)
     --28.5 deg. ETR
     --51.6 deg. (CIS or VAFB launch)
     --33 deg. ETR ?

3. Single Launch Lunar (First Lunar Outpost)
   - 93 mt payload mass post-TLI
   - 100 nm circ. 28.5 deg. ETR direct insertion (100 nm MECO

J. B. McCurry
205-722-4509

Performance Assessment Groundrules

- 3-DOF ascent trajectory optimization
- Nominal performance (no engines out)
- Payload mass maximized subject to ascent constraints
- Each stage can be step-throttled for thrust acceleration limiting
- Lofting and step-throttling used for dynamic pressure limiting
- Stage flight performance reserve sized as 1 % of stage delta V
  --Each stage has its own FPR
- Dynamic pressure constraint = 900 psf
- Q-alpha constraint = +/- 5000 psf-deg
- Thrust acceleration constraint = 4 Gs
- No winds, '63 Patrick standard atmosphere

J. B. McCurry
205-722-4509

80K Configuration Definition

- Pick "best" 50K configuration

  --Add engine(s) and propellant tank barrel section(s)

  --Add strap-on booster(s)
  - •Solid
  - •Liquid
  - •Hybrid

J. B McCurry
205-722-4509

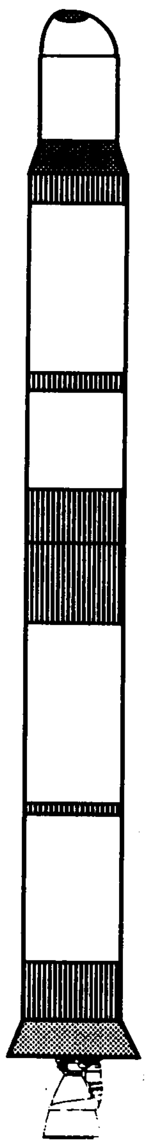
50 K lbm Vehicle Options

50 K Configuration	Cycle 1 Sizing	Cycle 1 Trajectory	Cycle 2 Sizing	Cycle 2 Trajectory
F-1A/SSME	X	X		
F-1A/J-2S	X			
F-1A/Vulcain (HM-60)				
M-1A/SSME	X			
M-1A/J-2S				
Rubber STME/SSME	X			
Rubber STME/J-2S				
Rubber STME/Rubber STME				
Staged Comb. Hybrid/SSME	X			
Staged Comb. Hybrid/J-2S	X			
Classical Hybrid/SSME				
Classical Hybrid/J-2S				
RD-170/RD-0120				

Last Update 2/3/93

J. B. McCurry
205-722-4509

Major Design Aspects (Typical Examples)



Outter Moldline Effects
- --Static Stability (gimbal range, C.G. envelop, passive stabilization)
- --First Bending Mode Stiffness
- --Base Drag
- --Load Relief (alpha/beta constraints)
- --Base Heating
- --Unsteady Aerodynamic Loads

Shroud/Payload Concept
- --Existing Shroud
- --Unshrouded Payload (PLS/ICTRV/MCTRV)
- --Optimized Shroud

Stage Propellant Tank Design
- --Conventional
- --Semitorroidal
- --Common/Nested Bulkhead

Construction Method
- --Mechanical fasteners
- --Adhesive bonding
- --Welding

Primary Structure Material
- --Aluminum/Aluminum-Lithium
- --Composites

Intertank/Interstage Design

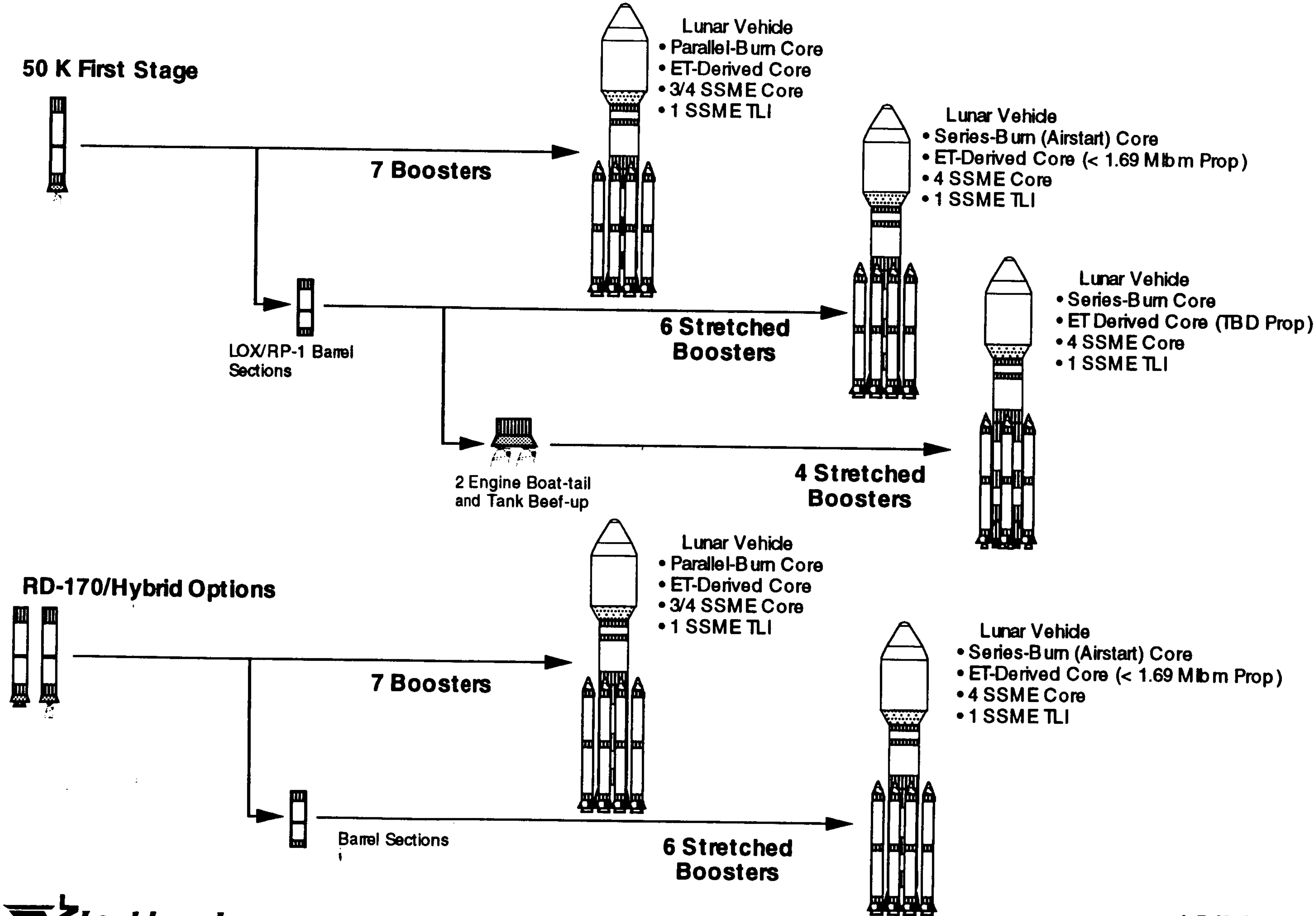
Stage Thrust Structure Design

Propellant Feed Subsystem Design

Main Stage Propulsion Type

J. B. McCurry
205-722-4509

50 K Evolution to Lunar Vehicle

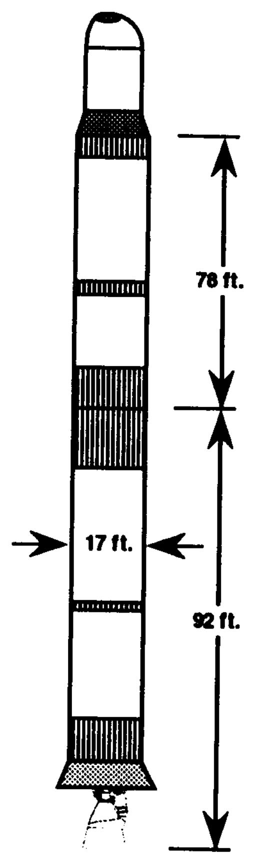


J. B. McCurry
205-722-4509

Sample Candidate Configurations

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, F-1A/SSME



Length/Diameter= 10.0

Payload:	58,800 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,166,286 lbm

First Stage:

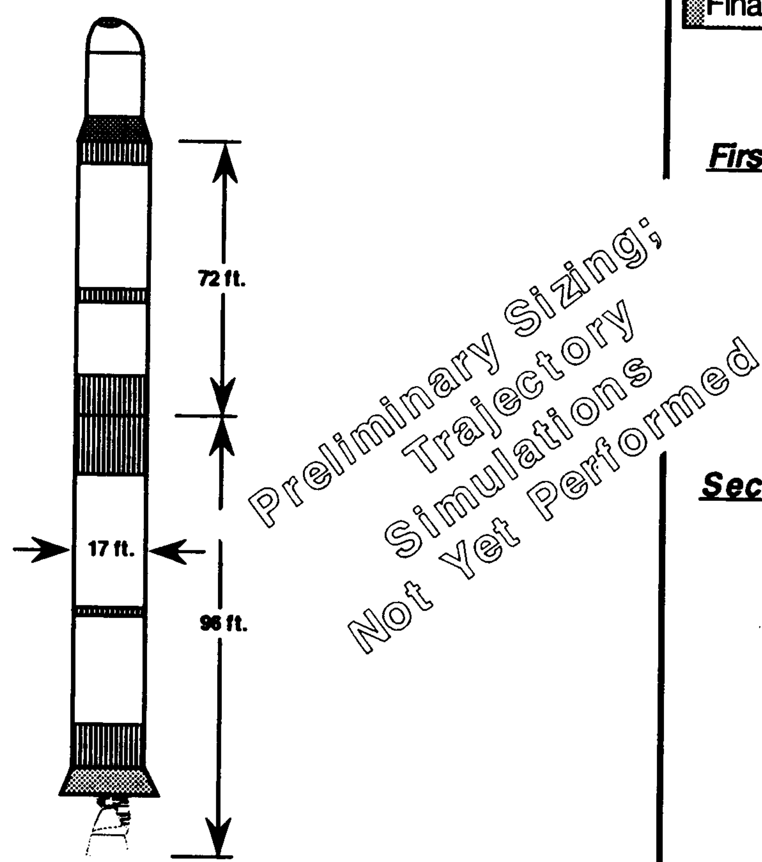
Inert Mass:	68,205 lbm
Usable Propellant:	761,978 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.522 g

Second Stage:

Inert Mass:	31,303 lbm
Usable Propellant:	236,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.423 g

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, F-1A/ J-2S



Length/Diameter= 9.9

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,172,892 lbm

First Stage:

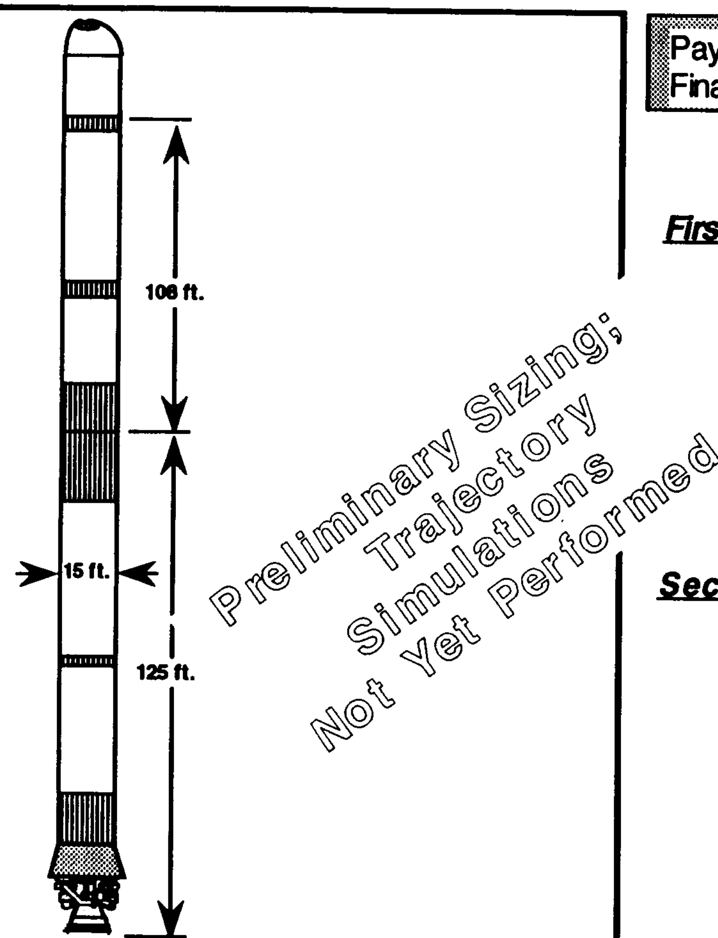
Inert Mass:	68,458 lbm
Usable Propellant:	819,635 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.535 g

Second Stage:

Inert Mass:	24,799 lbm
Usable Propellant:	210,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.930 g

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, Rubber STME/SSME



Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 828,442 lbm

First Stage:

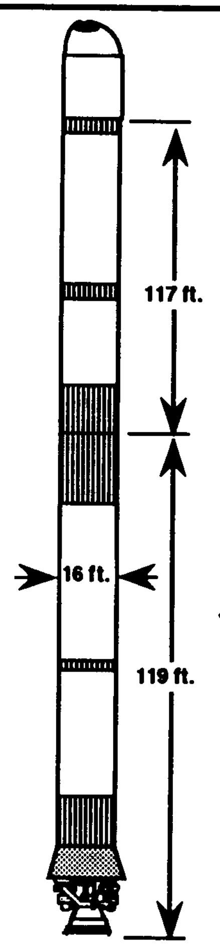
Inert Mass:	66,648 lbm
Usable Propellant:	377,669 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	15.0 ft
Thrust/Weight:	1.400 g
Sea Level Thrust:	1,159,819 lbf

Second Stage:

Inert Mass:	34,126 lbm
Usable Propellant:	300,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	15.0 ft
Thrust/Weight:	1.224 g

J. B McCurry
205-722-4509

Preliminary Data
50 k Vehicle, M-1A/SSME



Length/Diameter= 14.8

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 854,128 lbm

First Stage:

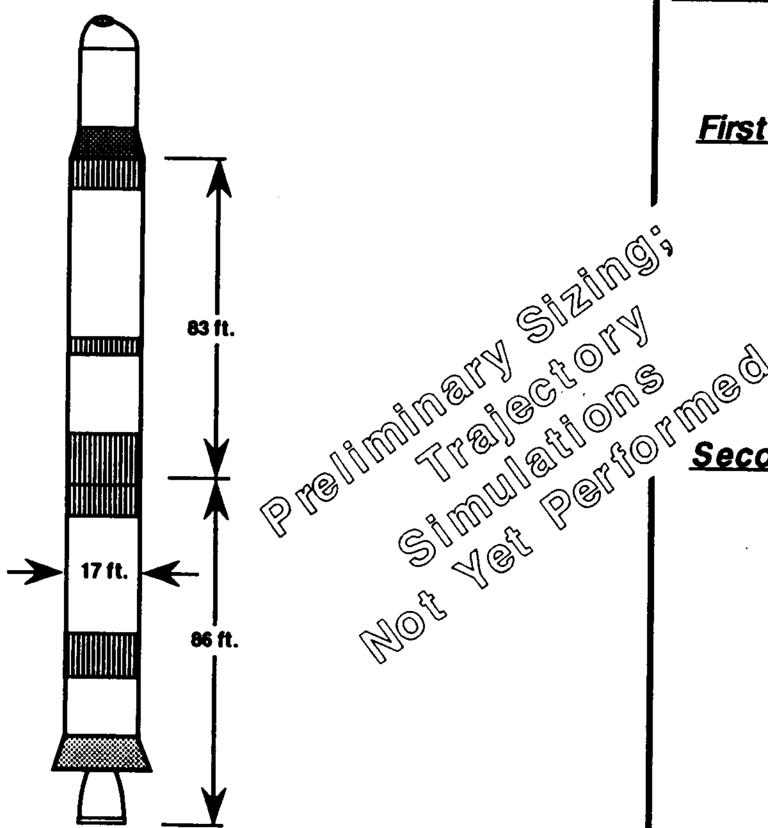
Inert Mass:	67,039 lbm
Usable Propellant:	318,712 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	M-1A/1
Diameter:	16.0 ft
Thrust/Weight:	1.522 g

Second Stage:

Inert Mass:	38,377 lbm
Usable Propellant:	380,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	16.0 ft
Thrust/Weight:	1.003 g

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, Staged Combustion Hybrid/J-2S



Length/Diameter= 9.9

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,222,969 lbm

First Stage:

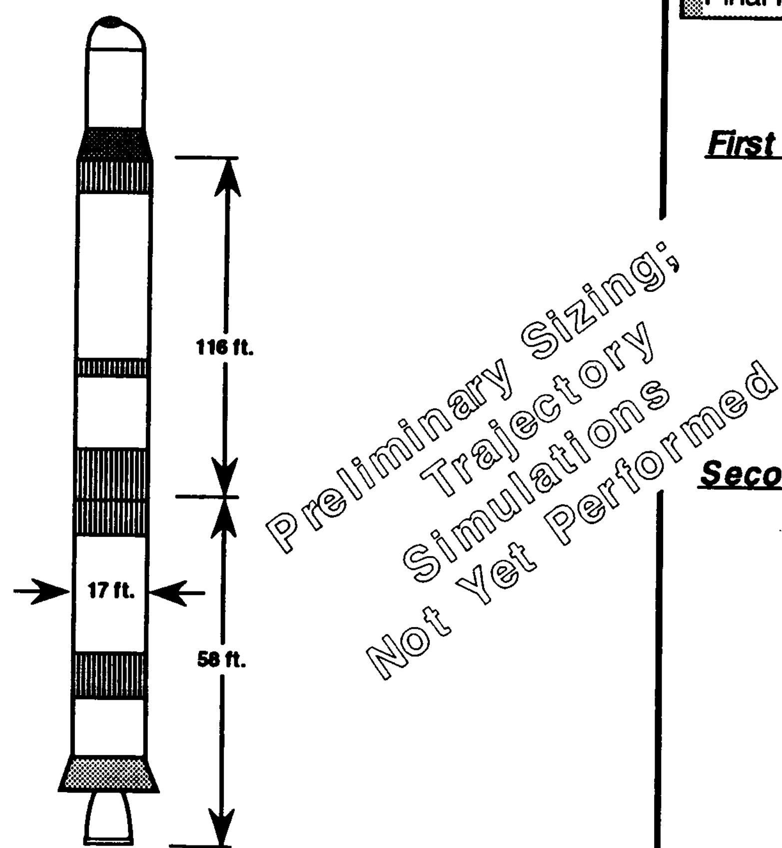
Inert Mass:	111,561 lbm
Usable Propellant:	774,091 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.472 g
Sea Level Thrust	1,800,000 lbf

Second Stage:

Inert Mass:	27,317 lbm
Usable Propellant:	260,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.788 g

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, Staged Combustion Hybrid/SSME



Length/Diameter= 10.2

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 999,805lbm

First Stage:

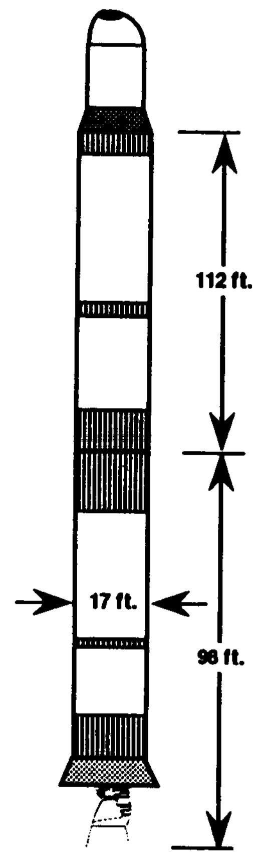
Inert Mass:	71,798 lbm
Usable Propellant:	417,571 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.402 g
Sea Level Thrust	1,400,000 lbf

Second Stage:

Inert Mass:	40,436 lbm
Usable Propellant:	420,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.921 g

J. B. McCurry
205-722-4509

Preliminary Data
80 k Vehicle, F-1A/SSME



Length/Diameter= 12.4

Payload:	78,500 lbm (35.6 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,437,645 lbm

First Stage:

Inert Mass:	69,734 lbm
Usable Propellant:	850,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.252 g

Second Stage:

Inert Mass:	39,411 lbm
Usable Propellant:	400,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.943 g

J. B. McCurry
205-722-4509

Advanced Transportation System Studies
Technical Area 2
Heavy Lift Launch Vehicle Development

Technical Interchange Meeting 11

Langley Research Center
11 February 1993

Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

Summary Of Large LOX/Hydrocarbon And LOX/LH2 Rocket Engines* (200K and Larger)

Engine	Country	Fuel	Oxidizer	Thrust, Klbf**		Specific Impulse, Sec.**	
				Sea Level	Vacuum	Sea Level	Vacuum
F-1	US	RP-1	LOX	1522	1746	265	304
MA-5A***	US	RP-1	LOX	423.5/50.5	473.4/85.0	264/220	295/309
RS-27A	US	RP-1	LOX	200	237	255	302
H-1	US	RP-1	LOX	205	230	264	296
LR-87	US	RP-1	LOX	300	344.4	252	289
XLR-109	US	RP-1	LOX	500	---	265	---
J-2	US	LH2	LOX	161.4	230	293.7	422.7
J-2S	US	LH2	LOX	201	265	330	435
M-1	US	LH2	LOX	---	1500	---	428
M-1A	US	LH2	LOX	1300	---	344.5	---
SSME	US	LH2	LOX	373.5	468.4	362	454

NOTES:
* Data Source: Ed Bair,Aerojet
** 100 percent rated power level
***MA-5A Data: Booster(2 Thrust Chambers)/Sustainer (1 Thrust Chamber)

J. B. McCurry
205-722-4509

Summary Of Large LOX/Hydrocarbon And LOX/LH2 Rocket Engines* (200K and Larger)

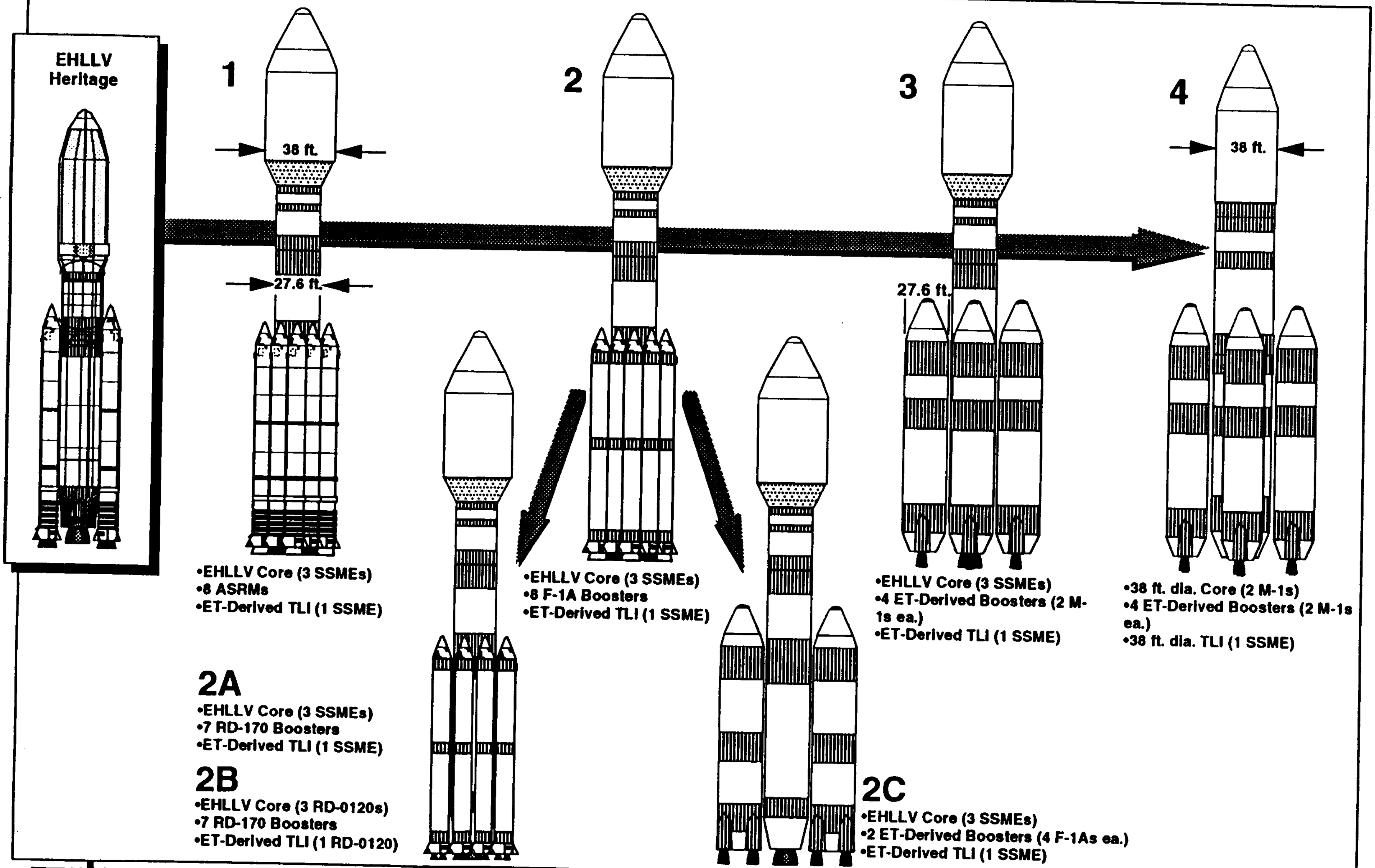
Engine	Country	Fuel	Oxidizer	Thrust, Klbf**		Specific Impulse, Sec.**	
				Sea Level	Vacuum	Sea Level	Vacuum
RD-107	CIS	Kerosene	LOX	184.6	224.8	257	314
RD-108	CIS	Kerosene	LOX	167.5	211.5	248	315
RD-170	CIS	Kerosene	LOX	1631	1777	309	337
RD-120	CIS	Kerosene	LOX	---	181.5	---	350
NK-33	CIS	Kerosene	LOX	339	378	297	331
NK-43	CIS	Kerosene	LOX	---	395	---	346
RD-0120	CIS	LH2	LOX	326	441	354	452.5

NOTES:
* Data Source: Ed Bair,Aerojet
** 100 percent rated power level

Lockheed, Aerojet, ECON

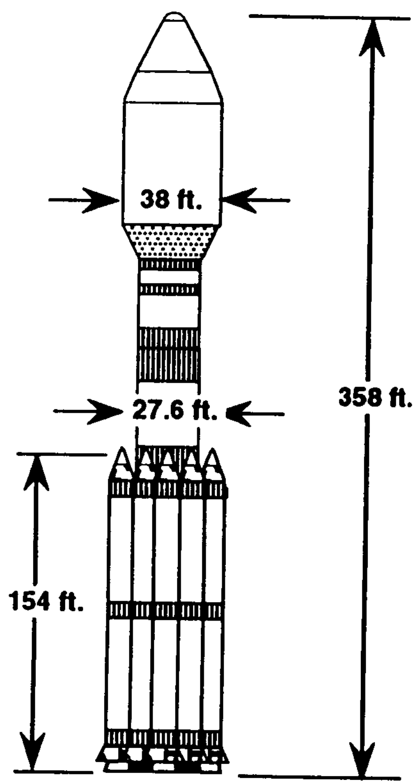
J. B. McCurry
205-722-4509

Lunar Mission, Parallel-Burn Configuration Options



J. B. McCurry
205-722-4509

Preliminary Data
Parallel Burn FLO Vehicle 2



• EHLLV Core (3 SSMEs)
• 8 F-1A Boosters
• EHLLV-Derived TLI (1 SSME)

Payload:	205,000 lbm (93 t)
Final Position:	TLI

GLOW: 11,094,236 lbm

CORE:

Inert Mass:	153,935 lbm
Usable Propellant Mass:	1,678,840 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/3
Diameter:	27.6 ft
Shroud Jettison Mass:	28 K lbm

BOOSTER:

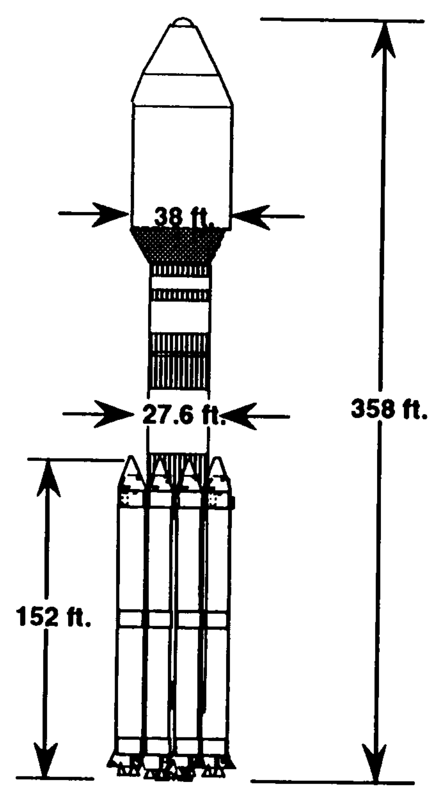
Number/Type:	8 Single F-1A Boosters
Inert Mass:	68,341 lbm
Usable Propellant:	950,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	15.0 ft

TLI Stage:

Inert Mass:	61,157 lbm
Usable Propellant Mass:	600,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	27.6 ft

J. B. McCurry
205-722-4509

Preliminary Data
Parallel Burn FLO Vehicle 2A



- EHLLV Core (3 SSMEs)
- 7 RD-170 Boosters
- EHLLV-Derived TLI (1 SSME)

Payload:Final Position:	205,000 lbm (93t) TLI

GLOW: 11,094,236 lbm

CORE:

Inert Mass:	153,935 lbm
Usable Propellant Mass:	1,678,840 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/3
Diameter:	27.6 ft
Shroud Jettison Mass:	28 K lbm

BOOSTER:

Number/Type:	7 Single F-1A Boosters
Inert Mass:	65,539 lbm
Usable Propellant:	800,000 lbm
Propellant Type:	LOX/SYN10
Engine Type/No.:	RD-170/1
Diameter:	14.0 ft

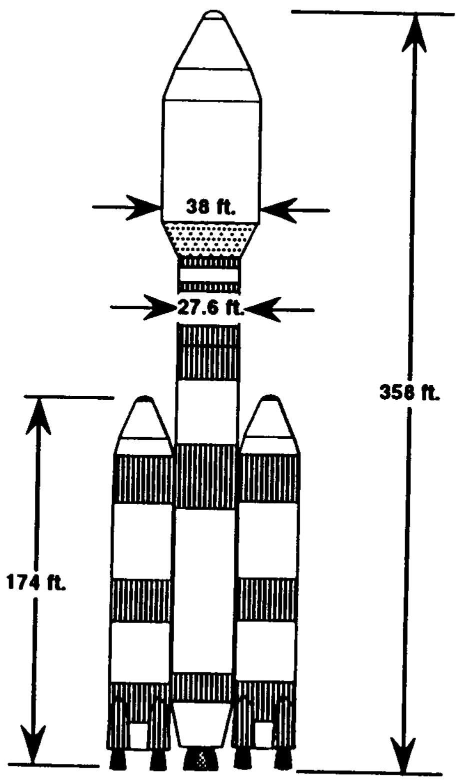
TLI Stage:

Inert Mass:	61,157 lbm
Usable Propellant Mass:	600,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	27.6 ft

Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

Preliminary Data
Parallel Burn FLO Vehicle 2C



Payload:	205,000 lbm (93 t)
Final Position:	TLI

GLOW: 11,056,511 lbm

CORE:

Inert Mass:	153,935 lbm
Usable Propellant Mass:	1,678,840 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/3
Diameter:	27.6 ft
Shroud Jettison Mass:	28 K lbm

BOOSTER:

Number/Type:	2 Boosters with 4 F-1As Each
Inert Mass:	254,501 lbm
Usable Propellant:	3,800,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/4
Diameter:	27.6 ft

TLI Stage:

Inert Mass:	61,157 lbm
Usable Propellant Mass:	600,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	27.6 ft

Lockheed, Aerojet, ECON

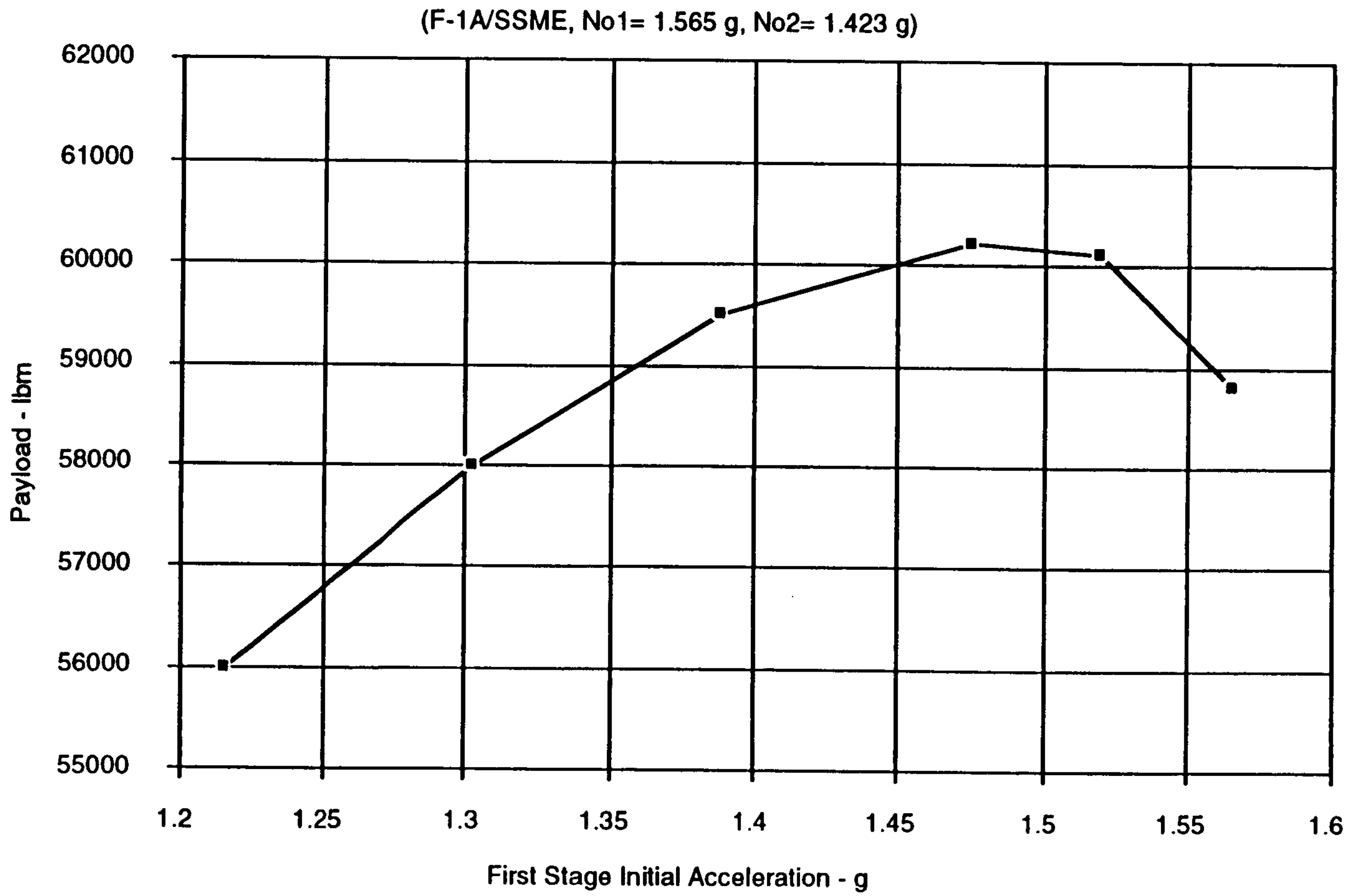
J. B. McCurry
205-722-4509

Performance Assessment Groundrules

- 3-DOF ascent trajectory optimization (SORT tool)
- Nominal performance (no engines out)
- Payload mass maximized subject to ascent constraints
- Each stage can be step-throttled for thrust acceleration limiting
- Lofting and step-throttling used for dynamic pressure limiting
- Stage flight performance reserve sized as 1 % of stage delta V
- Dynamic pressure constraint = 900 psf
- Q-alpha constraint = +/- 5000 psf-deg
- Thrust acceleration constraint = 4 Gs
- ETR launch, 28.5 deg. inclination
- Direct insertion target orbit 15 x 220 nm with SECO at 57 nm
- No winds, '63 Patrick standard atmosphere

J. B. McCurry
205-722-4509

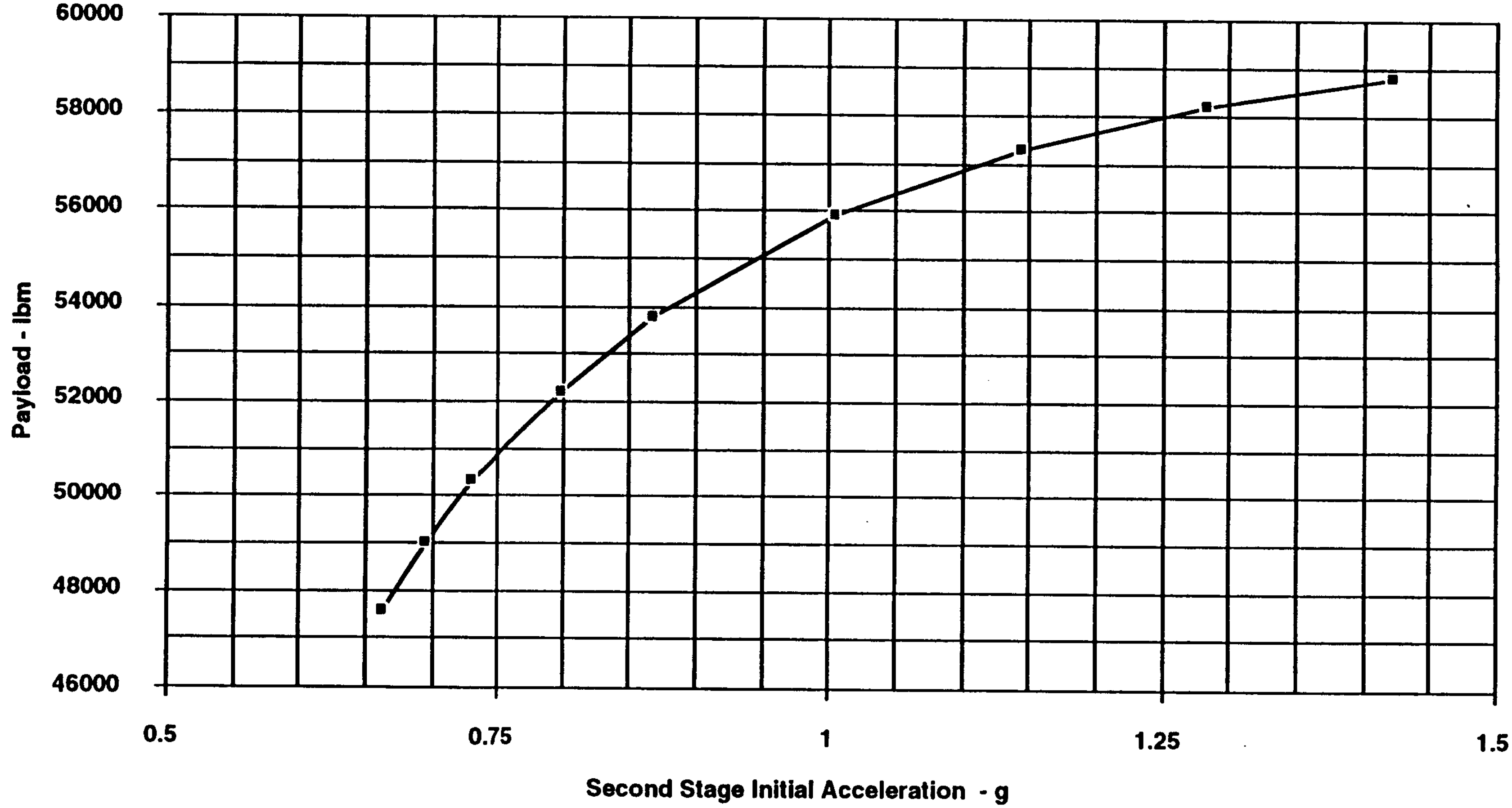
Payload as a Function of First Stage Initial Acceleration



J. B. McCurry
205-722-4509

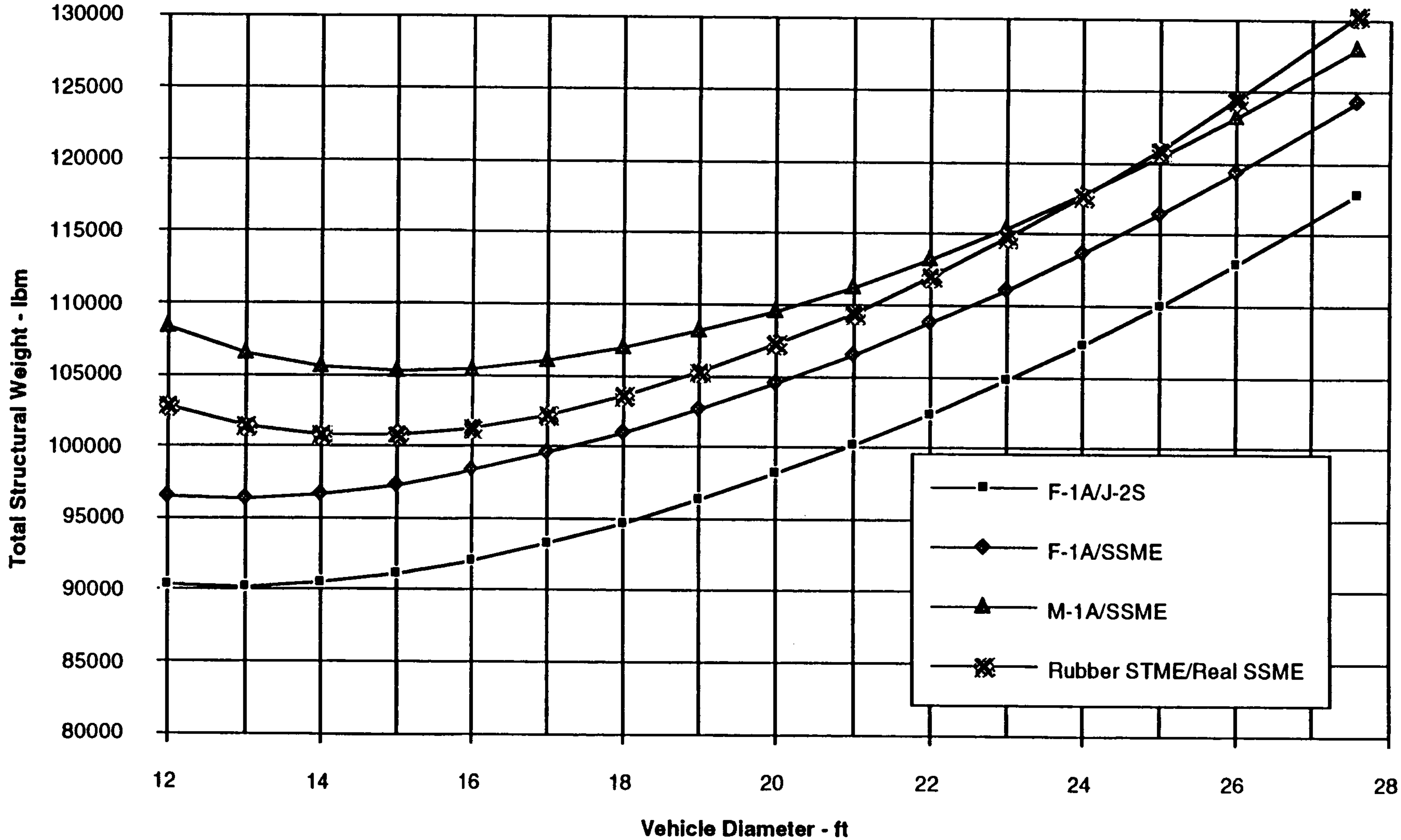
Payload as a Function of Second Stage Initial Acceleration

(F-1A/SSME, No1= 1.565g, No2= 1.423 g)



J. B. McCurry
205-722-4509

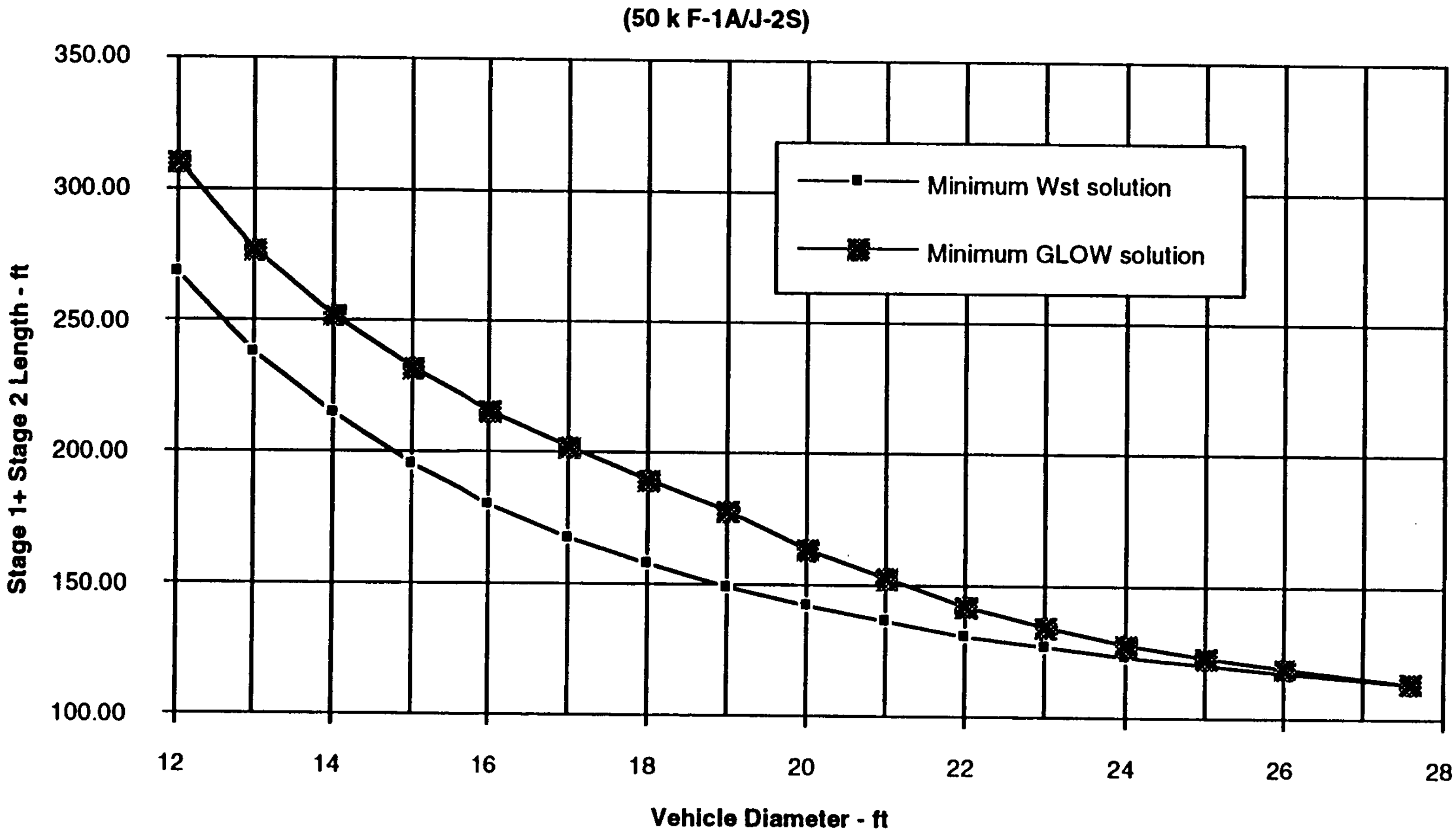
Vehicle Structural Weights



Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

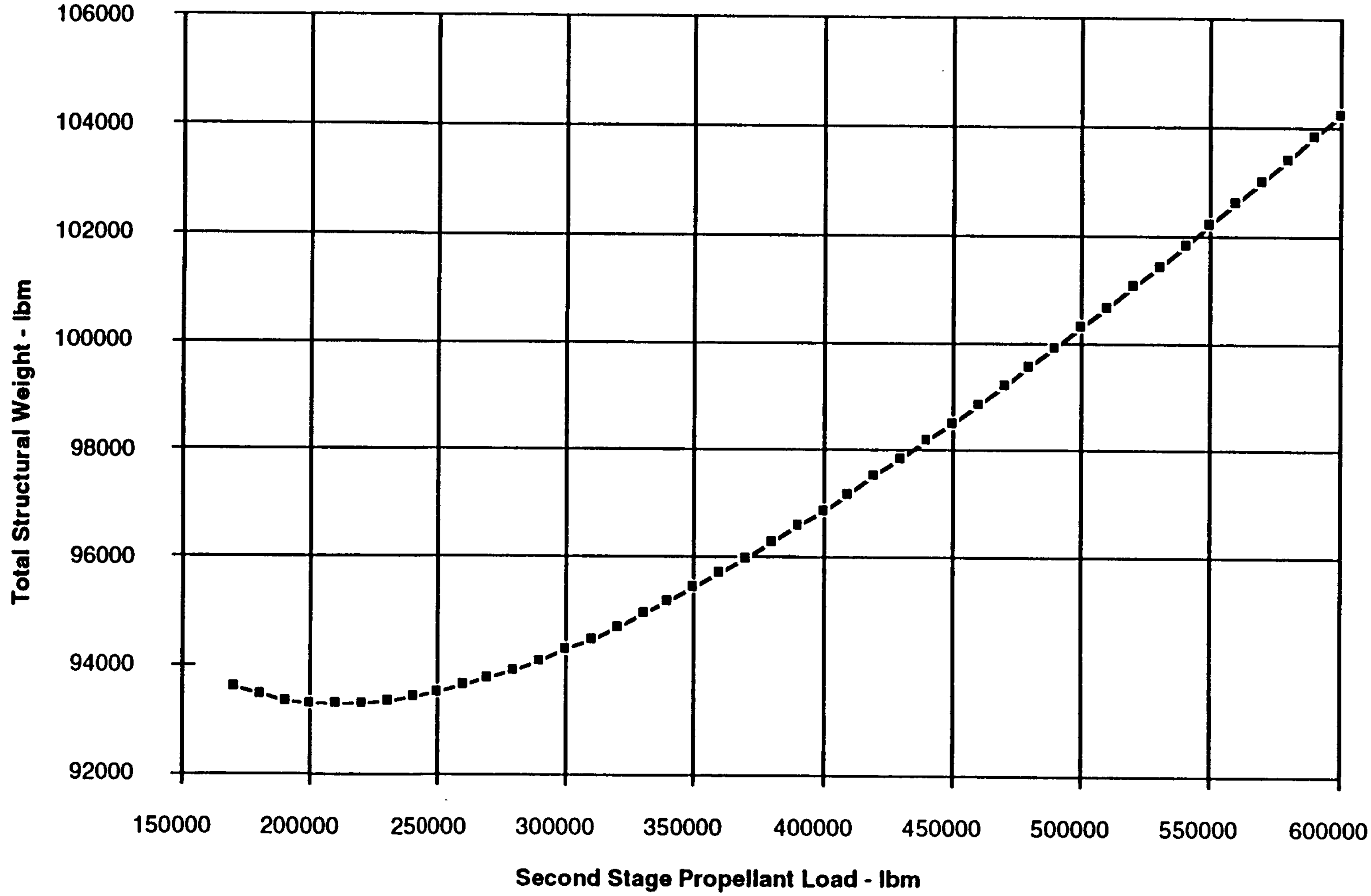
Vehicle Length as a Function of Diameter



J. B. McCurry
205-722-4509

Structural Weight as a Function of Second Stage Propellant Load

(17 foot diameter vehicle with F-1A and J-2S engines)



J. B. McCurry
205-722-4509

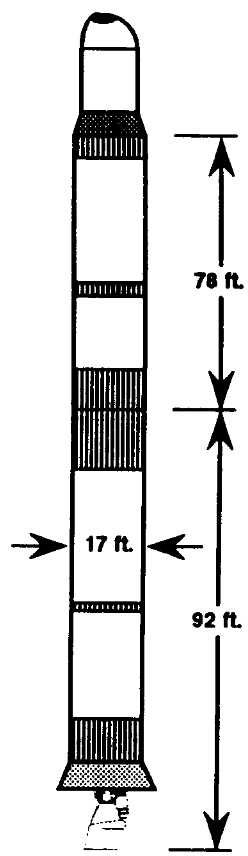
50/80 k lbm Vehicle Options

50 K Configuration	Cycle 1 Sizing	Cycle 1 Trajectory	Cycle 2 Sizing	Cycle 2 Trajectory
F-1A/SSME	X	X		
F-1A/J-2S	X			
F-1A/Vulcain (HM-60)				
M-1A/SSME	X			
M-1A/J-2S				
Rubber STME/SSME	X			
Rubber STME/J-2S				
S. C. Hybrid/SSME	X			
S. C. Hybrid/J-2S	X			
Classic Hybrid/SSME				
Classic Hybrid/J-2S				
RD-170/RD-0120				

80 K Configuration	Cycle 1 Sizing	Cycle 1 Trajectory	Cycle 2 Sizing	Cycle 2 Trajectory
F-1A/SSME	X	X		
F-1A/J-2S				
F-1A/Vulcain (HM-60)				
M-1A/SSME				
M-1A/J-2S				
Rubber STME/SSME				
Rubber STME/J-2S				
S. C. Hybrid/SSME				
S. C. Hybrid/J-2S				
Classic Hybrid/SSME				
Classic Hybrid/J-2S				
RD-170/RD-0120				

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, F-1A/SSME



Length/Diameter= 10.0

Payload:	58,800 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	1,166,286 lbm

First Stage:

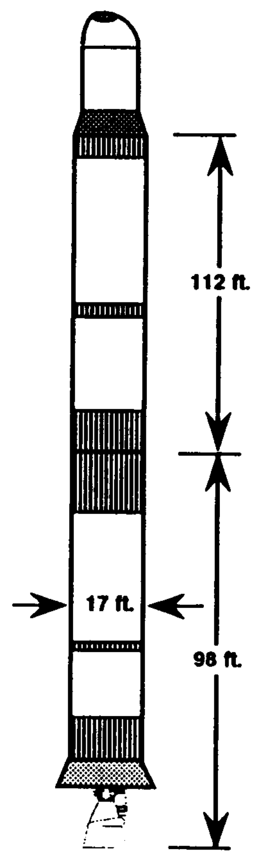
Inert Mass:	68,205 lbm
Usable Propellant:	761,978 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.522 g

Second Stage:

Inert Mass:	31,303 lbm
Usable Propellant:	236,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.423 g

J. B. McCurry
205-722-4509

Preliminary Data
80 k Vehicle, F-1A/SSME



Length/Diameter= 12.4

Payload:	78,500 lbm (35.6 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,437,645 lbm

First Stage:

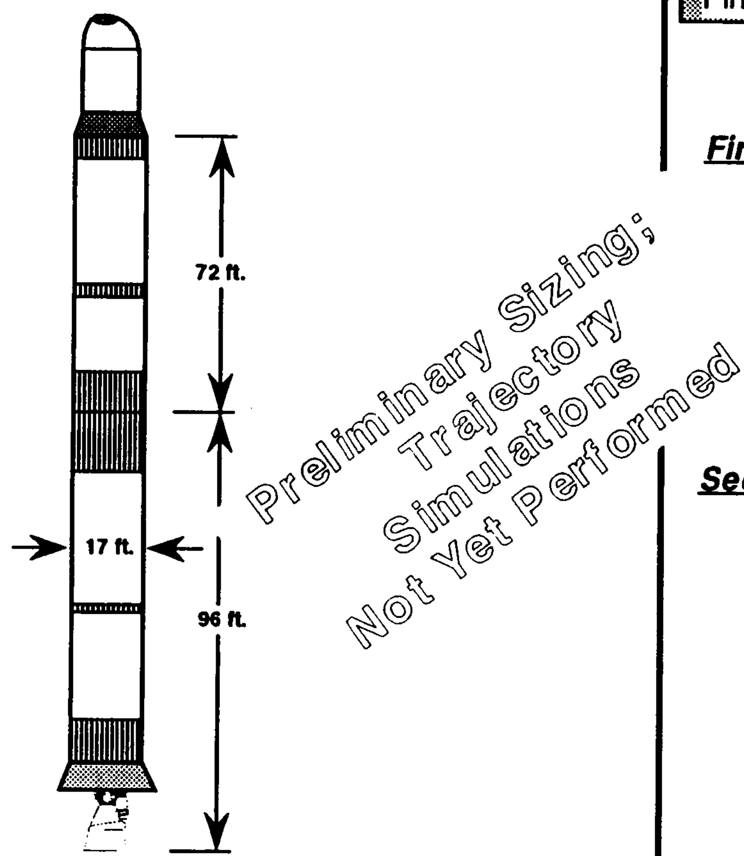
Inert Mass:	69,734 lbm
Usable Propellant:	850,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.252 g

Second Stage:

Inert Mass:	39,411 lbm
Usable Propellant:	400,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.943 g

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, F-1A/ J-2S



Length/Diameter= 9.9

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,172,892 lbm

First Stage:

Inert Mass:	68,458 lbm
Usable Propellant:	819,635 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.535 g

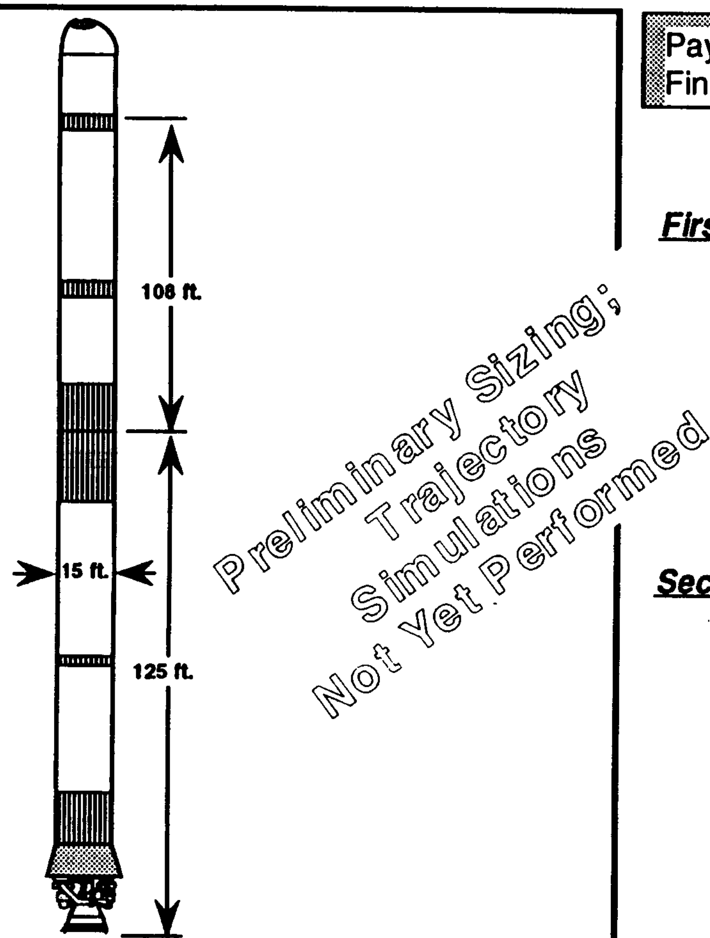
Second Stage:

Inert Mass:	24,799 lbm
Usable Propellant:	210,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.930 g

Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, Rubber STME/SSME



Length/Diameter= 15.5

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 828,442 lbm

First Stage:

Inert Mass:	66,648 lbm
Usable Propellant:	377,669 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	15.0 ft
Thrust/Weight:	1.400 g
Sea Level Thrust:	1,159,819 lbf

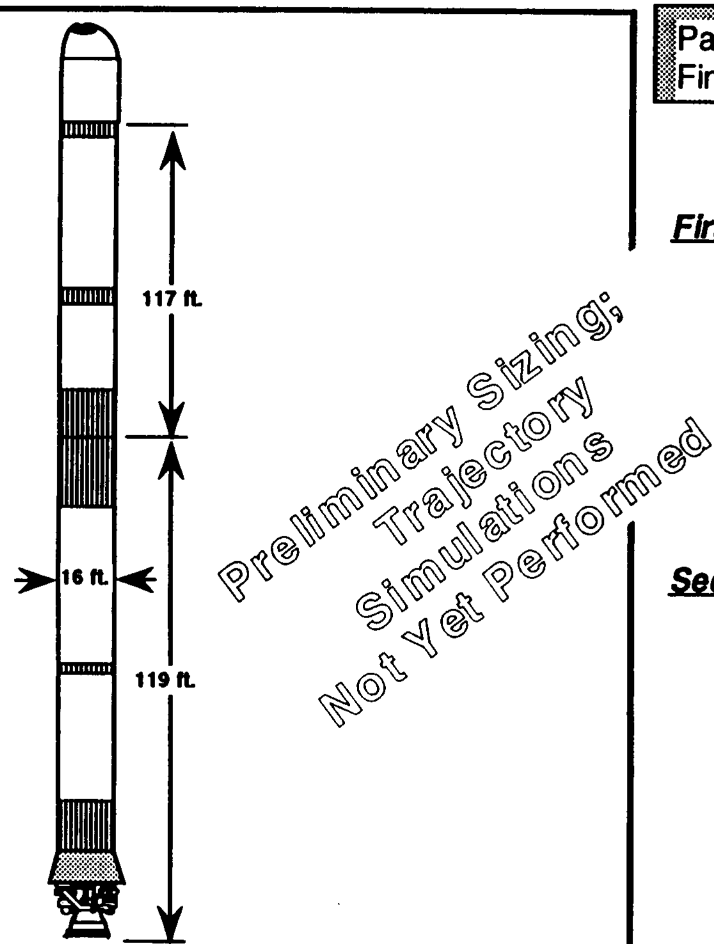
Second Stage:

Inert Mass:	34,126 lbm
Usable Propellant:	300,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	15.0 ft
Thrust/Weight:	1.224 g

Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, M-1A/SSME



Length/Diameter= 14.8

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 854,128 lbm

First Stage:

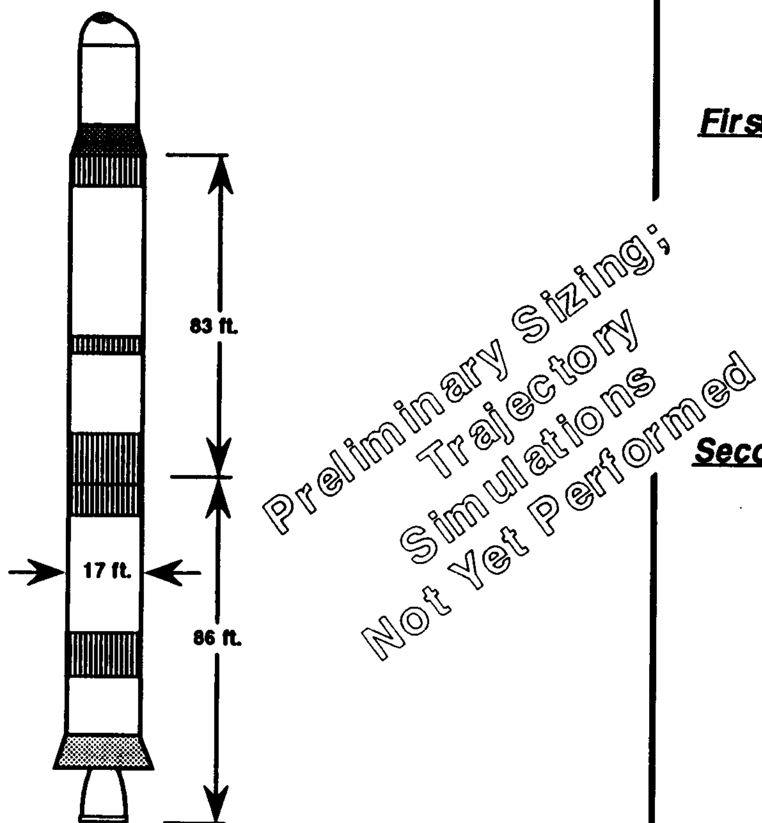
Inert Mass:	67,039 lbm
Usable Propellant:	318,712 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	M-1A/1
Diameter:	16.0 ft
Thrust/Weight:	1.522 g

Second Stage:

Inert Mass:	38,377 lbm
Usable Propellant:	380,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	16.0 ft
Thrust/Weight:	1.003 g

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, Staged Combustion Hybrid/J-2S



Length/Diameter= 9.9

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,222,969 lbm

First Stage:

Inert Mass:	111,561 lbm
Usable Propellant:	774,091 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.472 g
Sea Level Thrust	1,800,000 lbf

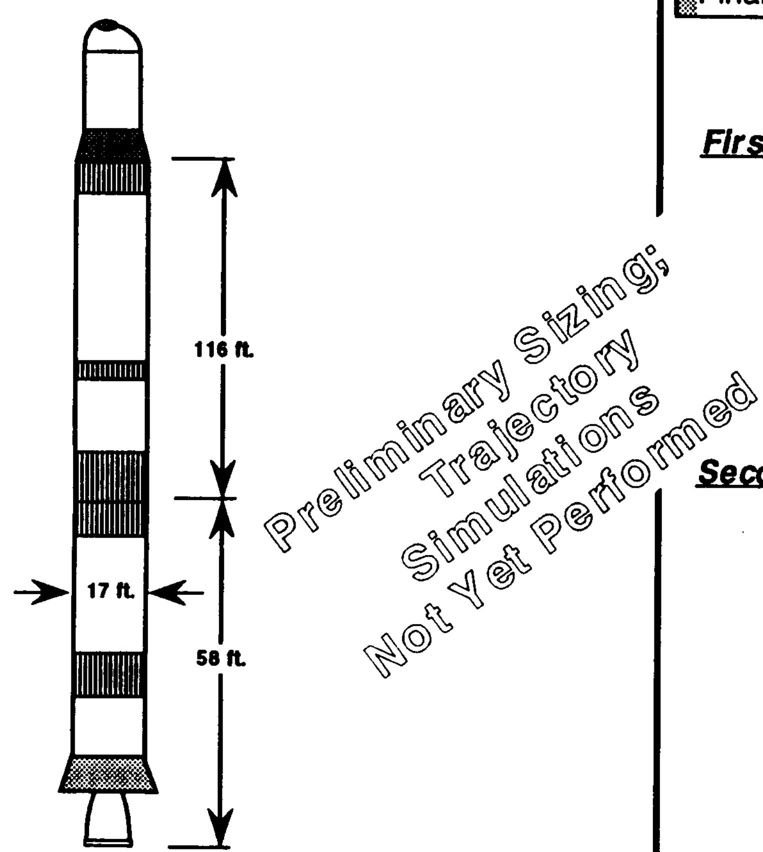
Second Stage:

Inert Mass:	27,317 lbm
Usable Propellant:	240,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.788 g

Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

Preliminary Data
50 k Vehicle, Staged Combustion
Hybrid/SSME



Length/Diameter= 10.2

Payload:	50,000 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 998,456 lbm

First Stage:

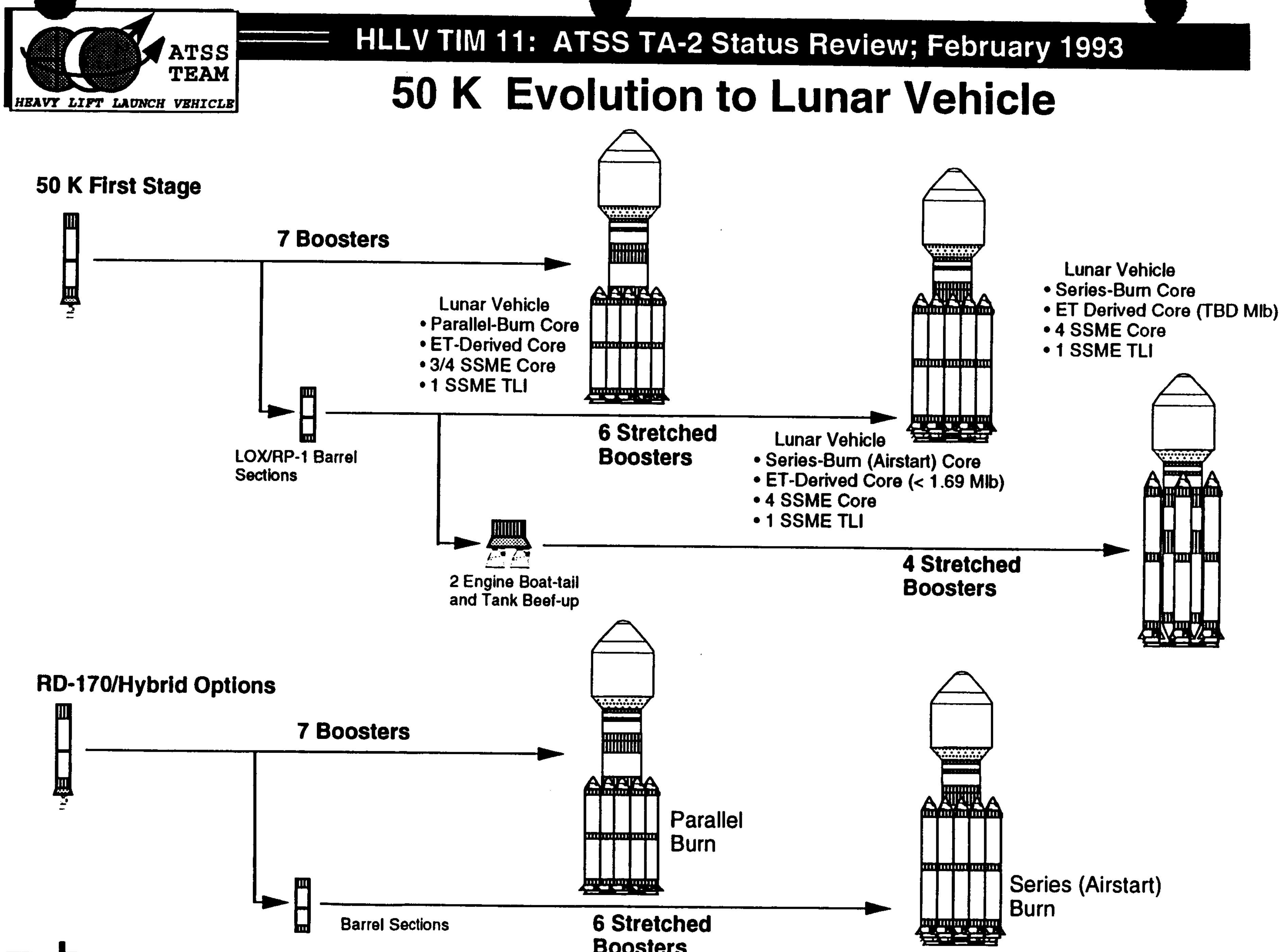
Inert Mass:	71,101 lbm
Usable Propellant:	417,571 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.402 g
Sea Level Thrust	1,400,000 lbf

Second Stage:

Inert Mass:	40,436 lbm
Usable Propellant:	420,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.921 g

J. B. McCurry
205-722-4509

50 K Evolution to Lunar Vehicle



Lockheed, Aerojet, ECON

J. B. McCurry
205-722-4509

Lunar Mission Architectures
Utilizing
Single Stage to Orbit
and
Heavy Lift Launch Vehicle
Candidates

J. B. McCurry
205-722-4509
2/27/93

SSTO/Lunar Guidelines & Assumptions

- At HLLV telecon (2/23) Gene Austin/PT01 solicited ideas for lunar mission architecture scenarios that utilize a Single Stage To Orbit (SSTO) fleet
  -- If 30-35K class (to 220nm/28.5 deg.) SSTO is a given, what other vehicle requirements are necessary to perform multiple-launch lunar mission?

  -- Presume SSTO cargo bay is 15 x 30 ft

  -- Presume on-orbit propellant farm to fuel lunar lander/ascent and/or TLI stage

  -- One going scenario is to use a 100-200K HLLV to augment SSTO fleet to accomplish lunar mission; Early Heavy Lift Launch Vehicle one possible candidate configuration

- Austin feels that RD-701 (tri-propellant) is only engine that enables SSTO concepts
  -- RD-701 can't be efficiently used in linear aerospike config. (?)

  -- Need to confirm that no other engines (i.e., Rocketdyne's latest linear aerospike design) will enable SSTO

J. B. McCurry
205-722-4509

SSTO/Lunar Guidelines & Assumptions (Concluded)

- <u>Questions that ExPO needs to answer</u>:
  - -- How many crewmembers to lunar surface? (i.e., are we backing off of FLO requirements?)
  - -- Is SSF hab. module still used, and if so, launch orientation? (would like to carry it vertically to reduce/eliminate hammerhead shroud; & might have to assemble it on-orbit with lander anyway)
  - -- What are volumetric limitations for lunar cargo & hab. module (to size shroud or determine how much can be carried by SSTO)?
  - -- SSTO presumed to provide propellant to on-orbit farm?
  - -- Would on-orbit prop. farm have ability to handle both LOX/LH2 & LOX/storable? (if not, one vehicle architecture might dictate LOX/LH2 for lunar ascent vehicle; if ascent vehicle placed to LEO empty)
  - -- Presume prop. farm reliquifaction capability for long-term storage?
  - -- Prop. farm at 220 nm, 28.5 deg.? (or is there a "Russian factor"?)
  - -- Assume PLS/CTRV launch vehicle exists before or during SSTO (if so, can use it to assemble HLLV)?

J. B. McCurry
205-722-4509

Architecture Candidates

1. • One SSTO is TLI/lander/ascent vehicle, carries crew & part of lunar cargo
   -- SSTO refueled on-orbit at prop. farm

   • One SSTO is TLI/lander/ascent vehicle,carries remainder of lunar cargo & hab for autonomous lunar landing (must be vertical take-off/vertical landing)
   -- SSTO refueled on-orbit at prop. farm
   -- Can either drop off hab. on lunar surface & return, or eventually bring hab back to Earth surface (will need to have long-term cryo storage or lunar refuel capability)

   • TBD (30-40?) SSTO flights to carry TLI/lander/ascent prop. to prop. farm

2. • One HLLV (~120K) carries lander/hab./cargo to LEO, 2ng stage is TLI stage
   -- Lander & TLI stage fueled on-orbit at prop. farm

   • One HLLV carries lander/ascent & crew to LEO, 2ng stage is TLI stage
   -- Lander/ascent & TLI stages fueled on-orbit at prop. farm

   • TBD (20-24) SSTO flights to carry prop. to prop. farm

J. B. McCurry
205-722-4509

Architecture Candidates (Continued)

3. • One HLLV (~120K) carries lander/hab. & cargo to LEO
   -- Lander/hab. fueled on-orbit at prop. farm

   • One HLLV (~120K) carries TLI stage for unmanned mission & partial TLI prop. load

   • One HLLV (~120K) carries lander/ascent/crew & TLI for manned mission
   -- TLI stage fueled on-orbit at prop. farm

   • TBD (20-24?) SSTO flights to carry remaining prop. to prop. farm
   -- Presumes that SSTO cannot be used as lunar lander (horizontal landing capability only)

4. • One HLLV (~80-100K) carries lander/hab & cargo
   -- Lander/hab. fueled on-orbit at prop. farm

   • One HLLV carries lander/ascent/crew & partial prop. load
   -- Lander/ascent fueled on-orbit at prop. farm

   • One SSTO carries TLI stage (empty) for unmanned mission and one SSTO carries TLI stage (empty) for manned mission
   -- TLI stages fueled on-orbit at prop. farm

   • TBD (24-30?) SSTO flights to carry TLI/lander/ascent prop.

J. B McCurry
205-722-4509

Architecture Candidates (Concluded)

5. • One HLLV (~80-100K) carries lander/hab., empty TLI stage & no cargo
   -- Lander and TLI stage fueled on-orbit at prop. farm

   • One HLLV (~80-100K) carries lander/ascent/crew & empty TLI stage
   -- Lander/ascent and TLI stages fueled on-orbit at prop. farm

   • One SSTO to carry cargo for unmanned mission

   • TBD (24-30?) SSTO flights to carry TLI/lander/ascent prop.

Note:

- Recovery/reuse of TLI stage (for other than SSTO being TLI stage) not addressed at this time

- One of two methods presumed for propellant tanker missions:
  -- SSTO carries separate LOX/LH2 "cargo" tanks that are used exclusively for prop. farm resupply

  -- SSTO prop. tanks are scavenged for residuals on-orbit in conjunction with carrying separate LOX/LH2 cargo tanks (this will reduce number of SSTO flights required but complicates prop. transfer by use of SSTO fill & drains)

HLLV Options

1. Take 50-80K MLLV first stage (LOX/RP?) to use as dual strap-ons with two-stage 50-80K core vehicle
   -- scar core design for strap-on attach hardware

   -- hope that new lunar payloads will not require hammerhead (fly SSF hab. vertically instead of horizontally as with FLO)

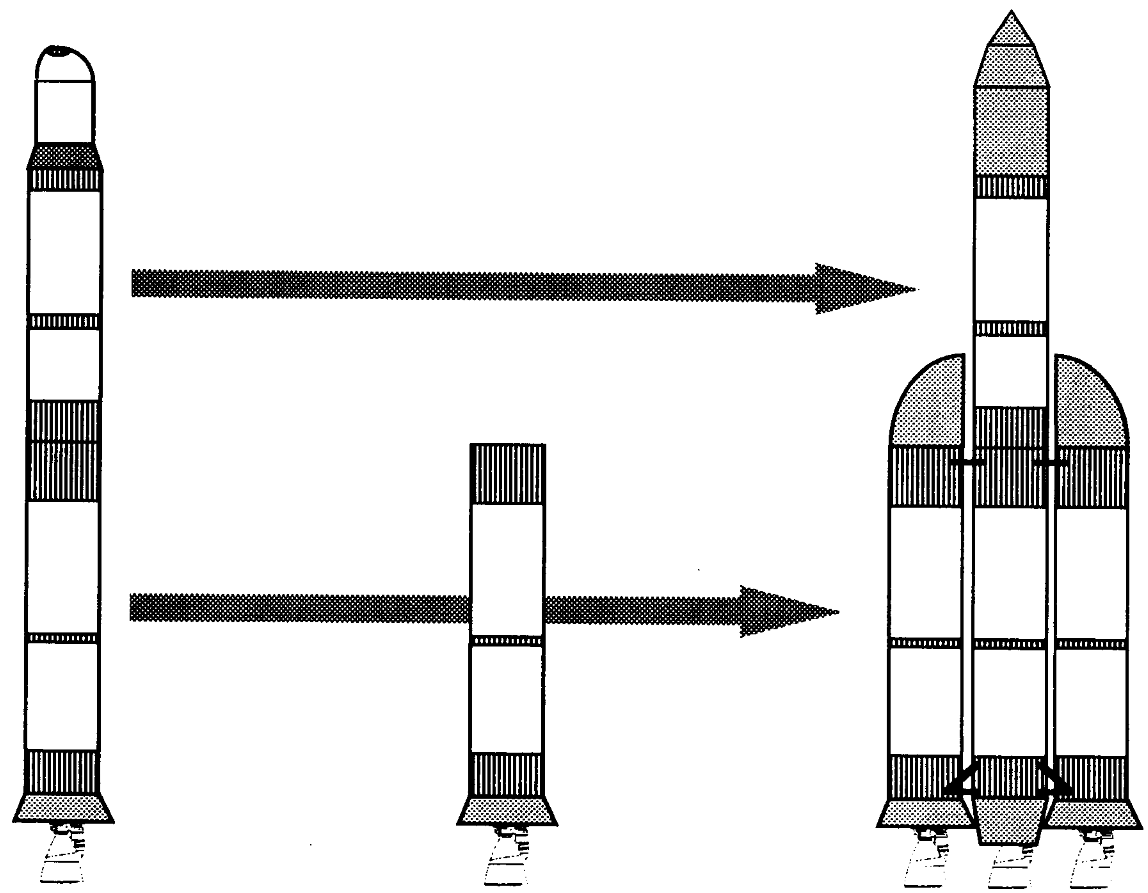
2. Use existing solid strap-ons with two-stage 50-80K core vehicle
   -- scar core design for strap-on attach hardware

3. Use new hybrid strap-ons with two-stage 50-80K core vehicle
   -- scar core design for strap-on attach hardware

4. Revive EHLLV baseline (2 ASRMs, 3-SSME ET-derived core)

J. B. McCurry
205-722-4509

Possible Evolution from 50K to 120(+)K HLLV



J. B. McCurry
205-722-4509

TA-2 Support To Other NASA Projects

March 1993

J. B. McCurry
205-722-4509

TA-2 SUPPORT TO OTHER PROJECTS

TA-2 Charter:

Heavy Lift Launch Vehicle concept definition, sizing, and analysis, including propulsion subsystem definition (foreign & domestic), ground operations & facilities assessment, and life cycle cost estimation, to assist NASA in the identification of future launch vehicle requirements

J. B. McCurry
205-722-4509

TA-2 SUPPORT TO OTHER PROJECTS

Past Project Support (Mar. '92 - Feb. '93)

- Projects Supported:
  - -- First Lunar Outpost
  - -- Griffin "Red Team" and "Super Red Team"

- Organizations Supported:
  - -- MSFC Space Transportation & Exploration Office (PT-01)
  - -- MSFC Preliminary Design Office (PD-01)
  - -- Exploration Programs Office (ExPO)
  - -- KSC Advanced Projects Office (DE-PT)

- Tasks Performed:
  - -- Lunar/Mars mission Heavy Lift Launch Vehicle (HLLV) configuration definition, sizing sensitivity trade studies, & ascent performance assessments
  - -- Stage element propulsion concept identification, sizing, and performance assessment
  - -- Mixed Fleet ground operations and facilities assessments
  - -- Integrated vehicle health management requirements definition (with emphasis on enhancing operability, safety, and reliability)

J. B. McCurry
205-722-4509

TA-2 SUPPORT TO OTHER PROJECTS

Current Project Support (Mar. '93)

- Projects Supported:
  - -- Littles' Existing Technologies Access-to-Space Panel
  - -- Griffin's Advanced Technologies Access-to-Space Panel

- Organizations Supported:
  - -- MSFC Space Transportation & Exploration Office (PT-01)
  - -- MSFC Heavy Lift Launch Vehicle Definition Office (HA-01)
  - -- MSFC Preliminary Design Office (PD-01)
  - -- KSC Advanced Projects Office (DE-PT)

- Tasks Being Performed:
  - -- Launch vehicle sizing & ascent performance assessments for 50K-100K+ payload mass range (Titan IV/PLS/CTRV missions) with assessment of evolution and commonality to lunar mission
  - -- Solid, liquid, and hybrid stage element propulsion concept identification, sizing, and performance assessment
  - -- Preliminary performance, technology requirements, and development cost estimates for the NPO Energomash RD-170 (LOX/kerosene) and RD-701 (LOX/LH_2/kerosene) engines (direct Energomash involvement via Pratt)
  - -- Mixed fleet launch vehicle ground processing and operability assessments
  - -- Integrated vehicle health management requirements definition (with emphasis on enhancing operability, safety, and reliability)

J. B. McCurry
205-722-4509

TA-2 SUPPORT TO OTHER PROJECTS

Future Project Support (Apr.-Dec. '93)

- Anticipated Projects:
  - -- Access-to-Space Panels
  - -- NASA's "Next Launch Vehicle" (supporting SSF re-design options)
  - -- SDIO

- Anticipated Organizations:
  - -- MSFC Space Transportation & Exploration Office (PT-01)
  - -- MSFC Heavy Lift Launch Vehicle Definition Office (HA-01)
  - -- MSFC Preliminary Design Office (PD-01)
  - -- KSC Advanced Projects Office (DE-PT)
  - -- SDIO Program Office

- Possible Tasks To Be Performed:
  - -- Heavy Lift Launch Vehicle (HLLV) definition & assessment
  - -- SSTO/ELV mixed fleet lunar mission assessments
  - -- Detailed performance, technology requirements, and development cost estimates for the NPO Energomash RD-170 and RD-701 engines
  - -- Hybrid launch vehicle options assessments
  - -- Launch vehicle ground processing and operability assessments
  - -- Launch vehicle technology development identification & assessment

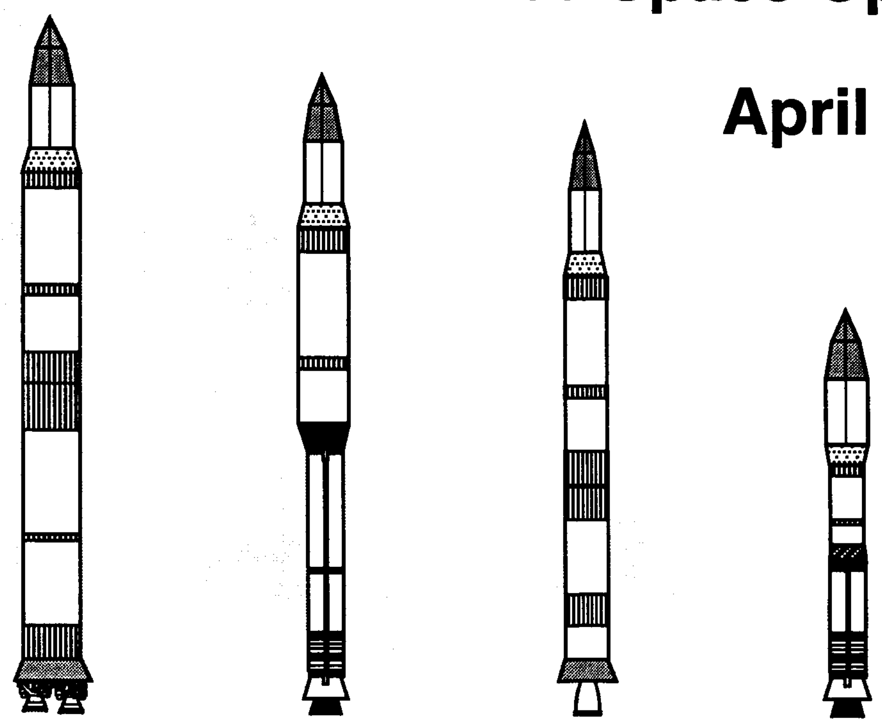
J. B. McCurry
205-722-4509

Alternative 50K Vehicle Concepts–

Operations Evaluation Scores

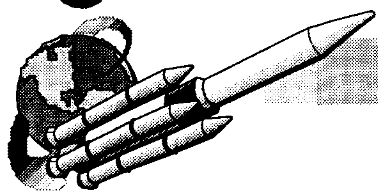
Lockheed Space Operations Company

April 1993



Lockheed

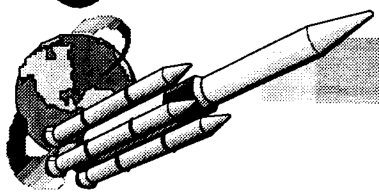
GFL042193 CX FCTR



OPERATIONS ANALYSIS METHODOLOGY

REQUIREMENTS

- Capable of evaluating ground operations for numerous architectures, each comprised of many systems, relative to each other
- Address first order ground operations impacts
- Simple to use for fast turnaround (approx. 1 day/architecture)
- Method accuracy commensurate with limited new system design data
- Prefer implementation on Excel software platform
- Avoid development of detailed launch site design solution for each system/architecture



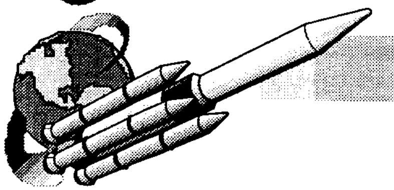
OPERATIONS ANALYSIS METHODOLOGY (CONT)

WHY AVOID DEVELOPMENT OF LAUNCH SITE DESIGN SOLUTION?

- Time/resource consuming
- Questionable value-added for comparative analysis, as opposed to simpler method
- Launch site design solution requires:
  - > Flight element and integrated vehicle (system) processing timelines (accuracy questionable for new systems)
  - > Multiflow system processing timelines for facility "bay" requirements (depends on maximum flight rate in manifest)
  - > Controversial decisions on mixed fleet facility sharing
  - > Multiflow mixed fleet processing timelines to assure adequate facility capacities

OPERATIONS ANALYSIS METHODOLOGY (CONT)

- Attribute or Figure of Merit (FOM_{ARC}) computed for Ground Operations
- FOM_{ARC} to be used as a top-level attribute, similar to Funding Profile, Environment, Probability of Mission Success, etc.
- FOM_{ARC} will be the weighted sum of normalized utility values
- The utility values will be a function of ground operations subattributes or Measures of Effectiveness (MOEs)
- The MOEs (subattributes) are defined as a series of complexity factors (CF_i)



OPERATIONS ANALYSIS METHODOLOGY (CONT)

COMPLEXITY FACTOR	DESCRIPTION
NUMBER OF ELEMENTS	TOTAL NUMBER OF SIGNIFICANT FLIGHT ELEMENTS IN EACH SYSTEM (CORE STAGE, UPPER STAGE, BOOSTERS, PAYLOAD CARRIER OR ORBITER, ETC)
MANNED/UNMANNED RATING	MANNED VERSUS UNMANNED SYSTEM CONFIGURATION
PROCESSING CONCEPT	INTEGRATE ON PAD (IOP) vs INTEGRATE TRANSFER LAUNCH (ITL) vs MIXED ITL/IOP
NUMBER OF FLUIDS	TOTAL NUMBER OF PROPELLANTS, GASES AND OTHER FLUIDS IN LAUNCH SYSTEM
RELIABILITY	PLANNED OR DEMONSTRATED RELIABILITY OF LAUNCH SYSTEM
EXPENDABLE/RECOVERABLE	RECOVERABLE VERSUS EXPENDABLE FLIGHT ELEMENT
PROPELLANT TYPE	PROPELLANT COMBINATION USED BY FLIGHT ELEMENT, IF ANY
NUMBER OF SIGNIFICANT COMPONENTS	NUMBER OF ENGINES, OMS/RCS PODS, SOLID SEGMENTS, OR OTHER SIGNIFICANT COMPONENTS IN FLIGHT ELEMENT

OPERATIONS ANALYSIS METHODOLOGY (CONT)

COMPLEXITY FACTOR	LEVEL OF ASSESSMENT		
	SYSTEM	ELEMENT	WEIGHTING (%)
NUMBER OF ELEMENTS	✓		15.6
MANNED/UNMANNED RATING	✓		16.7
PROCESSING CONCEPT	✓		8.9
NUMBER OF FLUIDS	✓		13.3
RELIABILITY	✓		5.6
EXPENDABLE/RECOVERABLE		✓	12.3
PROPELLANT TYPE		✓	10.0
NUMBER OF SIGNIFICANT COMPONENTS		✓	17.7

VEHICLE CONFIGURATION MATRIX

First Stage/Second Stage Options		
Liquid/Liquid *	**Hybrid/Liquid ***	**Solid/Liquid**
F-1A/LCSSME	Staged Combustion Hybrid/LCSSME	3 Segment ASRM/LCSSME
F-1A/J-2S	Staged Combustion Hybrid/Rubber STME	3 Segment ASRM/LCSSME
F-1A/SSME	Staged Combustion Hybrid/J-2S	
F-1A/RD-0120	Staged Combustion Hybrid/Vulcain	2 Segment ASRM/J-2S
F-1A/Vulcain	Staged Combustion Hybrid/RD-0120	2 Segment ASRM/LCSSME
		2 Segment ASRM/SSME
STME/LCSSME	Classical Hybrid/LCSSME	
STME/STME	Classical Hybrid/Rubber STME	1 Segment ASRM/Centaur
STME/RD-0120	Classical Hybrid/J-2S	
STME/Vulcain	Classical Hybrid/Vulcain	
	Classical Hybrid/RD-0120	
M-1A/LCSSME		
M-1A/RD-0120		
M-1A/Vulcain		
RD-170/LCSSME		
RD-170/J-2S		
RD-170/RD-0120		
RD-170/Vulcain		
LCSSME/LCSSME		
LCSSME/RD-0120		
LCSSME/Vulcain		

Note: * Configurations sized for 50 Klbm payload

OPERATIONS EVALUATION CONCLUSIONS

- Hybrid/Liquid configurations scored highest
- M-1A Liquid/Liquid family scored higher than other Liquid/Liquid configurations due to single 1st stage engine and propellant commonality
- Large ASRM/Liquid scored lowest due to number of components & propellants
- Narrow score bandwidth due to common 2nd stage and payload carrier design
- Engine selection should be final operations discriminator for configuration downselect

COMPLEXITY FACTOR INPUTS

3 Segment ASRM - All

Assumptions:

N2H4 ACS on 2nd stg.

FLUIDS
LO2 LH2 N2H4 HYD GHe

# ELEMENTS	3
MANNED?	M
PROCESSING CONCEPT	ITL
# FLUIDS	5
R	.99

	ELEMENTS				
	ASRM (1)	2nd STAGE	P/L CAR		
COMPONENTS	3 Segments 1 Aft Skirt	1 Tank Set 1 Engine 1 ACS	1 P/L 1 Shroud		
EXPEND?	Recover	Expend	Expend		
PROP TYPE	Solid / Storable	≥ 3	None		
# SIGNIFICANT COMPONENTS	4	3	2		

COMPLEXITY FACTOR INPUTS

2 Segment ASRM - All

Assumptions:

N2H4 ACS on 2nd stg.

FLUIDS
LO2 LH2 N2H4 HYD GHe

# ELEMENTS	3
MANNED?	M
PROCESSING CONCEPT	ITL
# FLUIDS	5
R	.99

	ELEMENTS				
	ASRM (1)	2nd STAGE	P/L CAR		
COMPONENTS	2 Segments 1 Aft Skirt	1 Tank Set 1 Engine 1 ACS	1 P/L 1 Shroud		
EXPEND?	Recover	Expend	Expend		
PROP TYPE	Solid / Storable	≥ 3	None		
# SIGNIFICANT COMPONENTS	3	3	2		

COMPLEXITY FACTOR INPUTS

1 Segment ASRM / Centaur

Assumptions:

N2H4 ACS on 2nd stg.

FLUIDS
LO2 LH2 N2H4 HYD GHe

# ELEMENTS	3
MANNED?	M
PROCESSING CONCEPT	ITL
# FLUIDS	5
R	.99

	ELEMENTS				
	ASRM (1)	2nd STAGE	P/L CAR		
COMPONENTS	1 Segment 1 Aft Skirt	1 Tank Set 2 Engines 1 ACS	1 P/L 1 Shroud		
EXPEND?	Recover	Expend	Expend		
PROP TYPE	Solid / Storable	≥ 3	None		
# SIGNIFICANT COMPONENTS	2	4	2		

COMPLEXITY FACTOR INPUTS

Staged Combustion Hybrid / Classical Hybrid - All

Assumptions:

EMA TVC
N2H4 ACS on 2nd stg.

FLUIDS
LO2
LH2
N2H4
GHe

# ELEMENTS	3
MANNED?	M
PROCESSING CONCEPT	ITL
# FLUIDS	4
R	.985

	ELEMENTS				
	1st STAGE	2nd STAGE	P/L CAR		
COMPONENTS	1 Tank Set 1 Segment 1 Engine	1 Tank Set 1 Engine 1 ACS	1 P/L 1 Shroud		
EXPEND?	Expend	Expend	Expend		
PROP TYPE	Solid / Cryo	≥ 3	None		
# SIGNIFICANT COMPONENTS	3	3	2		

COMPLEXITY FACTOR INPUTS

STME - All
LC SSME - All

Assumptions:

EMA TVC
N2H4 ACS on 2nd stg.

FLUIDS
LO2 LH2 GHe N2H4

# ELEMENTS	3
MANNED?	M
PROCESSING CONCEPT	ITL
# FLUIDS	4
R	.98

	ELEMENTS				
	1st STAGE	2nd STAGE	P/L CAR		
COMPONENTS	1 Tank Set 2 Engines	1 Tank Set 1 Engine 1 ACS	1 P/L 1 Shroud		
EXPEND?	Expend	Expend	Expend		
PROP TYPE	Cryo / Cryo	≥ 3	None		
# SIGNIFICANT COMPONENTS	3	3	2		

COMPLEXITY FACTOR INPUTS

M - 1A - All

Assumptions:

EMA TVC
N2H4 ACS on 2nd stg.

FLUIDS
LO2 LH2 GHe N2H4

# ELEMENTS	3
MANNED?	M
PROCESSING CONCEPT	ITL
# FLUIDS	4
R	.98

	ELEMENTS				
	1st STAGE	2nd STAGE	P/L CAR		
COMPONENTS	1 Tank Set 1 Engine	1 Tank Set 1 Engine 1 ACS	1 P/L 1 Shroud		
EXPEND?	Expend	Expend	Expend		
PROP TYPE	Cryo / Cryo	$\geq$ 3	None		
# SIGNIFICANT COMPONENTS	2	3	2		

COMPLEXITY FACTOR INPUTS

F1A - All
RD - 170 - All

Assumptions:

EMA TVC
N2H4 ACS on 2nd stg.

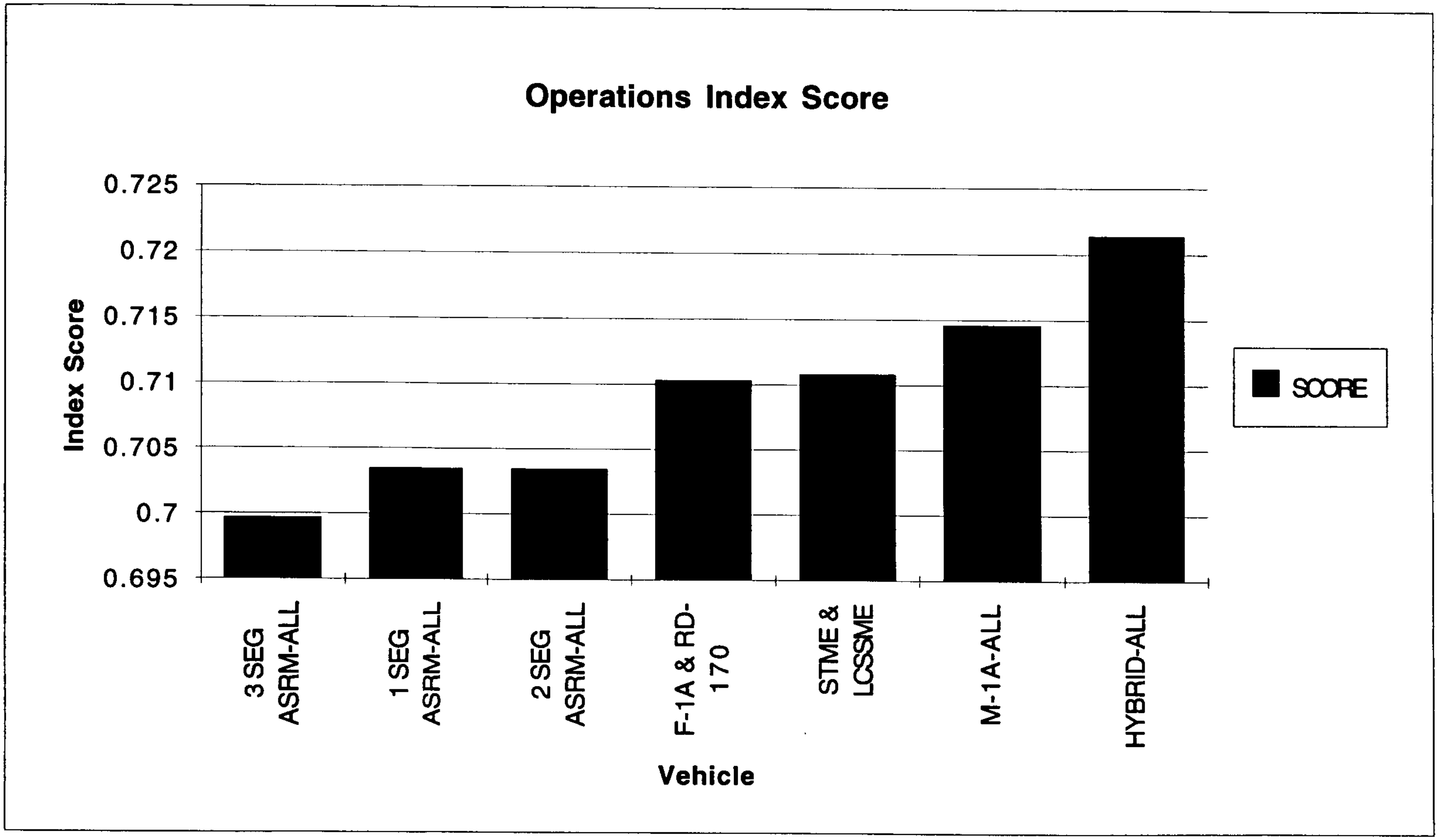
FLUIDS
LO2
LH2
RP -1
GHe
N2H4

# ELEMENTS	3
MANNED?	M
PROCESSING CONCEPT	ITL
# FLUIDS	5
R	.98

	ELEMENTS				
	1st STAGE	2nd STAGE	P/L CAR		
COMPONENTS	1 Tank Set 1 Engine	1 Tank Set 1 Engine 1 ACS	1 P/L 1 Shroud		
EXPEND?	Expend	Expend	Expend		
PROP TYPE	Cryo / Storable	≥ 3	None		
# SIGNIFICANT COMPONENTS	2	3	2		

Lockheed

COMPLEXITY FACTOR INPUTS



Note:
A higher score indicates better operability

NAME	SCORE
3 SEG ASRM-ALL	0.6997
1 SEG ASRM-ALL	0.7035
2 SEG ASRM-ALL	0.7035
F-1A & RD-170	0.7103
STME & LCSSME	0.7108
M-1A-ALL	0.7146
HYBRID-ALL	0.7214

ARCHITECTURE DATA				SYSTEM DATA											ELEMENT DATA								
Architecture Number	# of Flights	Flights Value	Sys. Com.	System Number / # Flights / % Flights	# of Elements	Elements Value	Manned or Unmanned	... or Unmanned ed ...	Proc.	Type Proc. Value	# Fluids	Fluids Value	R	R Value	Element Number	Expend ?	Expend Weighted Value	Prop. Type	Prop Type Value	... Type Weighted ...	# Sig. Com.	Sig Com Value	... Com Weighted ...
			(Enter as a Decimal)	(Should add up to total number of flights and also add up to 100%)			Unmanned = 1 HiValue = .5 Manned = .1		ITL = 1 Mixed = .5 IOP = .1							Expend = 1 Refurb = .5		None = 0 Solid = 1 Hyper Mono = 2 Solid/Storable = 3 Hyper BiProp = 4 Solid Cryo =					
N/A	1	1.0000	1.00	F1&RD170	3	0.8714	0.10	0.1000	1.00	1.0000	5	0.7000	0.9800	0.7800	1	1.00		6	0.3250		2	0.9357	
				1											2	1.00		8	0.1000		3	0.8714	
				100.00%											3	1.00	1.0000	0	1.0000	0.4750	2	0.9357	0.9143
				1																			
				100%																			
FOM INPUT DATA		1.0000	1.0000			0.8714		0.10		1.0000		0.7000		0.7800			1.0000			0.4750			0.9143
WEIGHTING FACT[illegible]		0.0	0.0			15.6		16.7		8.9		13.3		5.6			12.3			10.0			17.7
PRODUCT		0.0000	0.0000			0.1358		0.0167		0.0892		0.0932		0.0436			0.1225			0.0475			0.1619

FIGURE OF MERIT FOR THIS SYSTEM

0.7103

ARCHITECTURE DATA				SYSTEM DATA											ELEMENT DATA								
Architecture Number	# of Flights	Flights Value	Sys. Com.	System Number # Flights % Flights	# of Elements	Elements Value	Manned or Unmanned	or Unmanned ...	Proc.	Type Proc. Value	# Fluids	Fluids Value	R	R Value	Element Number	Expend ?	Expend Weighted Value	Prop. Type	Prop Type Value	Type Weighted ...	# Sig. Com.	Sig Com Value	Com Weighted ...
			(Enter as a Decimal)	(Should add up to total number of flights and also add up to 100%)			Unmanned = 1 HiValue = .5 Manned = .1		ITL = 1 Mixed = .5 IOP = .1							Expend = 1 Refurb = .5		None = 0 Solid = 1 Hyper Mono = 2 Solid/Storable = 3 Hyper BiProp = 4 Solid Cryo =					
N/A	1	1.0000	1.00	Hybrid-All	3	0.8714	0.10	0.1000	1.00	1.0000	4	0.7600	0.9850	0.8350	1	1.00		5	0.4375		3	0.8714	
				1											2	1.00		8	0.1000		3	0.8714	
				100.00%											3	1.00	1.0000	0	1.0000	0.5125	2	0.9357	0.8929
				1																			
				100%																			
FOM INPUT DATA		1.0000	1.0000			0.8714		0.10		1.0000		0.7600		0.8350			1.0000			0.5125			0.8929
WEIGHTING FACTO		0.0	0.0			15.6		16.7		8.9		13.3		5.6			12.3			10.0			17.7
PRODUCT		0.0000	0.0000			0.1358		0.0167		0.0892		0.1012		0.0467			0.1225			0.0512			0.1581

FIGURE OF MERIT FOR THIS SYSTEM

0.7214

ARCHITECTURE DATA				SYSTEM DATA											ELEMENT DATA								
Architecture Number	# of Flights	Flights Value	Sys. Com.	System Number / # Flights / % Flights	# of Elements	Elements Value	Manned or Unmanned	... or Unmanned ...	Proc.	Type Proc. Value	# Fluids	Fluids Value	R	R Value	Element Number	Expend ?	Expend Weighted Value	Prop. Type	Prop Type Value	... Type Weighted ...	# Sig. Com	Sig Com Value	... Com Weighted ...
			(Enter as a Decimal)	(Should add up to total number of flights and also add up to 100%)			Unmanned = 1 HiValue = .5 Manned = .1		ITL = 1 Mixed = .5 IOP = .1							Expend = 1 Refurb = .5		None = 0 Solid = 1 Hyper Mono = 2 Solid/Storable = 3 Hyper BiProp = 4 Solid Cryo =					
N/A	1	1.0000	1.00	M-1A-All	3	0.8714	0.10	0.1000	1.00	1.0000	4	0.7600	0.9800	0.7800	1	1.00		7	0.2125		2	0.9357	
				1											2	1.00		8	0.1000		3	0.8714	
				100.00%											3	1.00	1.0000	0	1.0000	0.4375	2	0.9357	0.9143
				1																			
				100%																			
FOM INPUT DATA		1.0000	1.0000			0.8714		0.10		1.0000		0.7600	.	0.7800			1.0000			0.4375			0.9143
WEIGHTING FACTO		0.0	0.0			15.6		16.7		8.9		13.3		5.6			12.3			10.0			17.7
PRODUCT		0.0000	0.0000			0.1358		0.0167		0.0892		0.1012		0.0436			0.1225			0.0437			0.1619

FIGURE OF MERIT FOR THIS SYSTEM

0.7146

ARCHITECTURE DATA				SYSTEM DATA											ELEMENT DATA								
Architecture Number	# of Flights	Flights Value	Sys. Com.	System Number # Flights % Flights	# of Elements	Elements Value	Manned or Unmanned	or Unmanned ...	Proc.	Type Proc. Value	# Fluids	Fluids Value	R	R Value	Element Number	Expend ?	Expend Weighted Value	Prop. Type	Prop Type Value	Type Weighted ...	# Sig. Com.	Sig Com Value	Com Weighted ...
			(Enter as a Decimal)	(Should add up to total number of flights and also add up to 100%)			Unmanned = 1 HiValue = .5 Manned = .1		ITL = 1 Mixed = .5 IOP = .1							Expend = 1 Refurb = .5		None = 0 Solid = 1 Hyper Mono = 2 Solid/Storable = 3 Hyper BiProp = 4 Solid Cryo =					
N/A	1	1.0000	1.00	STME&LC	3	0.8714	0.10	0.1000	1.00	1.0000	4	0.7600	0.9800	0.7800	1	1.00		7	0.2125		3	0.8714	
				1											2	1.00		8	0.1000		3	0.8714	
				100.00%											3	1.00	1.0000	0	1.0000	0.4375	2	0.9357	0.8929
				1																			
				100%																			
FOM INPUT DATA		1.0000	1.0000			0.8714		0.10		1.0000		0.7600		0.7800			1.0000			0.4375			0.8929
WEIGHTING FACT(		0.0	0.0			15.6		16.7		8.9		13.3		5.6			12.3			10.0			17.7
PRODUCT		0.0000	0.0000			0.1358		0.0167		0.0892		0.1012		0.0436			0.1225			0.0437			0.1581

FIGURE OF MERIT FOR THIS SYSTEM

0.7108

ARCHITECTURE DATA			SYSTEM DATA												ELEMENT DATA								
Architecture Number	# of Flights	Flights Value	Sys. Com.	System Number / # Flights / % Flights	# of Elements	Elements Value	Manned or Unmanned	or Unmanned ...	Proc.	Type Proc. Value	# Fluids	Fluids Value	R	R Value	Element Number	Expend ?	Expend Weighted Value	Prop. Type	Prop Type Value	Type Weighted ...	# Sig. Com.	Sig Com Value	Com Weighted ...
			(Enter as a Decimal)	(Should add up to total number of flights and also add up to 100%)			Unmanned = 1 HiValue = .5 Manned = .1		ITL = 1 Mixed = .5 IOP = .1							Expend = 1 Refurb = .5		None = 0 Solid = 1 Hyper Mono = 2 Solid/Storable = 3 Hyper BiProp = 4 Solid Cryo =					
N/A	1	1.0000	1.00	1 ASRM	3	0.8714	0.10	0.1000	1.00	1.0000	5	0.7000	0.9900	0.8900	1	0.50		3	0.6625		2	0.9357	
				1											2	1.00		8	0.1000		4	0.8071	
				100.00%											3	1.00	0.8333	0	1.0000	0.5875	2	0.9357	0.8929
				1																			
				100%																			
FOM INPUT DATA		1.0000	1.0000			0.8714		0.10		1.0000		0.7000		0.8900			0.8333			0.5875			0.8929
WEIGHTING FACTO		0.0	0.0			15.6		16.7		8.9		13.3		5.6			12.3			10.0			17.7
PRODUCT		0.0000	0.0000			0.1358		0.0167		0.0892		0.0932		0.0498			0.1021			0.0587			0.1581

FIGURE OF MERIT FOR THIS SYSTEM

0.7035

ARCHITECTURE DATA			SYSTEM DATA											
Architecture Number	# of Flights	Flights Value	Sys. Com.	System Number / # Flights / % Flights	# of Elements	Elements Value	Manned or Unmanned	or Unmanned	Proc.	Type Proc. Value	# Fluids	Fluids Value	R	R Value
			(Enter as a Decimal)	(Should add up to total number of flights and also add up to 100%)			Unmanned = 1 HiValue = .5 Manned = .1		ITL = 1 Mixed = .5 IOP = .1					
N/A	1	1.0000	1.00	2 ASRM	3	0.8714	0.10	0.1000	1.00	1.0000	5	0.7000	0.9900	0.8900
				1										
				100.00%										
				1										
				100%										
FOM INPUT DATA		1.0000	1.0000			0.8714		0.10		1.0000		0.7000		0.8900
WEIGHTING FACT		0.0	0.0			15.6		16.7		8.9		13.3		5.6
PRODUCT		0.0000	0.0000			0.1358		0.0167		0.0892		0.0932		0.0498

ELEMENT DATA								
Element Number	Expend ?	Expend Weighted Value	Prop. Type	Prop Type Value	Type Weighted	# Sig. Com.	Sig Com Value	Com Weighted
	Expend = 1 Refurb = .5		None = 0 Solid = 1 Hyper Mono = 2 Solid/Storable = 3 Hyper BiProp = 4 Solid Cryo =					
1	0.50		3	0.6625		3	0.8714	
2	1.00		8	0.1000		3	0.8714	
3	1.00	0.8333	0	1.0000	0.5875	2	0.9357	0.8929
		0.8333			0.5875			0.8929
		12.3			10.0			17.7
		0.1021			0.0587			0.1581

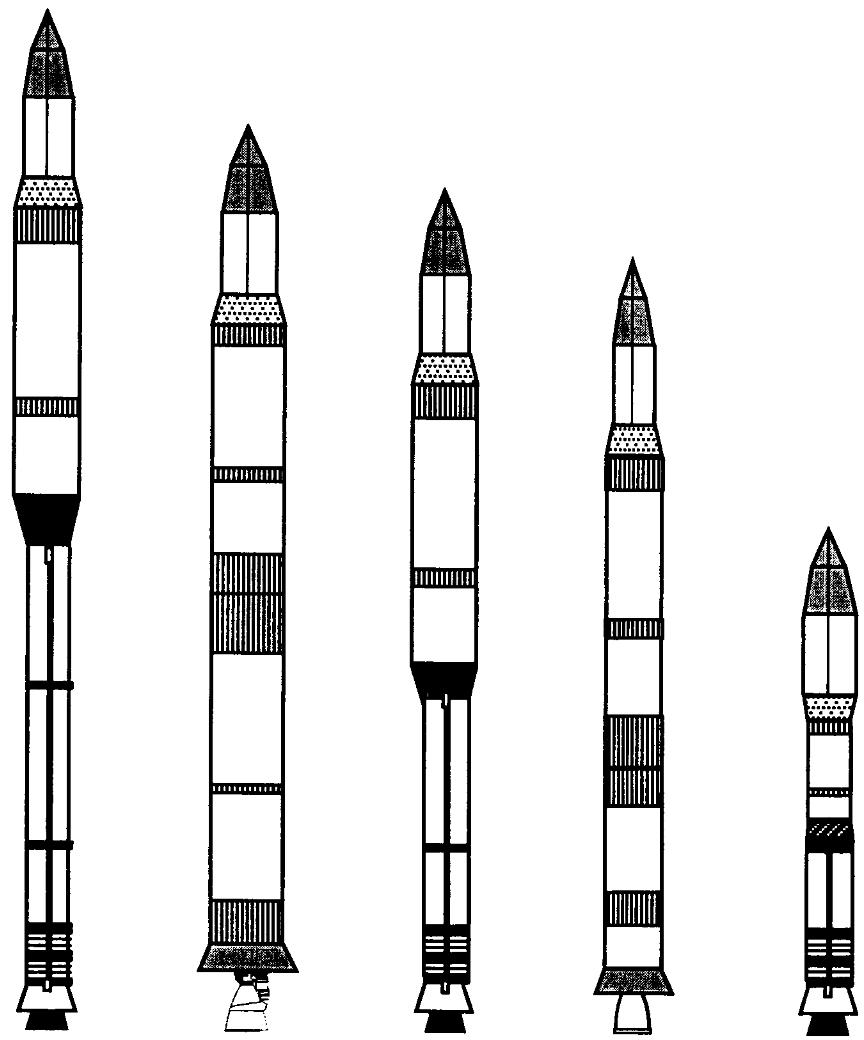
FIGURE OF MERIT FOR THIS SYSTEM

0.7035

ARCHITECTURE DATA				SYSTEM DATA											ELEMENT DATA								
Architecture Number	# of Flights	Flights Value	Sys. Com.	System Number / # Flights / % Flights	# of Elements	Elements Value	Manned or Unmanned	or Unmanned	Proc.	Type Proc. Value	# Fluids	Fluids Value	R	R Value	Element Number	Expend ?	Expend Weighted Value	Prop. Type	Prop Type Value	Type Weighted	# Sig. Com.	Sig Com Value	Com Weighted
			(Enter as a Decimal)	(Should add up to total number of flights and also add up to 100%)			Unmanned = 1 HiValue = .5 Manned = .1		ITL = 1 Mixed = .5 IOP = .1							Expend = 1 Refurb = .5		None = 0 Solid = 1 Hyper Mono = 2 Solid/Storable = 3 Hyper BiProp = 4 Solid Cryo =					
N/A	1	1.0000	1.00	3 ASRM	3	0.8714	0.10	0.1000	1.00	1.0000	5	0.7000	0.9900	0.8900	1	0.50		3	0.6625		4	0.8071	
				1											2	1.00		8	0.1000		3	0.8714	
				100.00%											3	1.00	0.8333	0	1.0000	0.5875	2	0.9357	0.8714
				1																			
				100%																			
FOM INPUT DATA		1.0000	1.0000			0.8714		0.10		1.0000		0.7000		0.8900			0.8333			0.5875			0.8714
WEIGHTING FACT(		0.0	0.0			15.6		16.7		8.9		13.3		5.6			12.3			10.0			17.7
PRODUCT		0.0000	0.0000			0.1358		0.0167		0.0892		0.0932		0.0498			0.1021			0.0587			0.1543

FIGURE OF MERIT FOR THIS SYSTEM

0.6997



Alternative Launch Vehicle Concepts for NASA's *Access to Space* Studies

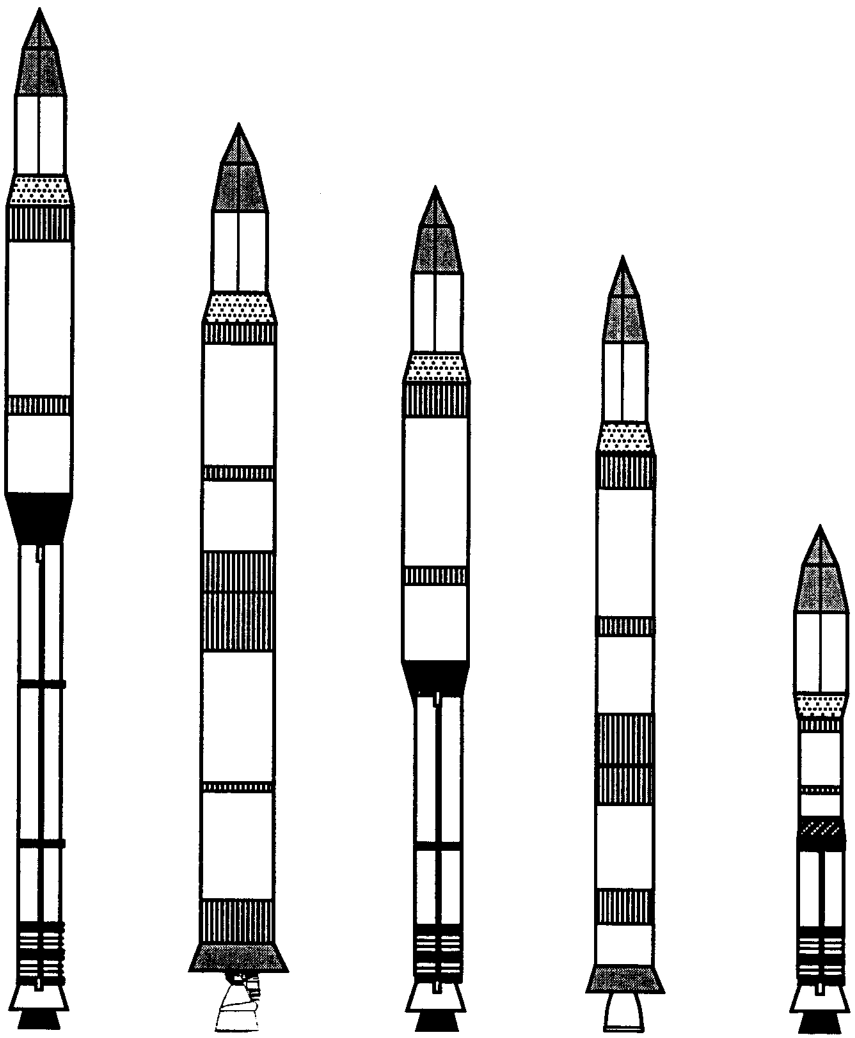
April 1993

K. A. Holden
205-722-4531
Kevin D. Sagis
205-722-4532

Lockheed

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Advanced Launch Vehicle Analysis



Topics

1.0 Vehicle Configuration Summaries

2.0 Hybrid Booster Groundrules

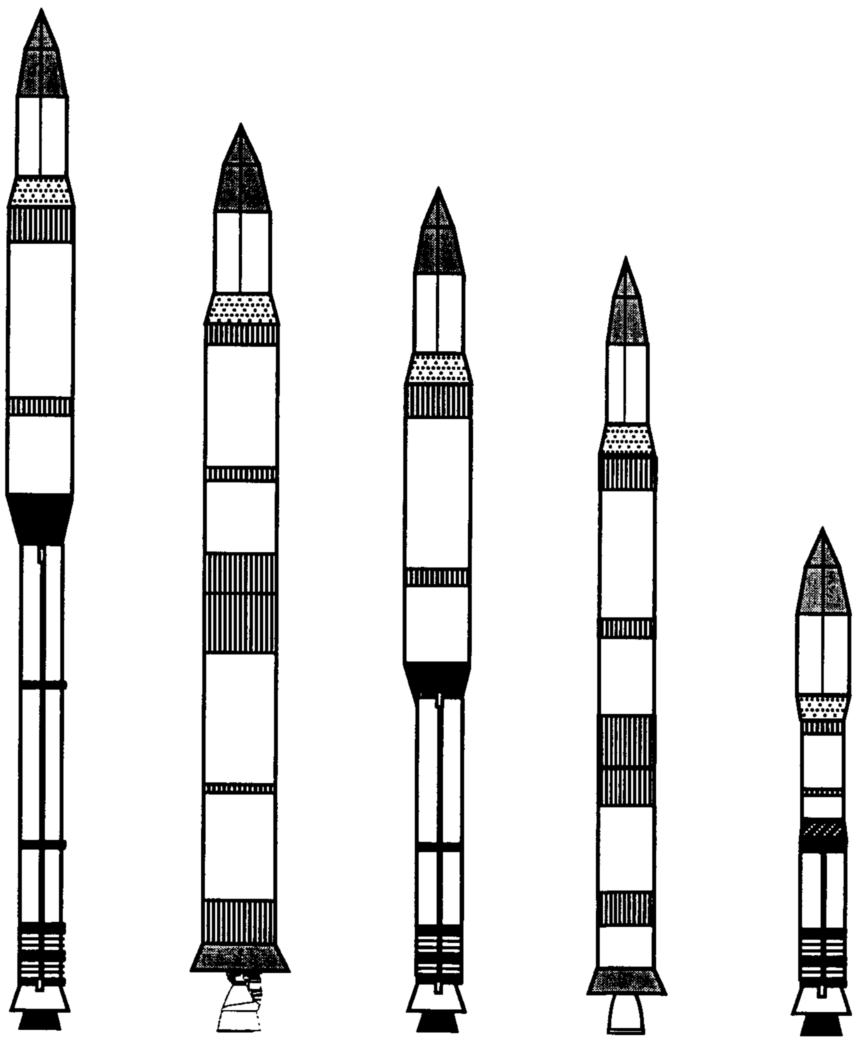
3.0 Liquid/Liquid Configurations

4.0 Hybrid /Liquid Configurations

5.0 ASRM/Liquid Configurations

Lockheed

K. A. Holden
205-722-4531
Kevin D. Sagis
205-722-4532



Vehicle Configuration Summaries

K. A. Holden
205-722-4531
Kevin D. Sagis
205-722-4532

Lockheed

Vehicle Configuration Matrix

First Stage/Second Stage Options		
Liquid/Liquid *	**Hybrid/Liquid ***	**Solid/Liquid**
F-1A/LCSSME F-1A/J-2S F-1A/SSME F-1A/RD-0120 F-1A/Vulcain	Staged Combustion Hybrid/LCSSME Staged Combustion Hybrid/Rubber STME Staged Combustion Hybrid/J-2S Staged Combustion Hybrid/Vulcain Staged Combustion Hybrid/RD-0120	3 Segment ASRM/LCSSME 3 Segment ASRM/LCSSME
STME/LCSSME STME/STME STME/RD-0120 STME/Vulcain	Classical Hybrid/LCSSME Classical Hybrid/Rubber STME Classical Hybrid/J-2S Classical Hybrid/Vulcain Classical Hybrid/RD-0120	2 Segment ASRM/J-2S 2 Segment ASRM/LCSSME 2 Segment ASRM/SSME
M-1A/LCSSME M-1A/RD-0120 M-1A/Vulcain		1 Segment ASRM/Centaur
RD-170/LCSSME RD-170/J-2S RD-170/RD-0120 RD-170/Vulcain		
LCSSME/LCSSME LCSSME/RD-0120 LCSSME/Vulcain		

Note: * Configurations sized for 50 Klbm payload

K. A. Holden
205-722-4531

Vehicle Configuration Payload

Liquid/Liquid *	Payload (lbm) **
F-1A/LCSSME	48,249
F-1A/J-2S	54,893
F-1A/SSME	51,098
F-1A/RD-0120	48,599
F-1A/Vulcain	49,155
STME/LCSSME	48,321
STME/STME	48,034
STME/RD-0120	50,186
STME/Vulcain	49,986
M-1A/LCSSME	47,992
M-1A/RD-0120	49,471
M-1A/Vulcain	48,993
RD-170/LCSSME	49,878
RD-170/J-2S	50,166
RD-170/RD-0120	48,598
RD-170/Vulcain	50,598
LCSSME/LCSSME	48,222
LCSSME/RD-0120	49,339
LCSSME/Vulcain	49,071

Note: * First Stage/Second Stage Propulsion Options
** Payloads verified by 3-DOF trajectory analysis

K. A. Holden
205-722-4531

Vehicle Configuration Payload (Concluded)

Hybrid/Liquid *	Payload (lbm) **
Staged Combustion Hybrid/LCSSME	51,773
Staged Combustion Hybrid/Rubber STME	54,836
Staged Combustion Hybrid/J-2S	50,610
Staged Combustion Hybrid/Vulcain	50,072
Staged Combustion Hybrid/RD-0120	51,354
Classical Hybrid/LCSSME	50,499
Classical Hybrid/Rubber STME	53,733
Classical Hybrid/J-2S	49,111
Classical Hybrid/Vulcain	47,887
Classical Hybrid/RD-0120	50,178

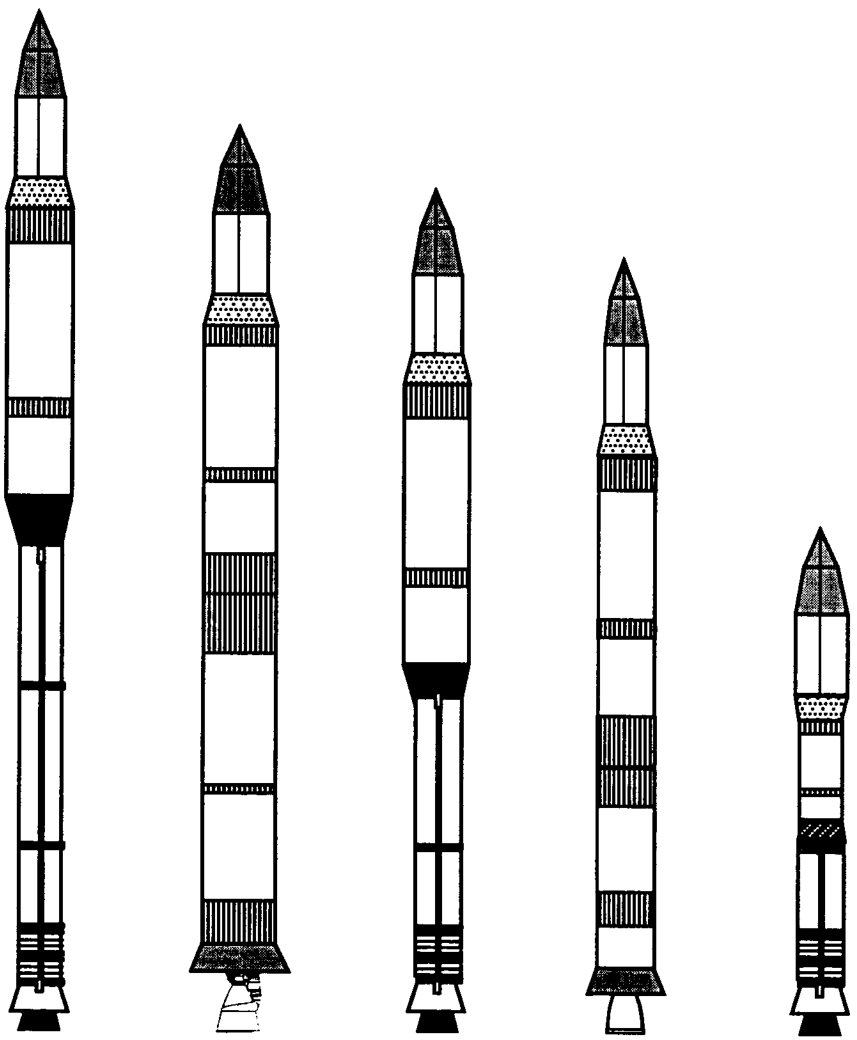
Solid/Liquid *	Payload (lbm) **
3 Segment ASRM/LCSSME	65,000
3 Segment ASRM/SSME	82,100
2 Segment ASRM/J-2S	43,600
2 Segment ASRM/LCSSME	49,300
2 Segment ASRM/SSME	56,600
1 Segment ASRM/Centaur	6,900

Note: * First Stage/Second Stage Propulsion Options
** Payloads verified by 3-DOF trajectory analysis

K. A. Holden
205-722-4531

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Advanced Launch Vehicle Analysis



Hybrid Booster Groundrules

K. A. Holden
205-722-4531
Kevin D. Sagis
205-722-4532

Lockheed

Hybrid Booster Concepts

Performance and Sizing Groundrules and Assumptions

- Staged Combustion Concepts Only
  - -- Oxidizer Type: Ammonium Perchlorate (AP)
  - -- Fuel Grain Oxidizer Content: overall (Booster) MR is 2.5; the 2.5 is split 1.9 as LOX and the remaining 0.6 is AP within the grain

- Staged Combustion and Classical Concepts:
  - -- Grain Ignition Method: redundant, forward end, tri-ethyl aluminum
  - -- Igniter Weight: 50 lbf
  - -- Thrust Termination Method: termination of LOX flow
  - -- LOX Tank Pressurization Method: autogenous, warm GOX, turbine exhaust
  - -- LOX Tank Ullage Pressure: 60 psia
  - -- Motor Chamber Pressure: 1700 psia
  - -- LOX Injector Inlet Pressure: 2000 psia
  - -- Thrust Chamber Cooling Method: regenerative
  - -- TVC Method: electro-mechanical actuators
  - -- Minimum Throttle Setting: 75 % Rated Power Level

- Residuals
  - -- 2 % of solid propellant load for staged combustion
  - -- 10 % of solid propellant load for classical combustion

K. A. Holden
205-722-4531

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Advanced Launch Vehicle Analysis

TA-2 HYBRID MATRIX, 17 FT DIAMETER "STAGED COMBUSTION DESIGN"

PERFORMANCE	PROPELLANT LOADING, LBF							
	800,000	762,963	700,000	650,000	600,000	500,000	400,000	270,000 *
THRUST (SL/VAC), LBFX10E+6	1.8/2	1.8/2	1.8/2	1.8/2	1.8/2	1.4/1.58	1.4/1.58	1.4/1.58
SPECIFIC IMPULSE, SL/VAC), SEC	270/305	270/305	270/305	270/305	270/305	270/305	270/305	270/305
BURN TIME, SEC	120	114.4	105	97.5	90	96.4	77.1	52
TOT. IMPULSE, SL, LB-SECX10E+6	216	206	189	175.5	162	135	108	72.9
OXGEN WT., LBF	524137	499872	458621	425862	393103	327506	262069	176896
FUEL WT. (SOLID W/OX.) LBF	275863	263090	241379	224138	206897	172413	137931	93103
OXYGEN TANK LGTH., FT	35.7	34.2	31.7	29.1	26.9	22.4	17.9	12.1
FUEL CASE LGTH., FT	24.7	23.6	21.6	20.1	18.6	15.5	12..4	8.4
LGTH., OF TANK/CASE ENDS, FT	17	17	17	17	17	17	17	17
TCA LGTH., FT	10.4	10.4	10.4	10.4	10.4	9.4	9.4	9.4
OVERALL LGTH., FT	87.8	85.2	80.7	76.6	72.9	64.3	56.7	46.9
OXYGEN TANK WT., LBF	20758	19797	18163	16856	15568	12952	10362	6994
FUEL CASE WT., LBF	74388	45193	41463	38501	35540	29681	23745	16027
STRUCTURE WT., LBF	5500	5500	5500	5500	5500	5500	5500	5500
TCA/TPA WT., LBF	14700	14700	14700	14700	14700	12000	12000	12000
PRESS.SYSTEM WT., LBF	2049	1989	1884	1788	1701	1500	1323	1094
OXYGEN FEED SYS WT., LBF	1431	1368	1252	1165	1078	606	485	329
TVC SYSTEM WT., LBF	1720	1720	1720	1720	1720	1337	1337	1337
TPS SYSTEM WT., LBF	752	752	752	752	752	500	500	500
SEP. SYSTEM WT., LBF	1983	1983	1983	1983	1983	1983	1983	1983
AVIONICS/ELECT.SYS WT., LBF	3367	3367	3367	3367	3367	3367	3367	3367
SAFETY SYSTEM WT., LBF	150	150	150	150	150	150	150	150
I/F ATTACH. SYS WT., LBF	1320	1320	1320	1320	1320	1320	1320	1320
SUBTOTAL WEIGHT LBF	101117	97839	92254	87811	83379	74896	62072	50601
MISCELLANEOUS WEIGHT, LBF	5000	5000	5000	4000	4000	4000	4000	4000
GRAND TOTAL HDWE WT., LBF	106117	102839	97254	91811	87379	74896	66072	54601
PROPELLANT WEIGHT, LBF	8000000	762963	7000000	650000	600000	500000	400000	270000
UNUSEABLES WEIGHT, LBF	8000	76500	7000	6500	6000	5000	4000	2700
GRAND TOTAL (STAGE) WT., LBF	914117	873402	804254	748311	693379	579896	470072	327301
PROPELLANT MASS FRACTION	0.87516	0.87355	0.87037	0.86862	0.86533	0.86222	0.85093	0.82498

NOTES: *RESULTS IN MINIMUM ALLOWABLE FUEL GRAIN LENGTH TO DIAMETER RATIO

Lockheed

K. A. Holden
205-722-4531

TA-2 HYBRID MATRIX, 17 FT DIAMETER "CLASSICAL DESIGN"

PERFORMANCE	PROPELLANT LOADING, LBF							
	800,000	777,358	700,000	650,000	600,000	500,000	400,000	240,000 *
THRUST (SL/VAC), LBF 10E+6	1.8/2	1.8/2	1.8/2	1.8/2	1.8/2	1.4/1.58	1.4/1.58	1.4/1.58
SPECIFIC IMPULSE, SL/VAC), SEC	265/300	265/300	265/300	265/300	265/300	265/300	265/300	265/300
BURN TIME, SEC	117.8	112.3	103.1	95.7	88.3	94.6	75.7	45.2
TOT. IMPULSE, SL, LBM-SEC 10E+6	212	206	185.5	172.3	159	132.5	106	63.6
OXGEN WT., LBF	571449	555276	500018	464303	428587	357143	285714	171428
FUEL WT. (SOLID W/OX.) LBF	228571	222102	200000	185714	171428	142857	114286	68572
OXYGEN TANK LGTH., FT	39.1	38	34.2	31.8	29.3	24.5	19.6	11.8
FUEL CASE LGTH., FT	27.7	27	24.3	22.6	20.8	17.4	13.9	8.3
LGTH., OF TANK/CASE ENDS, FT	17	17	17	17	17	17	17	17
TCA LGTH., FT	9.5	9.5	9.5	9.5	9.5	9.4	9.4	9.4
OVERALL LGTH., FT	93.3	91.5	85	80.9	76.6	68.3	59.9	46.5
OXYGEN TANK WT., LBF	22633	21997	19797	18408	16961	14183	11346	6808
FUEL CASE WT., LBF	53044	51704	46533	43278	39381	33272	26617	15970
STRUCTURE WT., LBF	5500	5504	5500	5500	5500	5500	5500	5500
TCA/TPA WT., LBF	13500	13500	13500	13500	13500	9300	9300	9300
PRESS.SYSTEM WT., LBF	2178	2136	1984	1887	1788	1593	1398	1085
OXYGEN FEED SYS WT., LBF	1606	1565	1409	1310	1206	680	544	326
TVC SYSTEM WT., LBF	1720	1720	1720	1720	1720	1337	1337	1337
TPS SYSTEM WT., LBF	752	752	752	752	752	500	500	500
SEP.SYSTEM WT., LBF	1983	1983	1983	1983	1983	1983	1983	1983
AVIONICS/ELECT. SYS WT., LBF	3367	3367	3367	3367	3367	3367	3367	3367
SAFETY SYSTEM WT., LBF	150	150	150	150	150	150	150	150
I/F ATTACH. SYS WT., LBF	1320	1320	1320	1320	1320	1320	1320	1320
SUBTOTAL WEIGHT LBF	107753	105694	98015	93175	87628	73185	63372	47646
MISCELLANEOUS WEIGHT, LBF	5000	5000	5000	4000	4000	4000	4000	4000
GRAND TOTAL HDWE WT., LBF	112753	110694	103015	97175	91628	77185	67372	51464
PROPELLANT WEIGHT, LBF	800000	777358	700000	650000	600000	500000	400000	270000
UNUSEABLES WEIGHT, LBF	8000	7700	7000	6500	6000	5000	4000	2400
GRAND TOTAL (STAGE) WT., LBF	920753	89572	810015	753675	697628	582185	471372	294046
PROPELLANT MASS FRACTION	0.86885	0.86783	0.86418	0.86244	0.86006	0.85883	0.84859	0.8162

NOTES: *RESULTS IN MINIMUM ALLOWABLE FUEL GRAIN LENGTH TO DIAMETER RATIO

K. A. Holden
205-722-4531

TA-2 HYBRID MATRIX, 17 FT DIAMETER "STAGED COMBUSTION DESIGN"

PERFORMANCE	PROPELLANT LOADING, LBF				
	800,000	762,963	700,000	650,000	600,000
THRUST(SL/VAC), LBFX10E+6	1.8/2	1.8/2	1.8/22	1.8/2	1.8/2
SPECIFIC IMPULSE, (SL/VAC),SEC	270/305	270/305	270/305	270/305	270/305
BURN TIME, SEC.	120	114.4	105	97.5	90
TOT. IMPULSE, SL, LB-SECX10+6	216	206	189	175.5	162
STAGE SIZING					
OXYGEN WT., LBF	524137	499872	458621	425862	393103
FUEL WT. (SOLID w/OX.), LBF	275863	263090	241379	224138	206897
OVERALL LGTH., FT	87.8	85.2	80.7	76.6	72.9
INERT WEIGHT, LBF	101117	97839	92254	87811	83379
CONTINGENCY WEIGHT, LBF	5000	5000	5000	4000	4000
TOTAL DRY WT., LBF	106117	102839	97254	91811	87379
PROPELLANT WEIGHT, LBF	800000	762963	700000	650000	600000
UNUSEABLES WEIGHT, LBF	8000	7600	7000	6500	6000
TOTAL STAGE WT., LBF	914117	873402	804254	748311	693379
PROPELLANT MASS FRACTION	0.87516	0.87355	0.87037	0.86862	0.86533
	500,000	400,000	270,000*		
THRUST(SL/VAC), LBFX10E+6	1.4/1.58	1.4/1.58	1.4/1.58		
SPECIFIC IMPULSE, (SL/VAC),SEC	270/305	270/305	270/305		
BURN TIME, SEC.	96.4	77.1	52		
TOT. IMPULSE, SL, LB-SECX10+6	135	108	72.9		
STAGE SIZING					
OXYGEN WT., LBF	327506	262069	176896		
FUEL WT (SOLID w/OX.) LBF	172413	137931	93103		
OVERALL LGTH., FT	64.3	56.7	46.9		
SUBTOTAL WEIGHT, LBF	74896	62072	50601		
MISCELLANEOUS WEIGHT, LBF	4000	4000	4000		
GRAND TOTAL HDW'E WT., LBF	74896	66072	54601		
PROPELLANT WEIGHT, LBF	500000	400000	270000		
UNUSEABLES WEIGHT, LBF	5000	4000	2700		
GRAND TOTAL STAGE WT., LBF	579896	470072	327301		
PROPELLANT MASS FRACTION	0.86222	0.85093	0.82498		

NOTES: *RESULTS IN MINIMUM ALLOWABLE FUEL GRAIN LENGTH TO DIAMETER RATIO

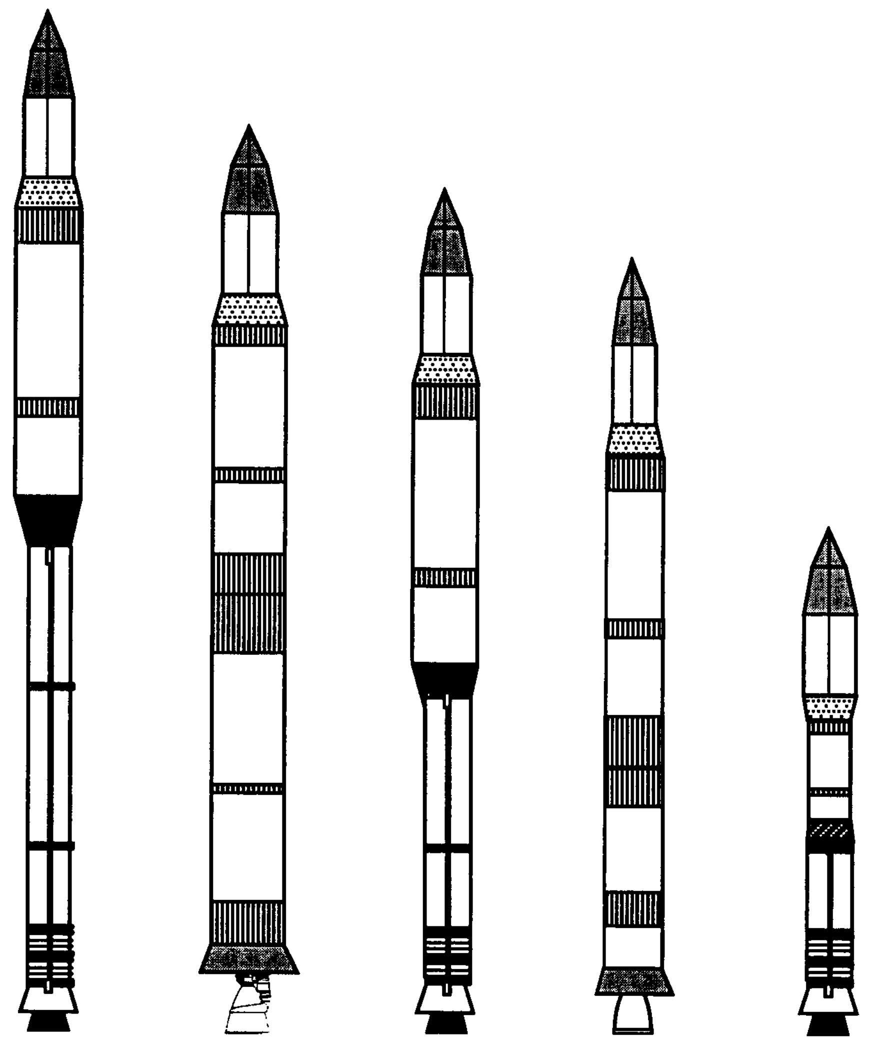
K. A. Holden
205-722-4531

TA-2 HYBRID MATRIX, 17 FT DIAMETER "CLASSICAL DESIGN"

PERFORMANCE	PROPELLANT LOADING, LBF				
	800,000	777,358	700,000	650,000	600,000
THRUST(SL/VAC), LBFX10E+6	1.8/2	1.8/2	1.8/2	1.8/2	1.8/2
SPECIFIC IMPULSE, (SL/VAC),SEC	265/300	265/300	265/300	265/300	265/300
BURN TIME, SEC.	117.8	112.3	103.1	95.7	88.3
TOT. IMPULSE, SL, LB-SECX10+6	212	206	185.5	172.3	159
STAGE SIZING					
OXYGEN WT., LBF	571449	555276	500018	464303	428587
FUEL WT. (SOLID w/OX.) LBF	228571	222102	200000	185714	171428
OVERALL LGTH., FT	93.3	91.5	85	80.9	76.6
SUBTOTAL WEIGHT, LBF	107753	105694	98015	93175	87628
MISCELLANEOUS WEIGHT, LBF	5000	5000	5000	4000	4000
GRAND TOTAL HDW'E WT LBF	112753	110694	103015	97175	91628
PROPELLANT WEIGHT, LBF	800000	777358	700000	650000	600000
UNUSEABLES WEIGHT, LBF	8000	7700	7000	6500	6000
GRAND TOTAL (STAGE) WT., LBF	920753	895752	810015	753675	697628
PROPELLANT MASS FRACTION	0.86885	0.86783	0.86418	0.86244	0.86006
	500,000	400,000	240,000*		
THRUST(SL/VAC), LBFX10E+6	1.4/1.58	1.4/1.58	1.4/1.58		
SPECIFIC IMPULSE, (SL/VAC),SEC	265/300	265/300	265/300		
BURN TIME, SEC	94.6	75.7	45.2		
TOT. IMPULSE, SL, LB-SECX10+6	132.5	106	63.6		
STAGE SIZING					
OXYGEN WT., LBF	357143	285714	171428		
FUEL WT (SOLID w/OX.) LBF	142857	114286	68572		
OVERALL LGTH., FT	68.3	59.9	46.5		
SUBTOTAL WEIGHT, LBF	73185	63372	47646		
MISCELLANEOUS WEIGHT, LBF	4000	4000	4000		
GRAND TOTAL HDW'E WT., LBF	77185	67372	51464		
PROPELLANT WEIGHT, LBF	500000	400000	240000		
UNUSEABLES WEIGHT, LBF	5000	4000	2400		
GRAND TOTAL STAGE WT., LBF	582185	471372	294046		
PROPELLANT MASS FRACTION	0.85883	0.84859	0.81620		

NOTES: *RESULTS IN MINIMUM ALLOWABLE FUEL GRAIN LENGTH TO DIAMETER RATIO

K. A. Holden
205-722-4531

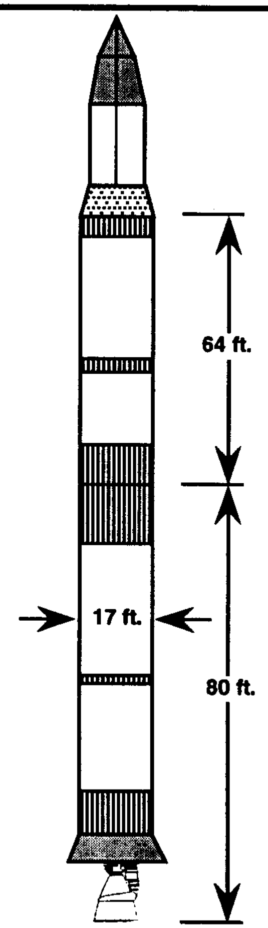


Liquid/Liquid Configurations

Lockheed

K. A. Holden
205-722-4531
Kevin D. Sagis
205-722-4532

50 k Vehicle, F-1A/ LCSSME



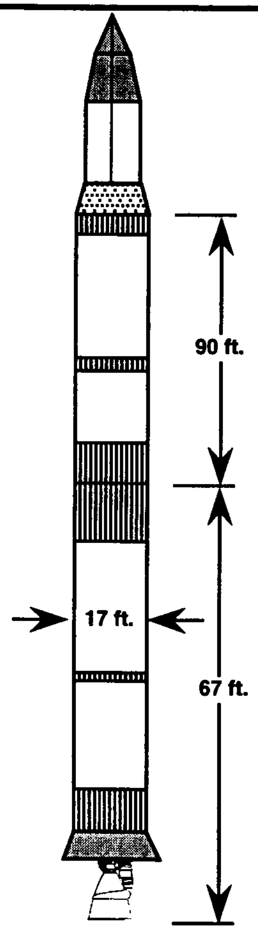
Length/Diameter= 8.5

Payload:	48,249 lbm (21.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	996,632 lbm
First Stage:	
Inert Mass:	69,190 lbm
Usable Propellant:	665,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	81.8 %
Second Stage:	
Inert Mass:	24,193 lbm
Usable Propellant:	190,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.236 g

K. A. Holden
205-722-4531

50 k Vehicle, F-1A/ SSME



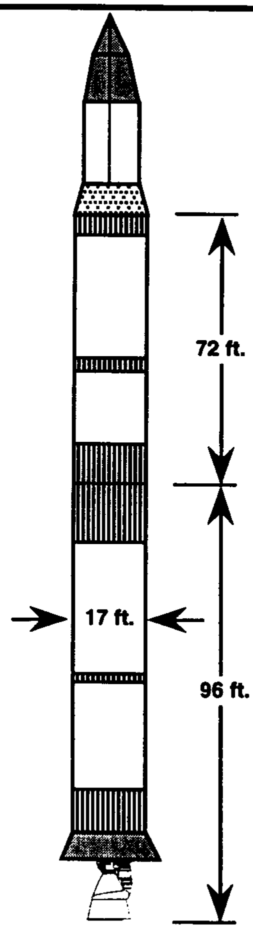
Length/Diameter= 9.2

Payload:	51,098 lbm (23.2 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	940,454 lbm
First Stage:	
Inert Mass:	64,841 lbm
Usable Propellant:	478,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	77.0 %
Second Stage:	
Inert Mass:	31,515 lbm
Usable Propellant:	315,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.233 g

K. A. Holden
205-722-4531

50 k Vehicle, F-1A/ J-2S



Length/Diameter= 9.9

Payload:	54,893 lbm (24.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,177,785 lbm

First Stage:

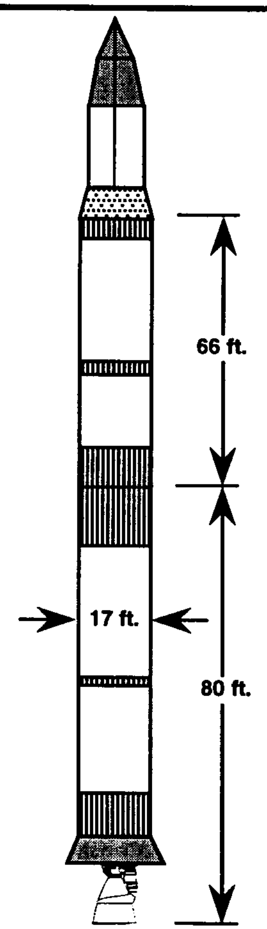
Inert Mass:	68,458 lbm
Usable Propellant:	819,635 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	96.1 %

Second Stage:

Inert Mass:	24,881 lbm
Usable Propellant:	210,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.930 g

K. A. Holden
205-722-4531

50 k Vehicle, F-1A/ RD-0120



Length/Diameter= 8.6

Payload:	48,599 lbm (22.0 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,004,641 lbm

First Stage:

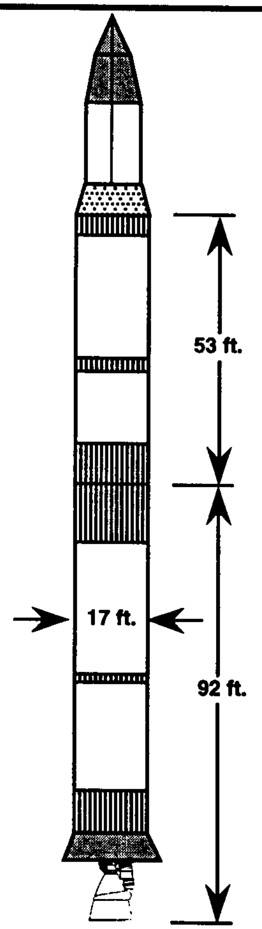
Inert Mass:	69,619 lbm
Usable Propellant:	665,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	82.4 %

Second Stage:

Inert Mass:	26,423 lbm
Usable Propellant:	195,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RD-0120/1
Diameter:	17.0 ft
Thrust/Weight:	1.422 g
Throttle Setting:	87.5 %

K. A. Holden
205-722-4531

50 k Vehicle, F-1A/ Vulcain



Length/Diameter= 8.5

Payload:	49,155 lbm (22.3 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,106,983 lbm

First Stage:

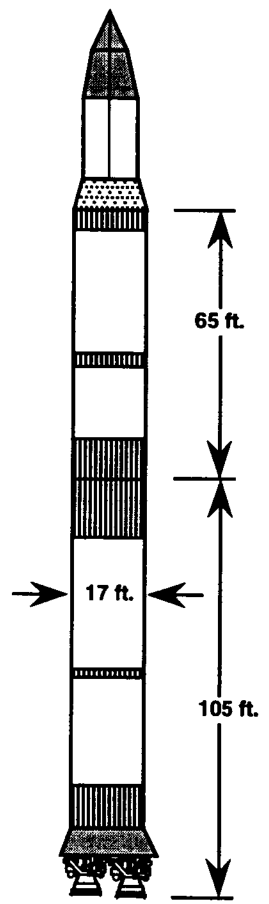
Inert Mass:	71,257 lbm
Usable Propellant:	825,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	90.8 %

Second Stage:

Inert Mass:	16,571 lbm
Usable Propellant:	145,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	1.087 g

K. A. Holden
205-722-4531

50 k Vehicle, STME/ LCSSME



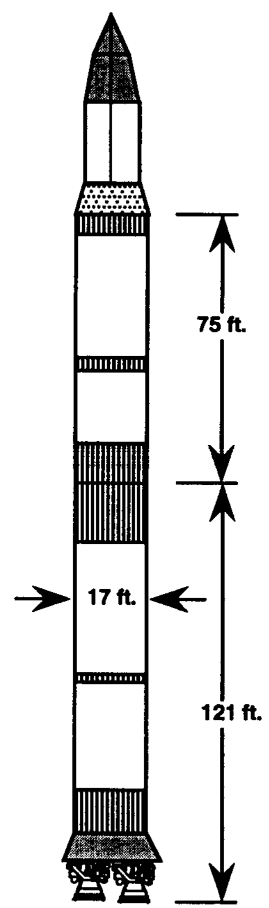
Length/Diameter= 10.0

Payload:	48,321 lbm (21.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	721,109 lbm
First Stage:	
Inert Mass:	68,505 lbm
Usable Propellant:	385,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/2
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	96.7 %
Second Stage:	
Inert Mass:	24,283 lbm
Usable Propellant:	195,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.213 g

K. A. Holden
205-722-4531

50 k Vehicle, STME/ STME



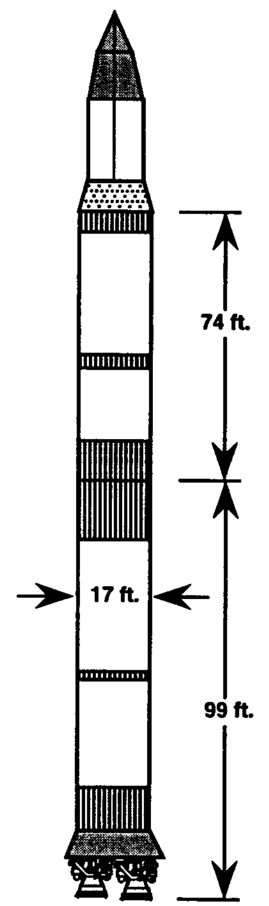
Length/Diameter= 11.5

Payload:	48,034 lbm (21.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	865,286 lbm
First Stage:	
Inert Mass:	73,559 lbm
Usable Propellant:	460,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/2
Diameter:	17.0 ft
Thrust/Weight:	1.272 g
Throttle Setting:	100.0 %
Second Stage:	
Inert Mass:	33,693 lbm
Usable Propellant:	250,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/1
Diameter:	17.0 ft
Thrust/Weight:	1.422 g
Throttle Setting	73.0 %

K. A. Holden
205-722-4531

50 k Vehicle, STME/ RD-0120



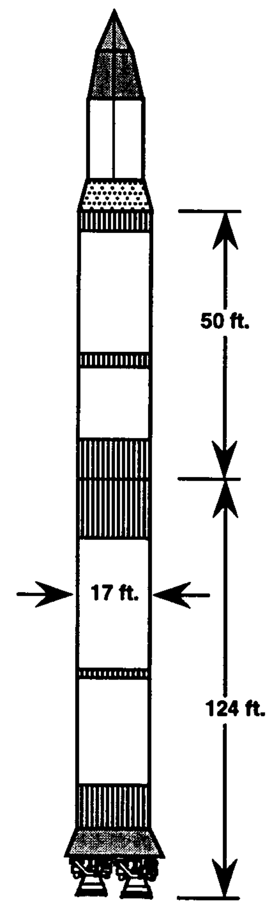
Length/Diameter= 10.2

Payload:	50,186 lbm (22.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	735,478 lbm
First Stage:	
Inert Mass:	67,325 lbm
Usable Propellant:	355,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/2
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	98.3 %
Second Stage:	
Inert Mass:	27,967 lbm
Usable Propellant:	235,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RD-0120/1
Diameter:	17.0 ft
Thrust/Weight:	1.409 g

K. A. Holden
205-722-4531

50 k Vehicle, STME/ Vulcain



Length/Diameter= 10.2

Payload:	49,986 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 747,902 lbm

First Stage:

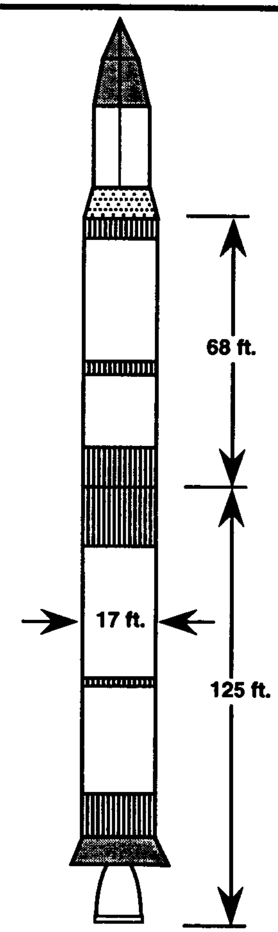
Inert Mass:	72,099 lbm
Usable Propellant:	480,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/2
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	100.0 %

Second Stage:

Inert Mass:	15,817 lbm
Usable Propellant:	130,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	1.175 g

K. A. Holden
205-722-4531

50 k Vehicle, M-1A/ LCSSME



Length/Diameter= 11.3

Payload:	47,992 lbm (21.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	763,823 lbm

First Stage:

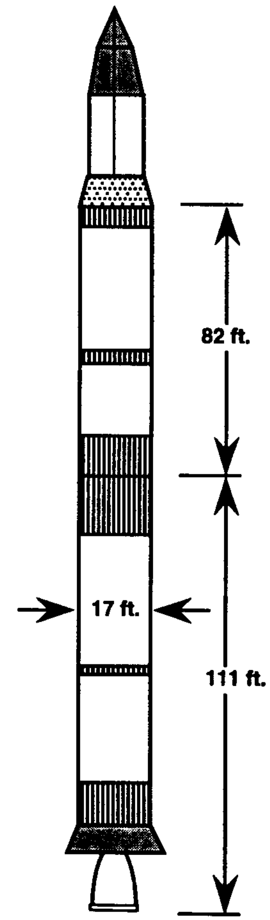
Inert Mass:	75,801 lbm
Usable Propellant:	405,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	M-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	87.2 %

Second Stage:

Inert Mass:	25,035 lbm
Usable Propellant:	210,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.146 g

K. A. Holden
205-722-4531

50 k Vehicle, M-1A/ RD-0120



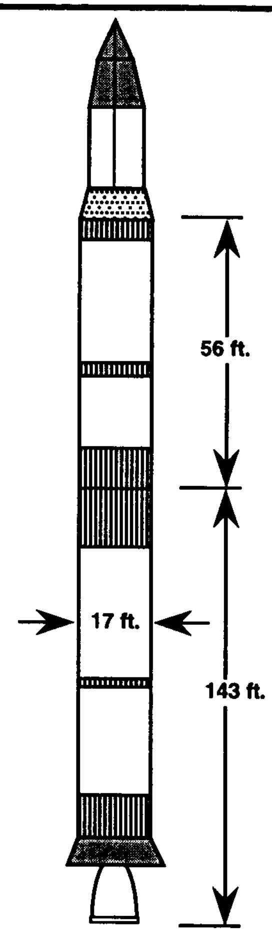
Length/Diameter= 11.4

Payload:	49,471 lbm (22.4 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	770,963 lbm
First Stage:	
Inert Mass:	71,752 lbm
Usable Propellant:	345,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	M-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	87.6%
Second Stage:	
Inert Mass:	29,740 lbm
Usable Propellant:	275,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RD-0120/1
Diameter:	17.0 ft
Thrust/Weight:	1.243 g

K. A. Holden
205-722-4531

50 k Vehicle, M-1A/ Vulcain



Length/Diameter= 11.7

Payload:	48,993 lbm (22.2 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 790,876 lbm

First Stage:

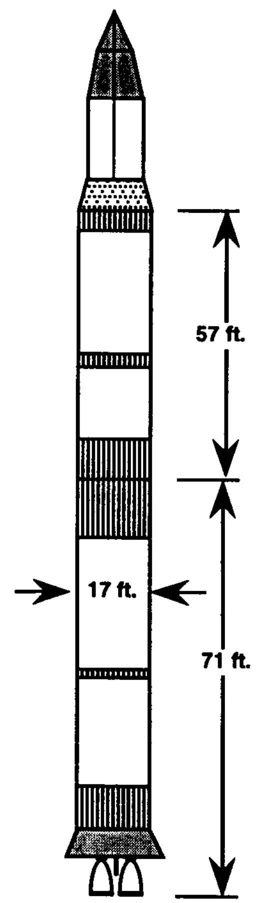
Inert Mass:	79,553 lbm
Usable Propellant:	485,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	M-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	89.8 %

Second Stage:

Inert Mass:	17,330 lbm
Usable Propellant:	160,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	1.012 g

K. A. Holden
205-722-4531

50 k Vehicle, RD-170/ LCSSME



Length/Diameter= 7.5

Payload:	49,878 lbm (22.6 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 929,189 lbm

First Stage:

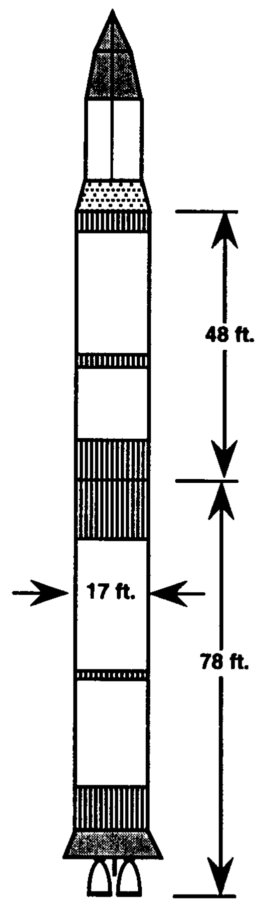
Inert Mass:	66,792 lbm
Usable Propellant:	635,000 lbm
Propellant Type:	LOX/SYN 10
Engine Type/No.:	RD-170/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	84.0 %

Second Stage:

Inert Mass:	22,519 lbm
Usable Propellant:	155,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.422 g
Throttle Setting:	99.1 %

K. A. Holden
205-722-4531

50 k Vehicle, RD-170/ J-2S



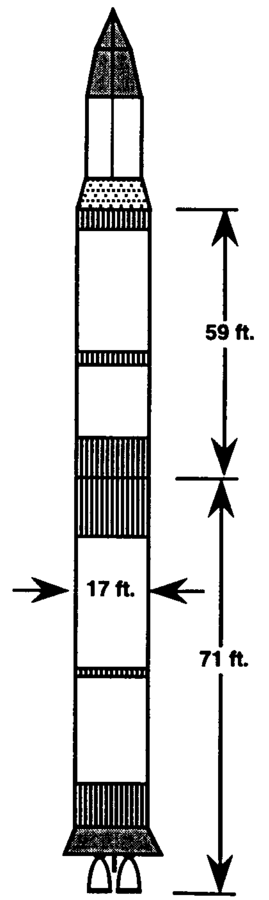
Length/Diameter= 7.4

Payload:	50,166 lbm (22.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	984,311 lbm
First Stage:	
Inert Mass:	67,486 lbm
Usable Propellant:	730,000 lbm
Propellant Type:	LOX/SYN 10
Engine Type/No.:	RD-170/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	88.9 %
Second Stage:	
Inert Mass:	16,659 lbm
Usable Propellant:	120,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	1.420 g

K. A. Holden
205-722-4531

50 k Vehicle, RD-170/ RD-0120



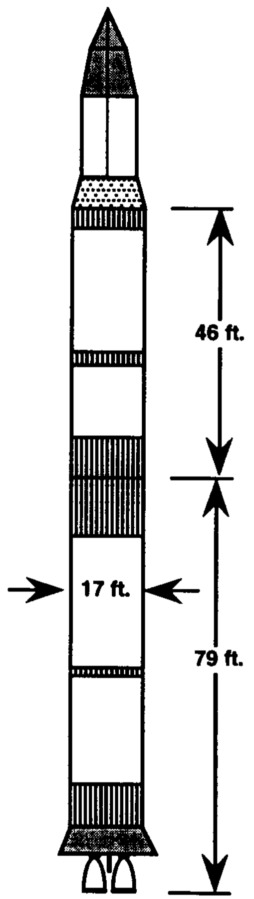
Length/Diameter= 7.6

Payload:	48,598 lbm (22.0 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	940,686 lbm
First Stage:	
Inert Mass:	67,346 lbm
Usable Propellant:	640,000 lbm
Propellant Type:	LOX/SYN 10
Engine Type/No.:	RD-170/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	85.1 %
Second Stage:	
Inert Mass:	24,742 lbm
Usable Propellant:	160,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RD-0120/1
Diameter:	17.0 ft
Thrust/Weight:	1.422 g
Throttle Setting:	75.7 %

K. A. Holden
205-722-4531

50 k Vehicle, RD-170/ Vulcain



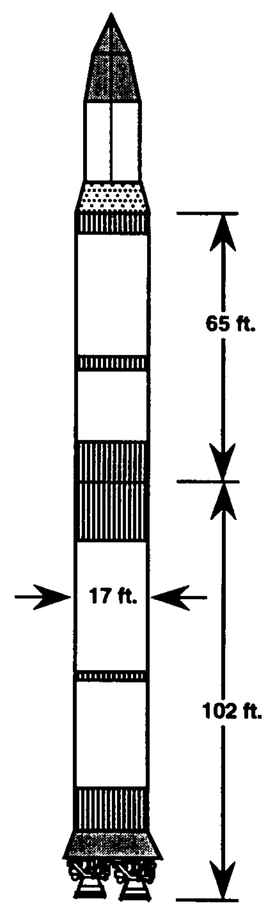
Length/Diameter= 7.3

Payload:	50,598 lbm (22.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	987,885 lbm
First Stage:	
Inert Mass:	67,220 lbm
Usable Propellant:	740,000 lbm
Propellant Type:	LOX/SYN 10
Engine Type/No.:	RD-170/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	89.2 %
Second Stage:	
Inert Mass:	15,066 lbm
Usable Propellant:	115,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	1.277 g

K. A. Holden
205-722-4531

50 k Vehicle, LCSSME/ LCSSME



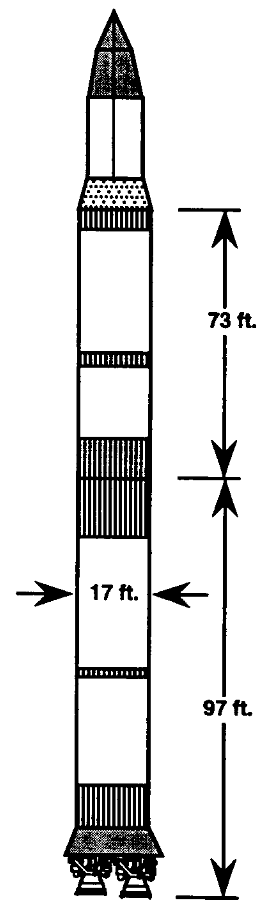
Length/Diameter= 9.8

Payload:	48,222 lbm (21.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	692,379 lbm
First Stage:	
Inert Mass:	59,920 lbm
Usable Propellant:	365,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/2
Diameter:	17.0 ft
Thrust/Weight:	1.458 g
Throttle Setting:	100.0 %
Second Stage:	
Inert Mass:	24,237 lbm
Usable Propellant:	195,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.213 g

K. A. Holden
205-722-4531

50 k Vehicle, LCSSME/ RD-0120



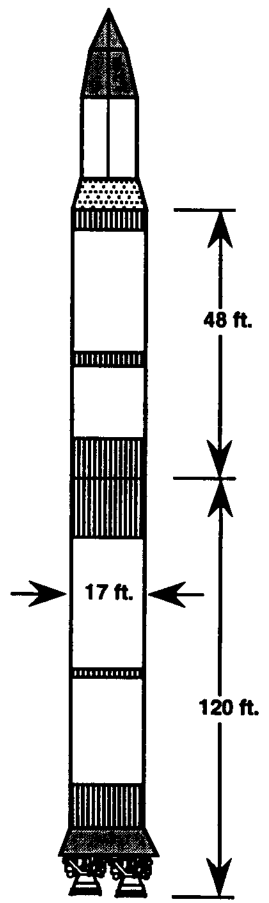
Length/Diameter= 10.0

Payload:	49,339 lbm (22.4 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	705,883 lbm
First Stage:	
Inert Mass:	58,869 lbm
Usable Propellant:	340,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/2
Diameter:	17.0 ft
Thrust/Weight:	1.432 g
Throttle Setting:	100.0 %
Second Stage:	
Inert Mass:	27,675 lbm
Usable Propellant:	230,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RD-0120/1
Diameter:	17.0 ft
Thrust/Weight:	1.422 g
Throttle Setting:	99.2 %

K. A. Holden
205-722-4531

50 k Vehicle, LCSSME/ Vulcain



Length/Diameter= 9.9

Payload:	49,071 lbm (22.2 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 702,207 lbm

First Stage:

Inert Mass:	62,820 lbm
Usable Propellant:	455,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/2
Diameter:	17.0 ft
Thrust/Weight:	1.439 g
Throttle Setting:	100.0 %

Second Stage:

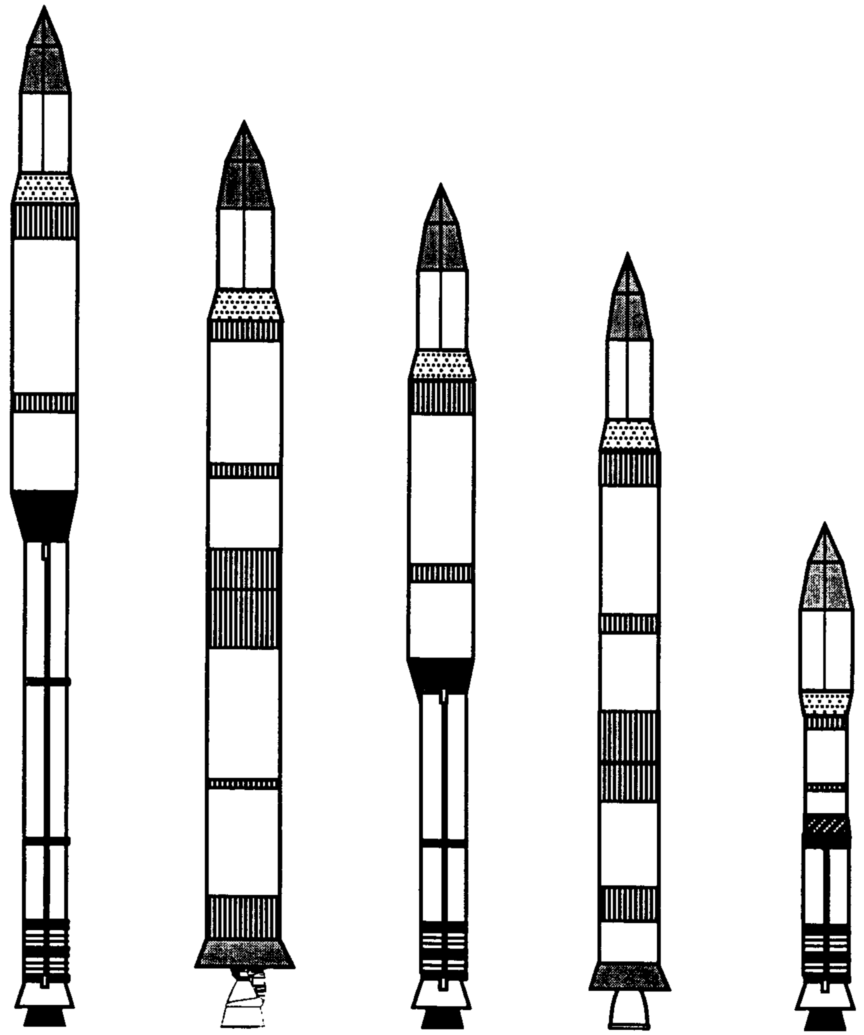
Inert Mass:	15,316 lbm
Usable Propellant:	120,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.241 g

K. A. Holden
205-722-4531

ATSS TEAM

HEAVY LIFT LAUNCH VEHICLE

Advanced Launch Vehicle Analysis

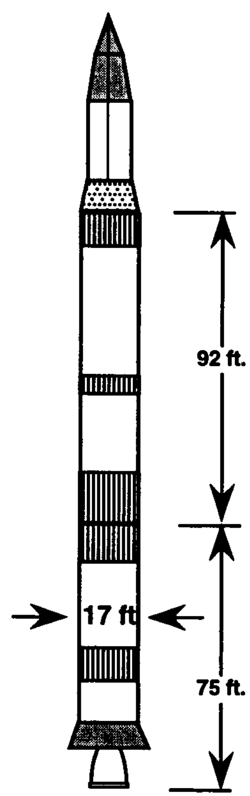


Hybrid/Liquid Configurations

Lockheed

K. A. Holden
205-722-4531
Kevin D. Sagis
205-722-4532

50 k Vehicle, Staged Combustion Hybrid/LCSSME



Length/Diameter= 9.8

Payload:	51,773 lbm (23.5 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,078,893 lbm

First Stage:

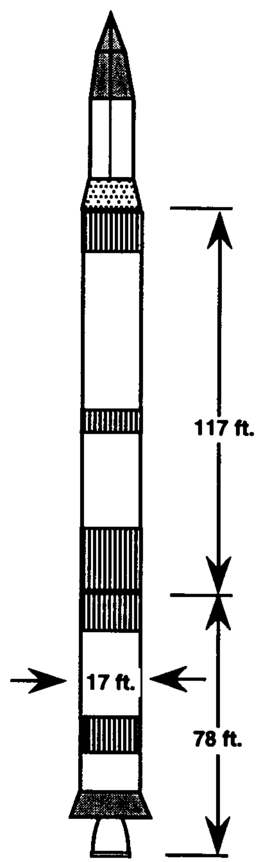
Inert Mass:	90,880 lbm
Usable Propellant:	581,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	88.3 %

Second Stage:

Inert Mass:	30,240 lbm
Usable Propellant:	325,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.806 g

K. A. Holden
205-722-4531

50 k Vehicle, Staged Combustion Hybrid/Rubber STME



Length/Diameter= 11.5

Payload:	54,836 lbm (24.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,302,605 lbm

First Stage:

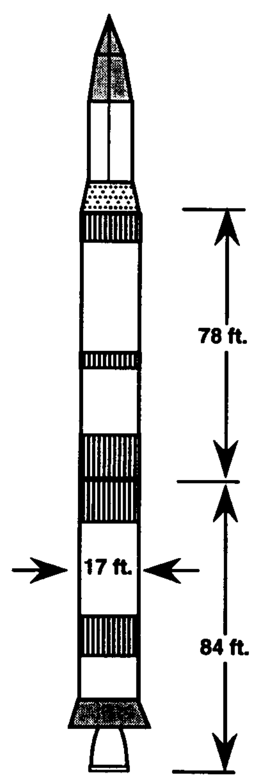
Inert Mass:	95,402 lbm
Usable Propellant:	620,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.443 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	36,367 lbm
Usable Propellant:	446,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	17.0 ft
Vacuum Thrust	425,894 lbf
Thrust/Weight:	0.800 g

K. A. Holden
205-722-4531

Preliminary Data
50 k Vehicle, Staged Combustion Hybrid/J-2S



Length/Diameter= 9.5

Payload:	50,610 lbm (23.0 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,122,891 lbm

First Stage:

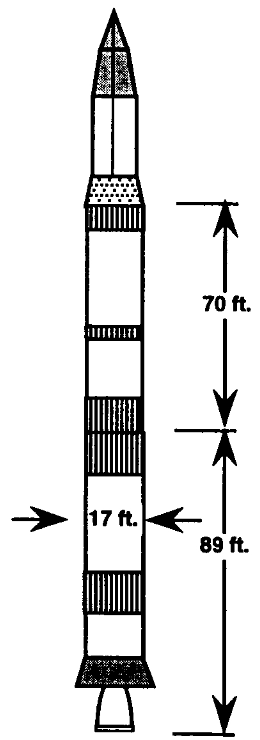
Inert Mass:	102,615 lbm
Usable Propellant:	686,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	92.0 %

Second Stage:

Inert Mass:	23,667 lbm
Usable Propellant:	260,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.794 g

K. A. Holden
205-722-4531

50 k Vehicle, Staged Combustion Hybrid/Vulcain



Length/Diameter= 9.3

Payload:	50,072 lbm (22.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,160,645 lbm

First Stage:

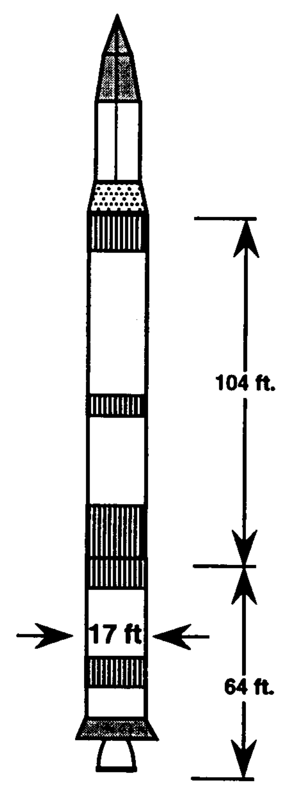
Inert Mass:	110,162 lbm
Usable Propellant:	760,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	95.1 %

Second Stage:

Inert Mass:	20,411 lbm
Usable Propellant:	220,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	0.792 g

K. A. Holden
205-722-4531

50 k Vehicle, Staged Combustion Hybrid/RD-0120



Length/Diameter= 9.9

Payload:	51,354 lbm (23.3 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 980,070 lbm

First Stage:

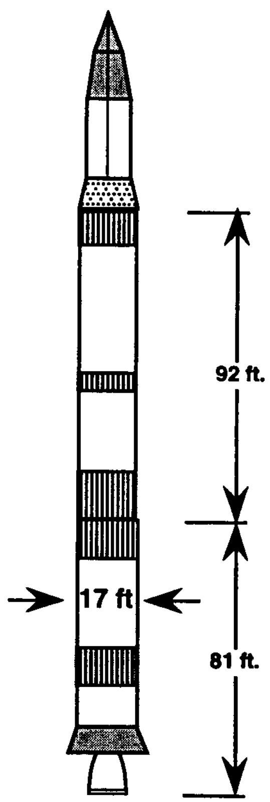
Inert Mass:	74,352 lbm
Usable Propellant:	440,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.430 g
Sea Level Thrust	1,400,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	34,364 lbm
Usable Propellant:	380,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RD-0120/1
Diameter:	17.0 ft
Thrust/Weight:	0.950 g

K. A. Holden
205-722-4531

50 k Vehicle, Classical Hybrid/LCSSME



Length/Diameter= 10.1

Payload:	50, 499 lbm (22.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,114,359 lbm

First Stage:

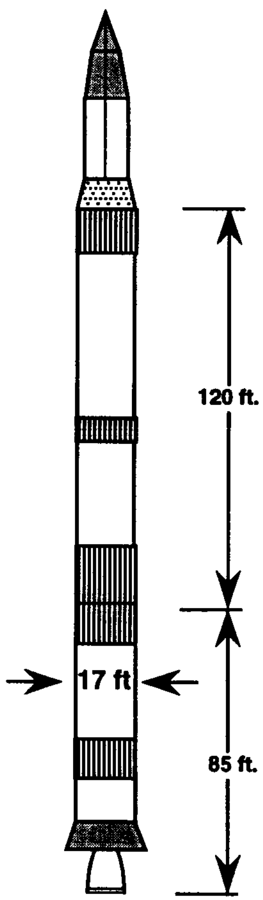
Inert Mass:	108,583 lbm
Usable Propellant:	600,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,630,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	30,277 lbm
Usable Propellant:	325,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.806 g

K. A. Holden
205-722-4531

50 k Vehicle, Classical Hybrid/Rubber STME



Length/Diameter= 12.0

Payload:	53,733 lbm (24.4 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,314,534 lbm

First Stage:

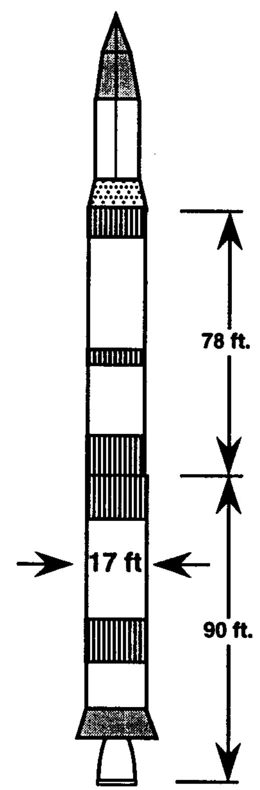
Inert Mass:	115,675 lbm
Usable Propellant:	650,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.386 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	37,126 lbm
Usable Propellant:	458,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	17.0 ft
Vacuum Thrust	436,101 lbf
Thrust/Weight:	0.800 g

K. A. Holden
205-722-4531

50 k Vehicle, Classical Hybrid/J-2S



Length/Diameter= 9.9

Payload:	49,111 lbm (22.3 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,172,288 lbm

First Stage:

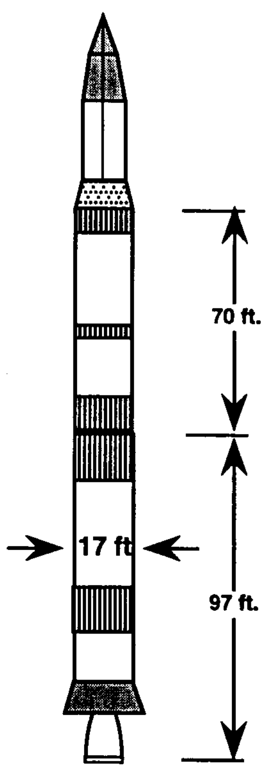
Inert Mass:	123,510 lbm
Usable Propellant:	716,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,714,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	23,667 lbm
Usable Propellant:	260,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.794 g

K. A. Holden
205-722-4531

50 k Vehicle, Classical Hybrid/Vulcain



Length/Diameter= 9.8

Payload:	47,887 lbm (21.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,207,921 lbm

First Stage:

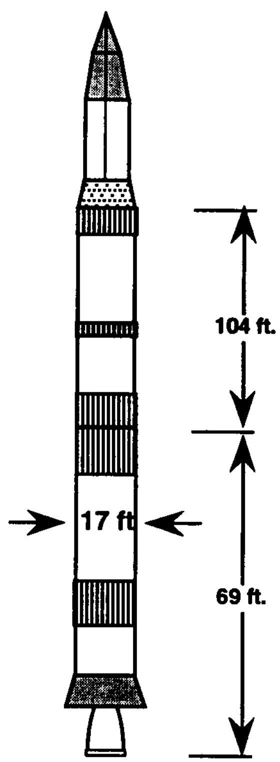
Inert Mass:	129,623 lbm
Usable Propellant:	790,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,785,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	20,411 lbm
Usable Propellant:	220,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	0.792 g

K. A. Holden
205-722-4531

50 k Vehicle, Classical Hybrid/RD-0120



Length/Diameter= 10.2

Payload:	50,178 lbm (22.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,006,598 lbm

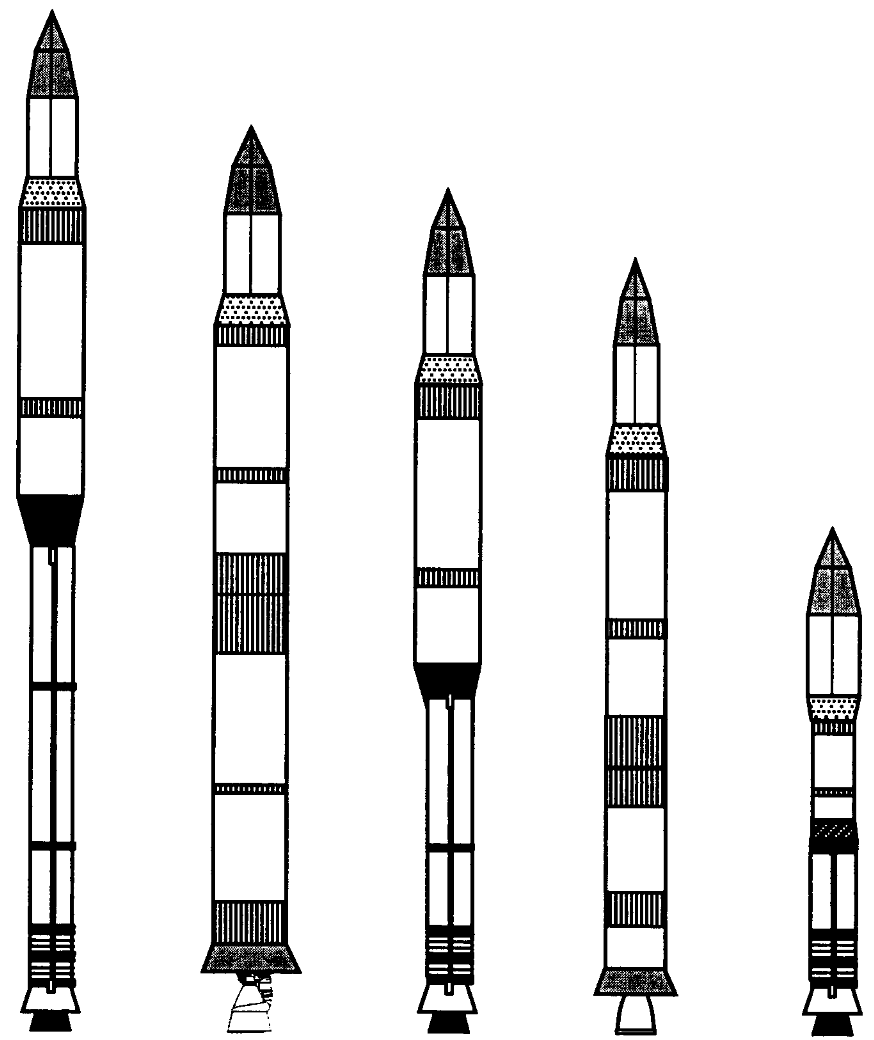
First Stage:

Inert Mass:	87,072 lbm
Usable Propellant:	455,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.404 g
Sea Level Thrust	1,400,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	34,348 lbm
Usable Propellant:	380,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RD-0120/1
Diameter:	17.0 ft
Thrust/Weight:	0.950 g

K. A. Holden
205-722-4531



ASRM/Liquid Configurations

K. A. Holden
205-722-4531
Kevin D. Sagis
205-722-4532

ASRM Configuration Matrix Payload to Low Earth Orbit

First Stage Options	Second Stage Options			
	Centaur	J-2S	LCSSME	SSME
3 Segment ASRM	N.V. *	N.V. *	65,000 lbm	82,100 lbm
2 Segment ASRM	N.V. *	43,600 lbm	49,300 lbm	56,600 lbm
1 Segment ASRM	6,900 lbm	N.V.**	N.V. **	N.V. **

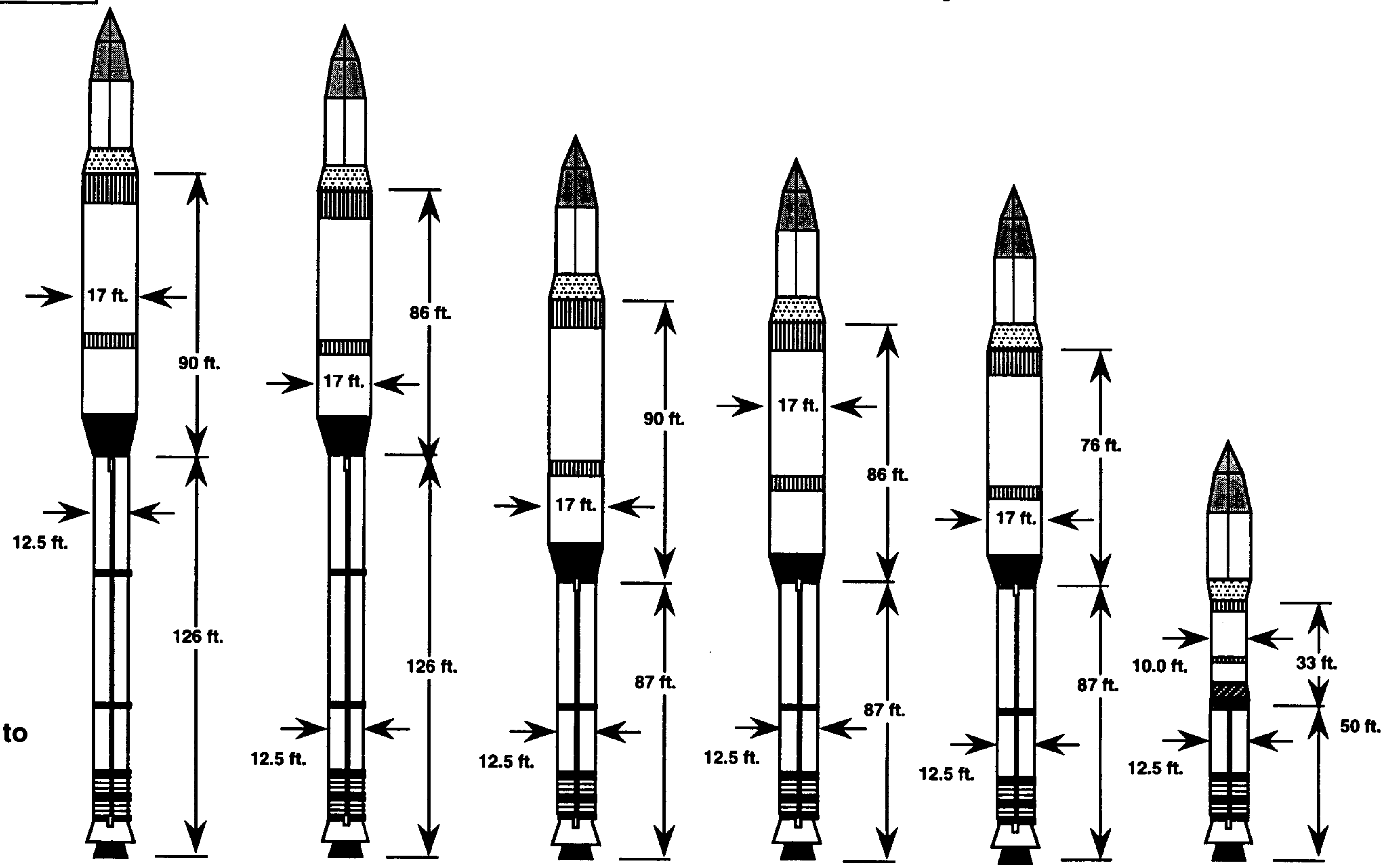
Note: N.V. - Not a Viable Solution
* - Second Stage Thrust to Small
** - First Stage Thrust to Small

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Advanced Launch Vehicle Analysis Focus

ASRM-Derived Launch Vehicle Family



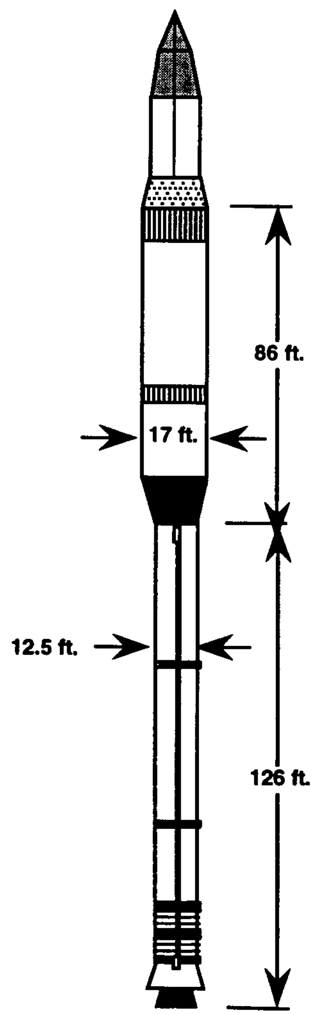
Vehicles not to Scale

1 st Stage	3 Seg ASRM	3 Seg ASRM	2 Seg ASRM	2 Seg ASRM	2 Seg ASRM	1 Seg ASRM
2 nd Stage	1 SSME	1 LCSSME	1 SSME	1 LCSSME	1 J-2s	Centaur
Payload	82,100 lbm	65,000 lbm	56,600 lbm	49,300 lbm	43,600 lbm	6,900 lbm

Lockheed

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

50 k Vehicle, 3 Segment ASRM/LCSSME



Payload:	65,000 lbm (29.5 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,736,481 lbf

First Stage

Inert Mass:	179,947 lbm**
Usable Propellant:	1,214,401 lbm
Propellant Type:	HTPB
Engine Type/No.:	3 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.740 g
Sea Level Thrust	3,020,812 lbf

Second Stage

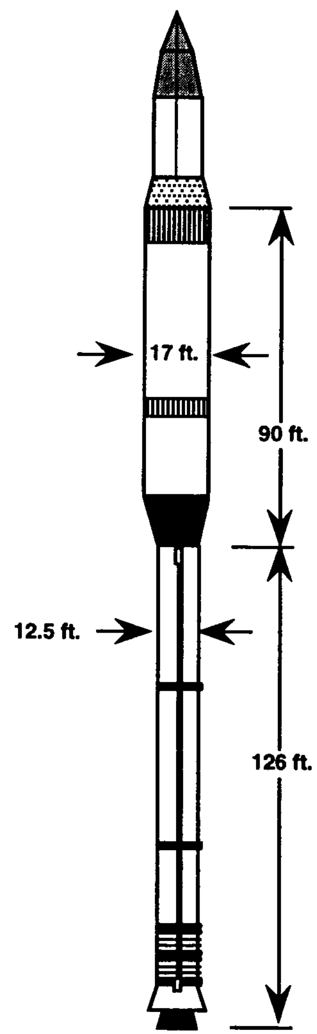
Inert Mass:	27,133 lbm
Usable Propellant:	250,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft*
Thrust/Weight:	0.955 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for best total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 42,300 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses

Lockheed

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

80 k Vehicle, 3 Segment ASRM/SSME



Payload:	82,100 lbm (37.2 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,822,963 lbf

First Stage:

Inert Mass:	179,947 lbm**
Usable Propellant:	1,214,401 lbm
Propellant Type:	HTPB
Engine Type/No.:	3 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.657 g
Sea Level Thrust	3,020,812 lbf

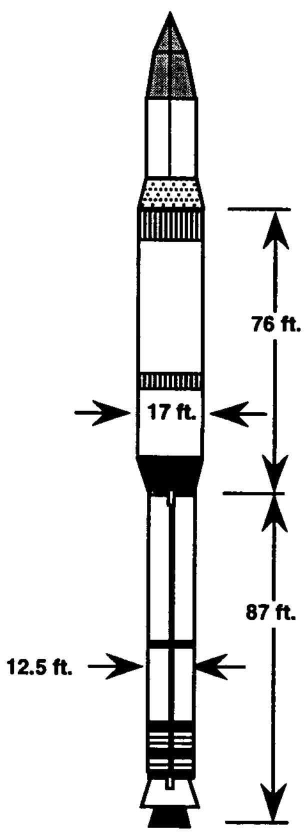
Second Stage:

Inert Mass:	31,515 lbm
Usable Propellant:	315,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft*
Thrust/Weight:	1.140 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for best total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 42,300 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

50 k Vehicle, 2 Segment ASRM/J-2S



Payload:	43,600 lbm (19.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,293,267 lbf

First Stage

Inert Mass:	158,788 lbm**
Usable Propellant:	807,212 lbm
Propellant Type:	HTPB
Engine Type/No.:	2 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.552 g
Sea Level Thrust	2,007,933 lbf***

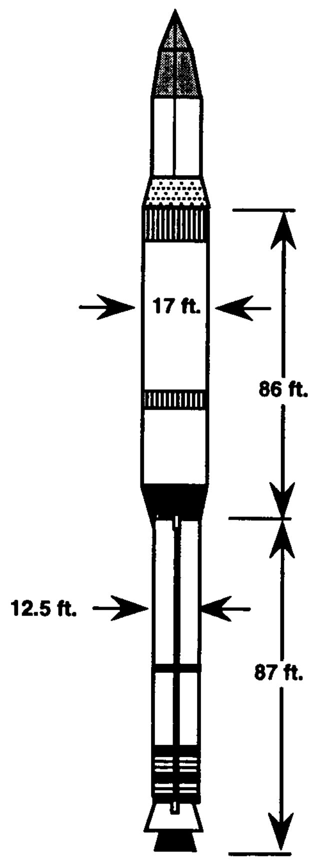
Second Stage

Inert Mass:	23,667 lbm
Usable Propellant:	260,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	12.5 ft *
Thrust/Weight:	0.998 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for <u>best</u> total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 40,500 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses
*** Thrust profile assumed to be a ratio of segment propellant loads

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

50 k Vehicle, 2 Segment ASRM/LCSSME



Payload:	49,300 lbm (22.4 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,339,257 lbf

First Stage

Inert Mass:	158,788 lbm**
Usable Propellant:	807,212 lbm
Propellant Type:	HTPB
Engine Type/No.:	2 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.499 g
Sea Level Thrust	2,007,933 lbf***

Second Stage

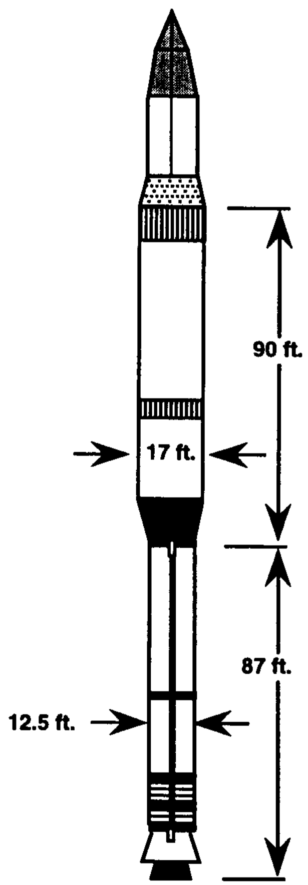
Inert Mass:	28,957 lbm
Usable Propellant:	295,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft*
Thrust/Weight:	0.875 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for best total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 40,500 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses
*** Thrust profile assumed to be a ratio of segment propellant loads

Lockheed

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

50 k Vehicle, 2 Segment ASRM/SSME



Payload:	56,600 lbm (25.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,369,115 lbf

First Stage

Inert Mass:	158,788 lbm**
Usable Propellant:	807,212 lbm
Propellant Type:	HTPB
Engine Type/No.:	2 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.467 g
Sea Level Thrust	2,007,933 lbf ***

Second Stage

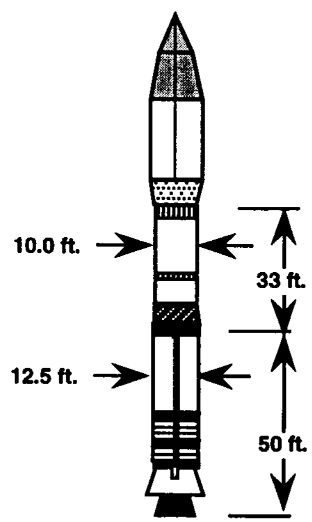
Inert Mass:	31,515 lbm
Usable Propellant:	315,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft*
Thrust/Weight:	1.213 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for best total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 40,500 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses
*** Thrust profile assumed to be a ratio of segment propellant loads

Lockheed

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

1 Segment ASRM/Centaur



Payload:	6,900 lbm (3.2 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 550,700 lbf

First Stage

Inert Mass:	121,530 lbm**
Usable Propellant:	380,470 lbm
Propellant Type:	HTPB
Engine Type/No.:	1 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.719 g
Sea Level Thrust	946,416 lbf ***

Second Stage *

Inert Mass:	4,800 lbm
Usable Propellant:	37,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	RL10A-4/2
Diameter:	10.0 ft *
Thrust/Weight:	0.854 g
Throttle Setting	100.0 %

Note:
* Second stage is a Centaur from the Atlas IIA & IIAS programs
** First stage contains 16,100 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses
*** Thrust profile assumed to be a ratio of segment propellant loads

K. A. Holden 205-722-4531
K. D. Sagis 205-722-4532

Agenda

- BACKGROUND
- METHODOLOGY
- GROUNDRULES & ASSUMPTIONS
- EXAMPLE ESTIMATE
- DISCUSSION

ECON, INC.
JOHN SKRATT
(408) 249-6364

Cost Model Background

THE ADVANCED AEROSPACE COST MODEL (AAFM) ORIGINATED OUT OF ECON'S ADVANCED MISSIONS COST MODEL (AMCM CONTRACT COMPETETIVELY WON OUT OF NASA JOHNSON SPACE CENTER

- DEVELOP GENERAL-CASE PARAMETRIC MODEL
- USE "CLEAN SHEET" APPROACH

DUE TO CIRCUMSTANCES BEYOND OUR CONTROL, NASA TECHNICAL MONITOR DIRECTED ECON TO ABANDON ORIGINAL TECHNICAL APPROACH

ECON CONTINUED ALGORITHM AND MODEL DEVELOPMENT IN-HOUSE WHICH RESULTED IN THE ECON PROPRIETARY GENERAL-CASE TECHNOLOGY FORECASTING COST MODEL

GENERAL-CASE MODEL HAS SINCE BEEN TAILORED AND APPLIED TO

- AIRCRAFT AIRFRAMES
- LAUNCH VEHICLES

ECON, INC.
JOHN SKRATT
(408) 249-6364

Cost Model Methodology

AAFM IS :

- A NON-LINEAR PARAMETRIC COST MODEL APPLICABLE TO SYSTEMS, SUBSYSTEMS AND COMPONENTS
- DEVELOPED FOR "BULLETS TO BATTLESTARS"
- FORECASTS COST IMPACT OF TECHNOLOGY TRENDS
- INCORPORATES BOTH TECH-UP AND TECH-DOWN VARIABLES
- EMPLOYS STATE-OF-THE-ART AND SPECIFICATION VARIABLES
- USES A CALIBRATED PRODUCT DIFFICULTY INDEX

ECON, INC.
JOHN SKRATT
(408) 249-6364

INPUT DATA SET FOR AAFM

VARIABLE NAME	VARIABLE EXPANSION	DIMENSION /UNIT	PRICE EQUIVALENT
WEIGHT	Dry weight including contingency	POUNDS	WT
ELECTRONICS	Factor for electronics in total dry weight	FRACTION	WE
SPECS.	Degree of specifications, standards, imposed on program	(NONE)	PLATFORM
SOTA	Ranking of degrees of product newness	INTEGER (1-12)	ECMPLX, NEWST/EL
QUANTITY	1. Numbers of whole flight articles delivered in production phase 2. Equivalent units used in test	INTEGER REAL	QTY PROTOS
IOC	Year of initial operational capability	YEAR	(NONE)
DIFFICULTY	Degree of product sophistication, as measured at the IOC date	(NONE)	MCPLXS/E
SLOPE	Historical rate of annual complexity growth	FRACTION	(NONE)

AAFM SPECIFICATION-LEVEL (CULTURE) VARIABLE *

VARIABLE	CONTENT	EQUIV PRICE PLATFORM
1.00	GROUND BASED EQUIPMENT	1.0
1.15	SHIPBORNE OR GROUND MOBILE	1.4
1.30	AIRBORNE, MIL-SPEC	1.8
1.67	SPACEBORNE, UNMANNED	2.0
1.80	SPACEBORNE, MANNED	2.5

* SPECIFICATION VARIABLE CAN BE ADJUSTED TO REFLECT ALTERNATIVE CULTURES SUCH AS COMMERCIAL PRACTICES OR SKUNK WORKS

STATE-OF-THE-ART RANKING

RANK	EXPLANATION	EQUIV NASA/OAST TECH. READINESS
1	Virtually 100 percent of drawings exist and need not be renumbered; this is the continuation of an existing product. Example: Orbiter OV-105, as derived from OV-104	NO PARALLEL
2	Predominant number of drawings exist; drawings may have been renumbered. Example: Saturn S-IVB/V stage as derived fromS-IVB/1B	NO PARALLEL
3	Majority of drawings exist; minor resizing of hardware is possible. Example: Gemini Spacecraft structure, as derived from Mercury	NO PARALLEL
4	Roughly half of drawings exist; significant resizing of hardware is possible. Example: Gemini crew systems, as derived from Mercury	NO PARALLEL
5	Only a minority of drawings exist; however, drawings that do not exist are based on a familiar product line. Example: Gemini electrical power system, as derived from Mercury	NO PARALLEL
6	Drawings are essentially new; however, a design point-of-departure is known to exist. Example: Apollo environmental control system, as derived from Gemini	7
7	Drawings are new; the mission or the design concepts are, in part, unfamiliar. Example: Apollo LM landing structure, as derived from Surveyor	6
8	Drawings are new; either the mission or the design concept is unfamiliar. Example: Gemini fuel cells	5
9	Drawings are new; both the mission and the design concepts are unfamiliar. Example: Apollo CSM/LM guidance navigation, as extrapolated from Gemini guidance	4
10	Drawings are new and the design concepts transcend the state-of-the-art. Example: Apollo CM thermal protection	3
11	Drawings are new and the design concepts transcend the state-of-the-art; in addition, multiple design paths are to be followed. Example: Apollo mission as envisioned at the time the manned lunar landing goal was announced	2
12	Drawings are new and the design concepts transcend the state-of-the-art; in addition, only the principles of the mission are known. Example: manned lunar landing mission as viewed from before Sputnik	1

TECHNOLOGY FORECASTING

- SLVCM COSTING ALGORITHMS ALLOW FOR A "TECHNOLOGY" IMPACT
    - "TECH-UP" FORECASTS A DIFFICULTY INDEX OVER TIME
        - TENDENCY OF PARTICULAR TYPE OF "THING" TO BE COSTED, TO CHANGE IN COMPLEXITY FOR A CONSTANT STATE-OF-THE-ART RANKING FOR ANY GIVEN YEAR
    - "TECH-DOWN" ESTIMATES IMPROVEMENTS IN PRODUCEABILITY OF "THING" TO BE COSTED OVER TIME - REPRESENTED BY ORIGINAL TFCM EQUATION OF:

$$E_2 = E_1 e^{.0036(1987-IOC)E_{1.2}}$$

- ENGINEERING JUDGMENT AND DETAIL INTERPRETATION OF HISTORICAL DATA IS REQUIRED TO PROVIDE RATIONALE FOR EXTRAPOLATION OF TECH-UP AND DOWN EFFECTS.

AAFM CALCULATION LOGIC

FACTOR E1 (RECURRING PRODUCTION)

1. Establish cost from product difficulty (complexity) Index
2. Modify cost for specification level (culture) Index

FACTOR E2 (RECURRING PRODUCTION)

1. Modify cost for technology/process improvement

FACTOR E3 (RECURRING PRODUCTION)

1. Modify cost for weight influences
2. Modify cost for percent electronics

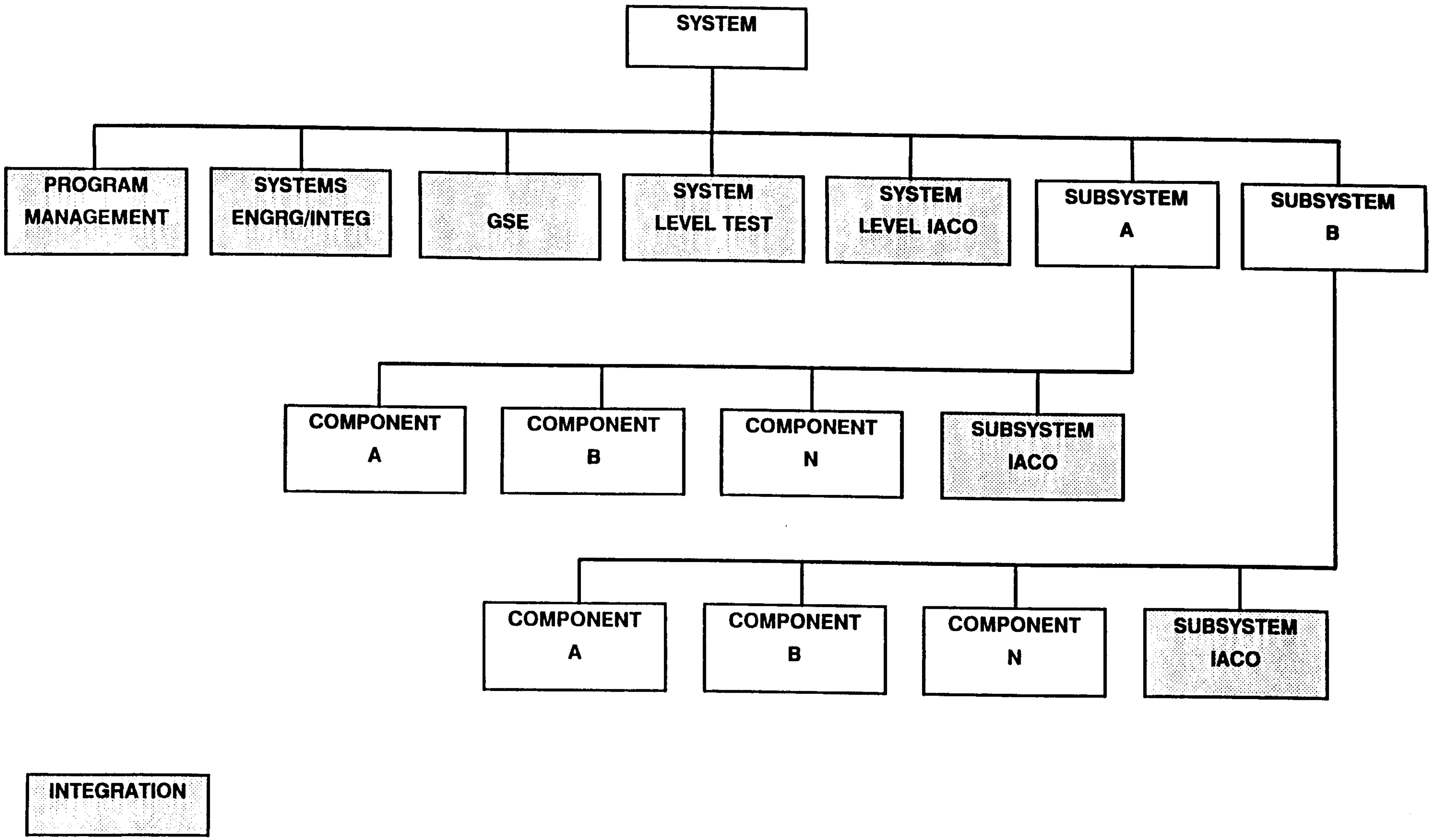
FACTOR E4 (RECURRING PRODUCTION)

1. Modify cost for quantity in production

FACTOR DDT&E (NON-RECURRING)

1. Establish design, development, test and engineering costs
2. Calculate schedule impact on cost (option)

HOW A TOP-DOWN MODEL SEES INTEGRATION



THE INTEGRATION ALGORITHMS

STEP 1: SELECT THE LEVEL AND DIFFICULTY OF INTEGRATION DESIRED
PICK INTEGRATION FACTOR (INFAC):
SUBSYSTEM OR SYSTEM LEVEL DEGREE OF DIFFICULTY

<u>INFAC</u>	
0	NO INTERFACE REQUIRED
	COMPONENT SUBSYSTEM INTEGRATION
3	SIMPLE INTERFACE
5	ROUTINE INTERFACE
7	MODERATELY DIFFICULT INTERFACE
15	DIFFICULT INTERFACE
25	EXCEEDINGLY DIFFICULT INTERFACE
	SUBSYSTEM/SYSTEM INTEGRATION
100	ROUTINE INTERFACE
200	MODERATELY DIFFICULT INTERFACE
300	DIFFICULT INTERFACE

STEP 2: CALCULATE RATIO OF INTEGRATION TO HARDWARE
EXPRESSION IS

SUBSYSTEM/SYSTEM INTEGRATION = 0.2 (INFAC/10).52

MODEL EXAMPLE - INTERSTAGE PRODUCTION COST ESTIMATION

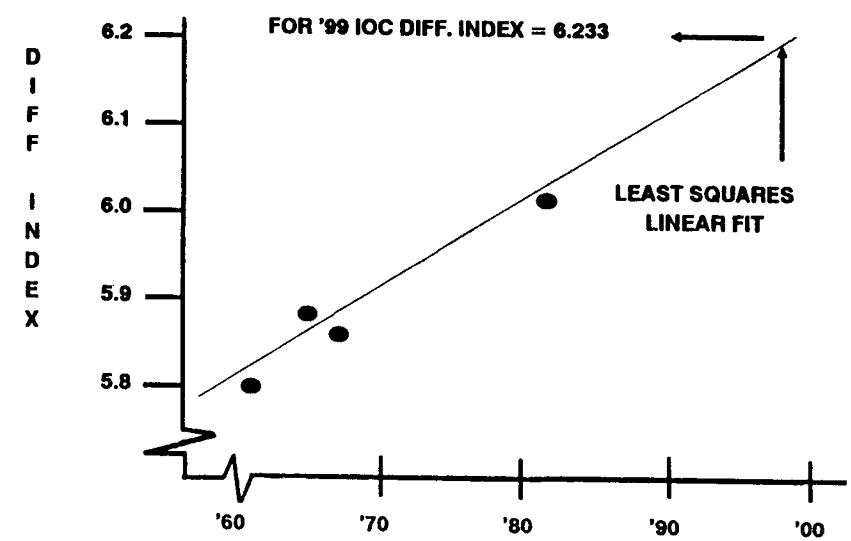
HISTORICAL COST DATA

NAME	IOC	WT	PROD #	YR$	PROD $
CENTAUR D-1A	'62	1154.	7	'72	2.434
S-IVB STAGE	'65	5700.	1.	'69	1.005
S-II STAGE	'67	10132.	15.	'65	10.820
ET	'81	12234.	21	'86	78.423

ECON'S PROPRIETARY SET
OF PRODUCTION COST
ESTIMATING EQUATIONS

VARIABLES INCLUDE:
SPEC. LEVEL, IOC, WEIGHT, QUANTITY PRODUCED
AND DIFFICULTY INDEX

CALIBRATED
VALUES OF DIFFICULTY INDEX & IOC
FOR EACH HISTORICAL DATA POINT



1.5 STAGE PRODUCTION COST

IOC = '99, WT. = 12,395
SPEC. LEVEL = 1.67
PRODUCTION QUANTITY = 26

WITH DIFF INDEX = 6.233
1992 $

TOTAL PRODUCTION COSTS = $110M
AVG UNIT PROD COST = $4.23M

Groundrules & Assumptions

- ALL COSTS PRESENTED IN FY 92 $s
- NASA CODE B NEW START ESCALATION TABLE USED TO NORMALIZE $s
- CURRENT ESTIMATED COSTS INCLUDE DDT&E AND PRODUCTION
- OPERATIONS & FACILITIES NOT YET DEFINED/COSTED
- WEIGHTS BASED ON MASS PROPERTIES SUPPLIED BY MR. KEITH HOLDEN OF LOCKHEED
- WEIGHT INCLUDES CONTINGENCY ALLOCATED TO SUBSYSTEMS (10%)
- MISSION MODEL FOR 50K VEHICLE BASED ON STS/PLS MODEL SUPPLIED BY MR. GENE AUSTIN. INCLUDES TOTAL OF 101 50K VEHICLE FLIGHTS OVER 2003-2010 TIME HORIZON
- WITH EXCEPTION OF ENGINES, ALL SUBSYSTEMS ASSUMED 2 EQUIVALENT TEST ARTICLES

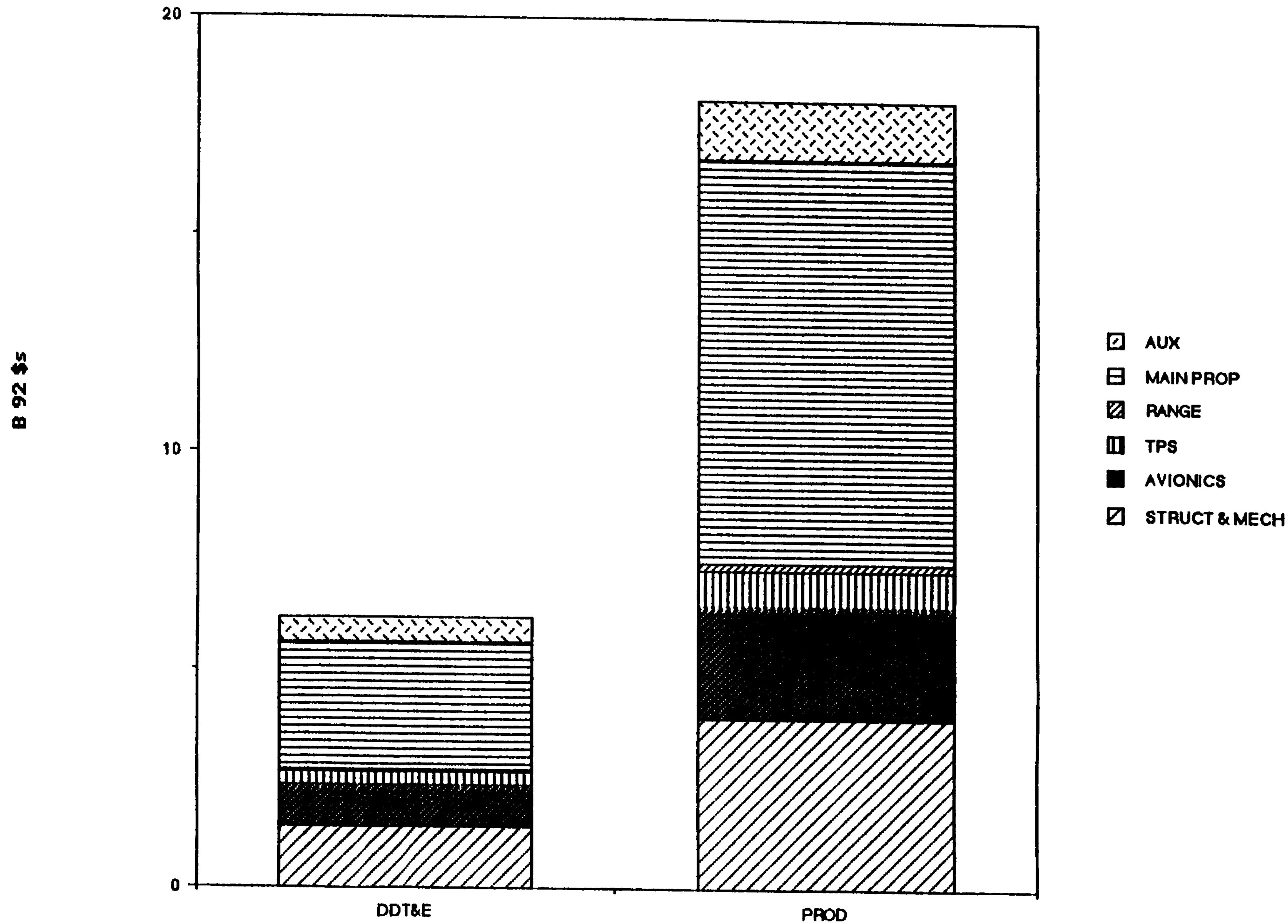
ECON, INC.
JOHN SKRATT
(408) 249-6364

Groundrules & Assumptions (Cont'd)

- ENGINES ASSUMED 7 EQUIVALENT TEST ARTICLES
  - RL-53 (EXPENDABLE SSME)
  - F-1A
- NO "TECH-UP" USED FOR ENGINES (SEE METHODOLOGY)
- NO SCHEDULE IMPACT ASSESSED IN COSTING
- STATE-OF-THE-ART RANKING ASSUMED TO BE NEW DRAWINGS WITH KNOWN POINT-OF-DEPARTURE. ENGINES ASSUME MOST DRAWINGS EXIST
- SPECIFICATION LEVEL SET AT MANNED SPACE DUE TO PLS MISSION
- NO GOVERNMENT "WRAPS" INCLUDED IN ESTIMATE

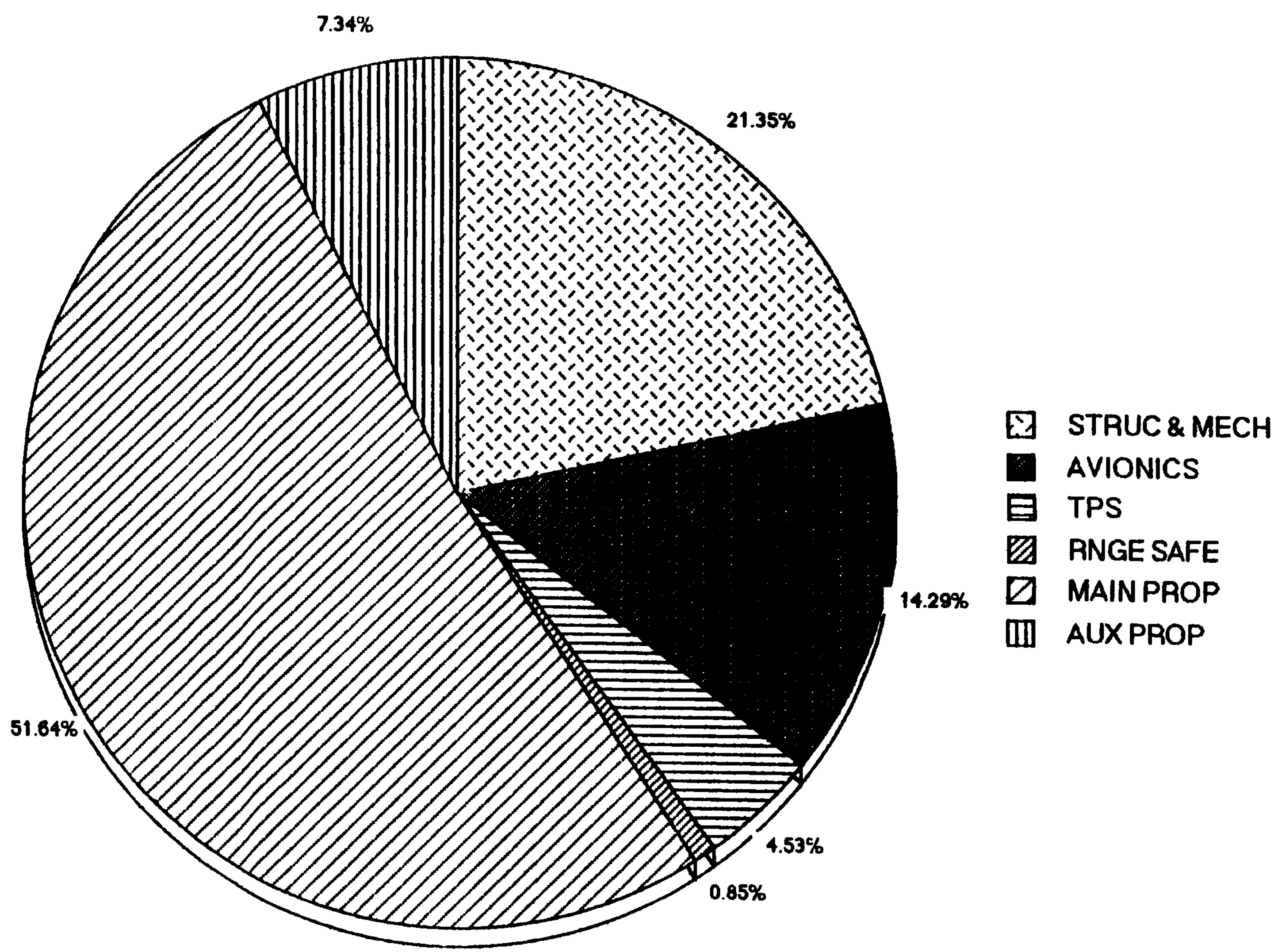
ECON, INC.
JOHN SKRATT
(408) 249-6364

50K/17FT DIAMETER F-1A/RS-53

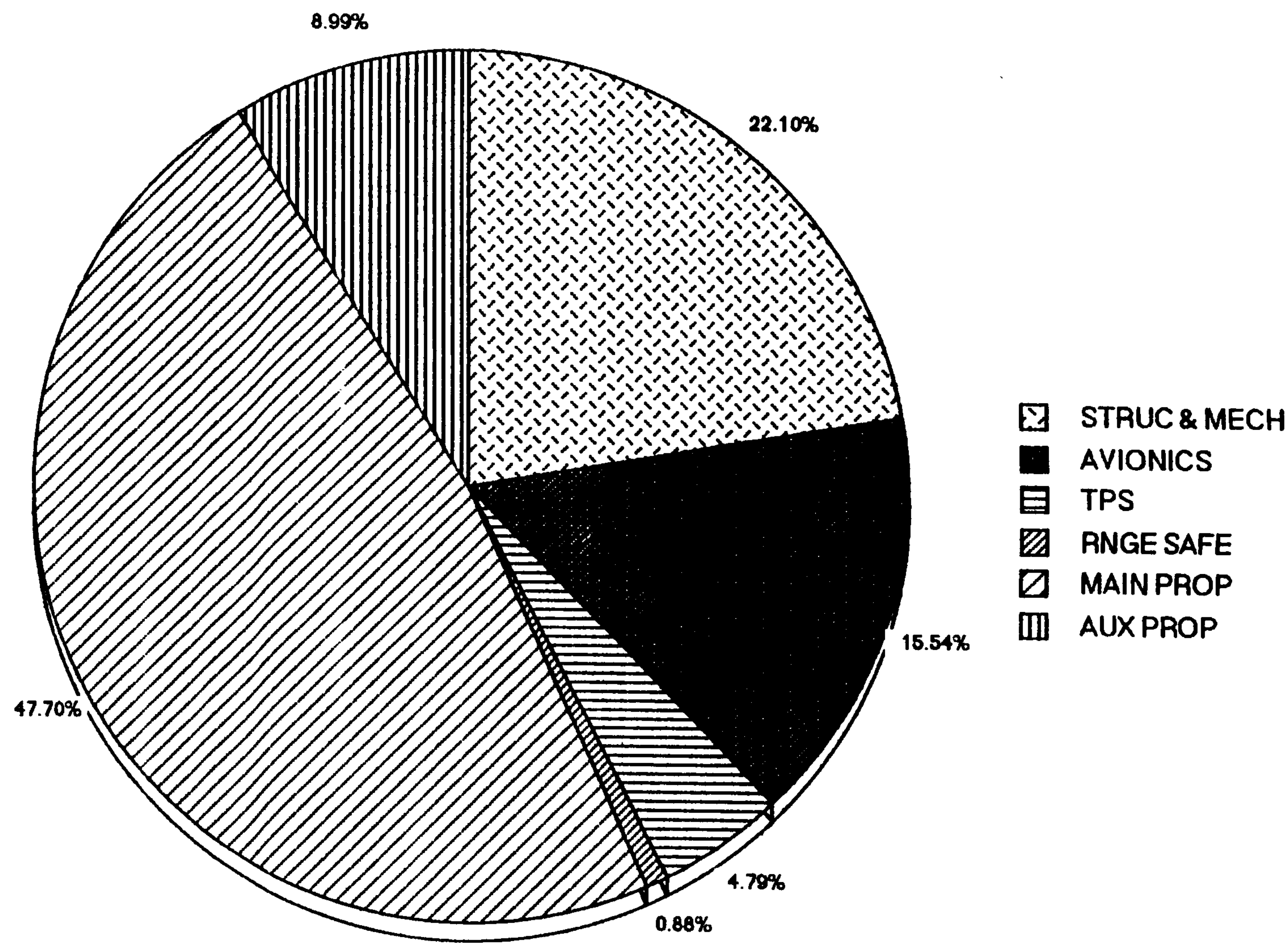


17 Diameter 50K Vehicle (F-1A/RS-53)	Direct Development Cost	Development Integration Cost	Total Development Cost	Direct Production Cost	Production Integration Cost	Total Production Cost	Total Operations Cost	TOTAL Cost
1. INTEGRATED VEHICLE	3.072	3,091	6,163	13,175	4,943	18,118	0	24,281
1. 1. FIRST STAGE ELEMENT	1.616	1,626	3,242	6,947	2,606	9,554	0	12,796
1. 1. 1. STRUCTURES & MECHANISMS	412	415	827	1,708	641	2,349	0	3,176
1. 1. 1. 1. FUEL TANK	87	87	174	359	135	494	0	668
1. 1. 1. 2. OXIDIZER TANK	70	71	141	292	110	402	0	543
1. 1. 1. 3. FWD SKIRT	47	48	95	197	74	271	0	366
1. 1. 1. 4. INTERTANK	16	16	31	64	24	89	0	120
1. 1. 1. 5. AFT SKIRT	56	56	111	230	86	317	0	428
1. 1. 1. 6. THRUST STRUCTURE	79	80	159	328	123	451	0	610
1. 1. 1. 7. INTERSTAGE	57	57	115	237	89	326	0	441
1. 1. 2. ELECTRICAL POWER & DISTRIBUTION	0	0	0	0	0	0	0	0
1. 1. 3. AVIONICS	235	237	472	928	348	1,276	0	1,747
1. 1. 4. THERMAL PROTECTION	62	63	125	253	95	348	0	473
1. 1. 5. RANGE SAFETY	13	14	27	56	21	77	0	104
1. 1. 6. MAIN PROPULSION	830	835	1,666	3,781	1,418	5,199	0	6,865
1. 1. 6. 1. ENGINE MOUNT/GIMBAL/PURGE	120	121	241	491	184	675	0	916
1. 1. 6. 2. FEED/PRESSURIZATION SYSTEM	333	335	669	1,349	506	1,854	0	2,523
1. 1. 6. 3. MAIN ENGINES	377	379	756	1,941	728	2,669	0	3,425
1. 1. 7. AUXILIARY PROPULSION - TVC	63	63	126	222	83	306	0	432
1. 2. SECOND STAGE ELEMENT	1.456	1,465	2,920	6,228	2,337	8,564	0	11,485
1. 2. 1. STRUCTURES & MECHANISMS	267	268	535	1,106	415	1,520	0	2,056
1. 2. 1. 1. FUEL TANK	90	90	180	371	139	510	0	690
1. 2. 1. 2. OXIDIZER TANK	43	44	87	180	67	247	0	334
1. 2. 1. 3. FWD SKIRT	34	35	69	143	54	196	0	266
1. 2. 1. 4. INTERTANK	12	12	23	48	18	66	0	89
1. 2. 1. 5. AFT SKIRT	52	53	105	218	82	299	0	405
1. 2. 1. 6. THRUST STRUCTURE	35	36	71	146	55	201	0	272
1. 2. 1. 7. INTERSTAGE	0	0	0	0	0	0	0	0
1. 2. 2. ELECTRICAL POWER & DISTRIBUTION	0	0	0	0	0	0	0	0
1. 2. 3. AVIONICS	242	244	486	955	358	1,314	0	1,799
1. 2. 4. THERMAL PROTECTION	85	85	170	343	129	472	0	642
1. 2. 5. RANGE SAFETY	13	14	27	56	21	77	0	104
1. 2. 6. MAIN PROPULSION	635	639	1,274	3,024	1,134	4,158	0	5,433
1. 2. 6. 1. ENGINE MOUNT/GIMBAL/PURGE	52	53	105	214	80	295	0	400
1. 2. 6. 2. FEED/PRESSURIZATION SYSTEM	143	144	287	579	217	796	0	1 083
1. 2. 6. 3. MAIN ENGINES	440	442	882	2,231	837	3,068	0	3,950
1. 2. 7. AUXILIARY PROPULSION	213	215	428	745	279	1,024	0	1,452
1. 2. 7. 1. AUXILIARY PROPULSION - RCS	125	125	250	432	162	594	0	843
1. 2. 7. 2. AUXILIARY PROPULSION - TVC	28	28	56	99	37	136	0	192
1. 2. 7. 3. AUXILIARY PROPULSION - Ullage mo	61	61	122	214	80	295	0	417

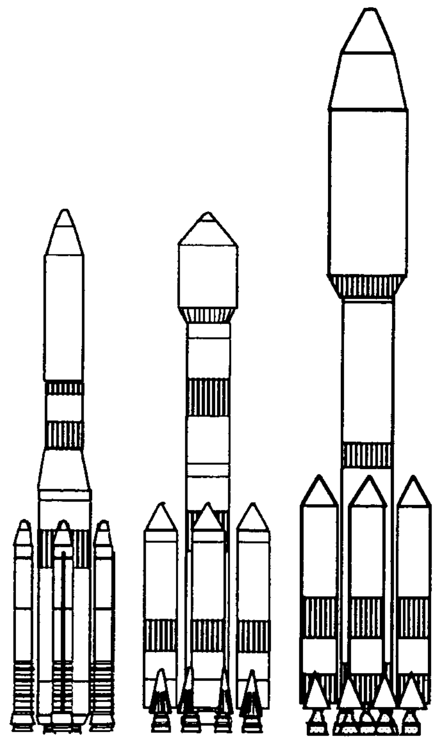
PRODUCTION COST BREAKDWON



DDT&E BREAKDOWN



MSD



Heavy Lift Launch Vehicle Development Contract Status

NAS8-39208

J. B. McCurry/8C-A1

16 April 1993

Lockheed
Missiles & Space Co., Inc.

MSD

ATSS TA-2 Contract (NAS8-39208)
Heavy Lift Launch Vehicle Development

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Current Contract Status:

- Advanced Transportation System Studies (ATSS) Technical Area 2 (TA-2) contract with the Marshall Space Flight Center (one-year FFP with two options)
  --Basic (first year) $925,717; 15 May 1992 through 14 May 1993

- Contract activities during first eight months supported NASA H.Q. Office of Exploration (Mike Griffin) on First Lunar Outpost studies and "Red Team" space transportation requirements studies

- In Dec. 1992 MSFC shifted vehicle design emphasis to 50K payload class launch vehicles, with evolution to lunar mission vehicle; focus on two-stage, series-burn configurations
  --Liquid/liquid; hybrid/liquid; solid/liquid stage options

  --Over 20 configurations assessed to date

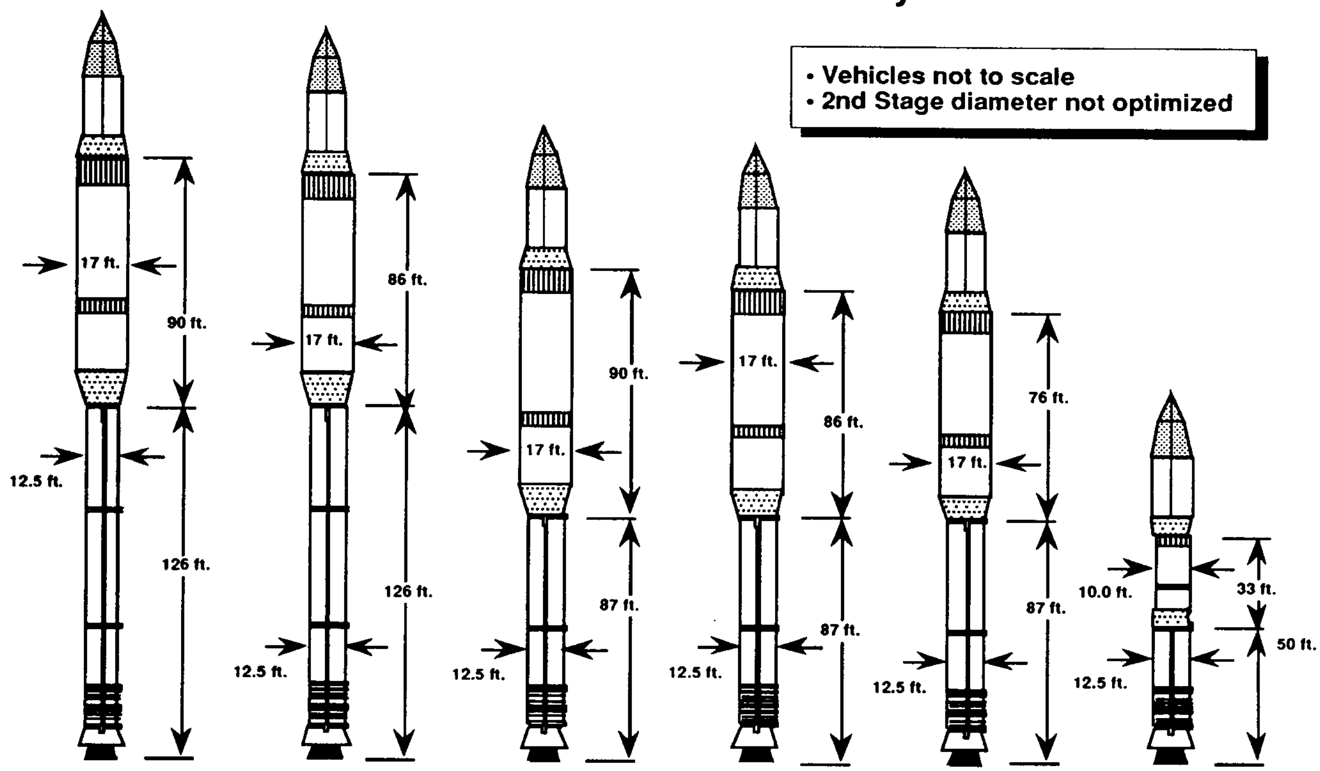
Lockheed
Missiles & Space Co., Inc.

16 April 1993

J. B. McCurry
205-722-4509

ATSS TA-2 Contract (NAS8-39208)
Heavy Lift Launch Vehicle Development

ASRM-Derived Launch Vehicle Family



- Vehicles not to scale
- 2nd Stage diameter not optimized

1 st Stage	3 Seg ASRM	3 Seg ASRM	2 Seg ASRM	2 Seg ASRM	2 Seg ASRM	1 Seg ASRM
2 nd Stage	1 SSME	1 LCSSME	1 SSME	1 LCSSME	1 J-2s	Centaur
Payload*	82,100 lbm	65,000 lbm	56,600 lbm	49,300 lbm	43,600 lbm	6,900 lbm

Note:
* Direct Insertion to 15 x 220 nm, 28.5 deg. inclination orbit

Lockheed
Missiles & Space Co., Inc.

J. B. McCurry
205-722-4509

ATSS TA-2 Contract (NAS8-39208)
Heavy Lift Launch Vehicle Development

Current Contract Status (Concluded):

- MSFC recently added $150K to Basic contract for additional propulsion system definition analysis
  - --Pratt & Whitney to identify performance data and ROM cost of manufacturing RD-170 rocket engine in C.I.S. or U.S.
    - Based on Pratt/NPO Energomash joint venture
  - --Aerojet to provide data on other Russian engines and perform hybrid motor sizing (classical & staged combustion)
- New Change Order being processed to add ~$500K of SDIO money for additional technology, test program, and cost definition of Russian RD-170 and RD-701 engines, with possibility of RD-701 component fab/test in CY93
  - --MSD "allowed" to bid 4 % fee & pgm. mgt. labor hours; remaining $s given to Pratt/Energomash
  - --Will cause Basic contract duration to be extended from 14 May to 31 Dec.

J. B. McCurry
205-722-4509

MSD

ATSS TA-2 Contract (NAS8-39208)
Heavy Lift Launch Vehicle Development

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Future Contract Options:

- Due to Basic contract extension, "2nd year" tasks will be rolled into Basic contract in May and conclude in Dec. '93
  --$678K over 7.5 months

  --Directed to utilize LSOC, Aerojet, & Pratt/Energomash, resulting in no MSD man-power growth (4 EPs); will receive 3 % fee & 4% procurement burden

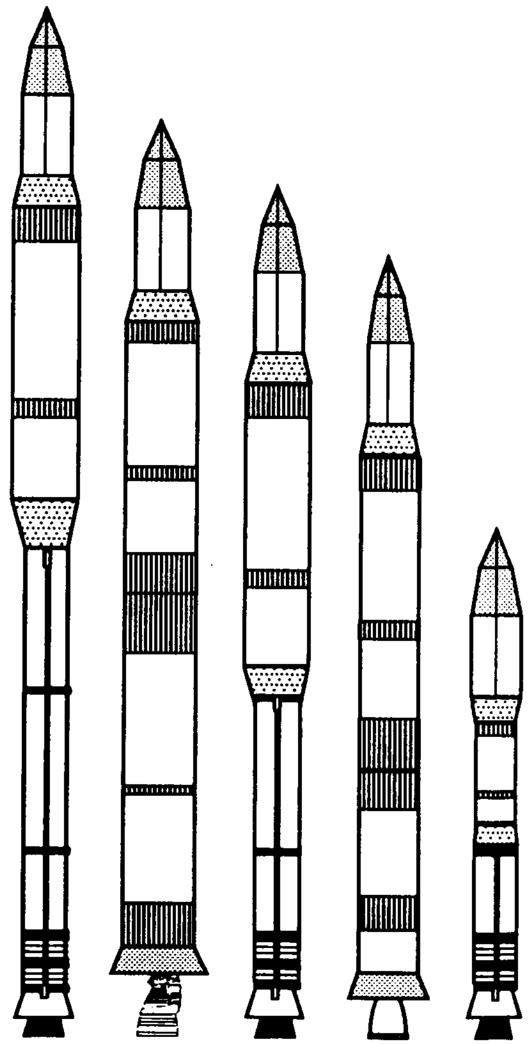
- MSFC rumoring that "3rd year" may be dropped, and TA-2 recompeted in Dec. '93 (along with other ATSS contracts)
  --Form and funding of recompete unknown

  --Current customer satisfaction makes recompetition win likely

Lockheed
Missiles & Space Co., Inc.

16 April 1993

J. B. McCurry
205-722-4509



TA-2 Contract Status

Presented to NASA Headquarters

24 May 1993

Lockheed, Aerojet, ECON, Pratt & Whitney

J. B. McCurry
205-722-4509

AGENDA

1. Contract Overview

2. Contract Major Events Chronology

3. Vehicle Assessment Summary
   -- First Lunar Outpost

   -- 50/80K Vehicle Sizing & Performance Assessment

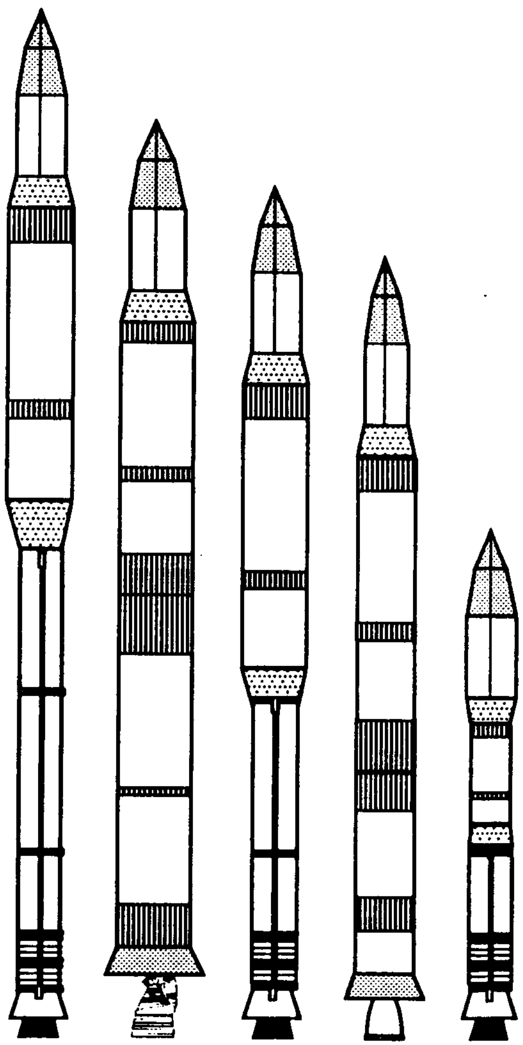
   -- 50/80K Vehicle Cost Assessment

4. Ground Operations Assessment

5. Russian Propulsion Assessment

6. Near-Term Activities

J. B. McCurry
205-722-4509



1. Contract Overview

J. B. McCurry
205-722-4509

TA-2 Charter:

Heavy Lift Launch Vehicle concept definition and analysis to assist NASA in the identification of future launch vehicle requirements

- Vehicle sizing and performance analysis
- Subsystem concept definition
- Propulsion definition (foreign and domestic)
- Ground operations and facilities analysis
- DDT&E and production cost estimation

J. B. McCurry
205-722-4509

Contract Overview

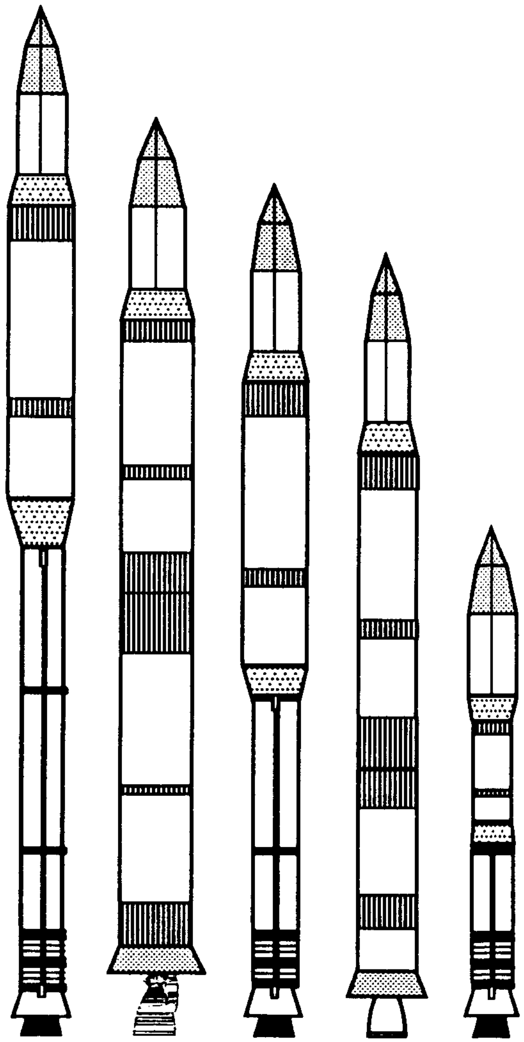
- Originally one year firm-fixed-price with two one-year options
  - -- Basic contract $925,717, duration 5/15/92-5/14/93
  - -- LMSC prime with support from LSOC, Aerojet, and ECON

- Two contract change orders executed to date
  - -- C. O. 1 ($150K) for additional hybrid motor definition (Aerojet) and new Pratt & Whitney subcontract for preliminary technical and cost assessments of Russian booster engines (RD-170 and RD-701); duration 2/26/93-5/14/93
  - -- C. O. 2 ($496K from SDIO) for additional technology assessment, test program requirements planning, and cost definition of RD-170 and RD-701 by Pratt; duration 5/14/93-12/31/93

- Option 1 SOW is to be added to Basic contract as C. O. 3 ($678K)
  - -- Replaces original Basic SOW and C. O. 1 tasks for additional launch vehicle definition and analysis
  - -- Funds LMSC, LSOC, Aerojet, and Pratt, with no-cost extension to ECON
  - -- Duration 5/15/93-12/31/93

- Option 2 not being exercised by MSFC

J. B. McCurry
205-722-4509



2. Contract Major Events Chronology

J. B. McCurry
205-722-4509

Contract Major Events Chronology

Jan.-May 1992: First Lunar Outpost (FLO) Support (via NLS NRA)

- NLS-derived parallel burn HLLV concept sizing and performance assessments
- FLO Technical Interchange Meeting support

June-Sept. 1992: FLO Support (via TA-2 Contract)

- FLO HLLV Team Preliminary Design Status report editing
- HLLV design goals identification and ranking via concurrent engineering process for min. DDT&E cost , min. recurring cost, and min. risk scenarios
- HLLV configuration identification and sizing for min. DDT&E scenario
- FLO HLLV tower drift requirements assessment (nominal and dispersed)

J. B McCurry
205-722-4509

Contract Major Events Chronology

June-Sept. 1992: FLO Support

- First-order HLLV launch site evaluation
- Ground op.s assessments of mixed fleet architectures supporting Red/Blue teams and SSF-assembly "Super Red Team"
- HLLV propulsion requirements identification at Propulsion Synergy Group quality function deployment meeting
- FLO TIM support (monthly)

Oct.-Dec. 1992: FLO Support

- Early HLLV derived parallel-burn and series-burn HLLV configuration and ground op.s assessments
- Alternative HLLV structural/manufacturing design concept assessments
- FLO TIM support

J. B McCurry
205-722-4509

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

Contract Major Events Chronology

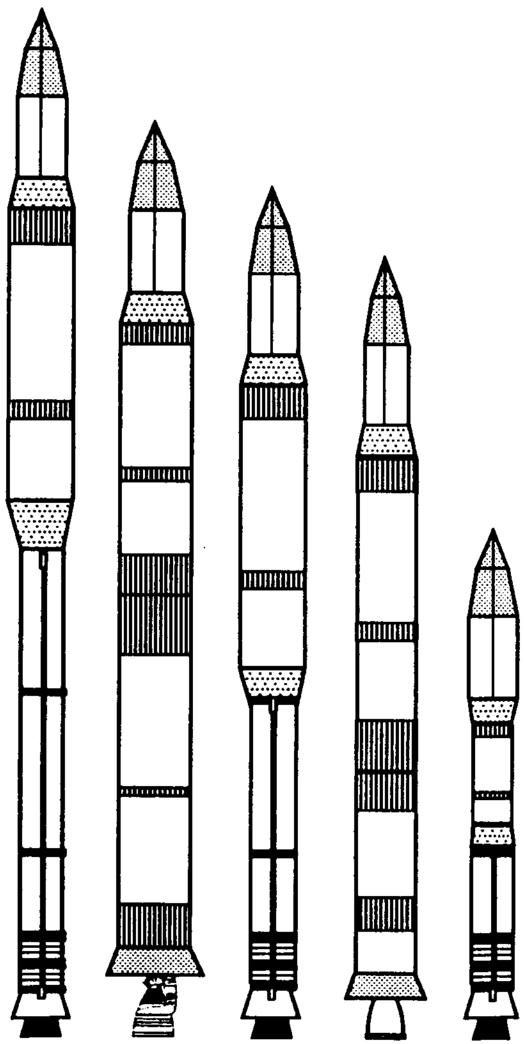
<u>Jan-Mar. 1993</u>: FLO Support

- Liquid and hybrid 50+K two-stage concept definition and assessment for evolution into FLO HLLV strap-on boosters
- Ground op.s assessments of mixed fleet architectures supporting SSF-assembly "Super Red Team" and multiple-booster FLO HLLV configurations (8 F-1A boosters vs. 7 RD-170 boosters)
- Integrated HLLV vehicle health management requirements identification supporting enhanced manufacturing and operability

<u>April-May 1993</u>: Access-to-Space Support

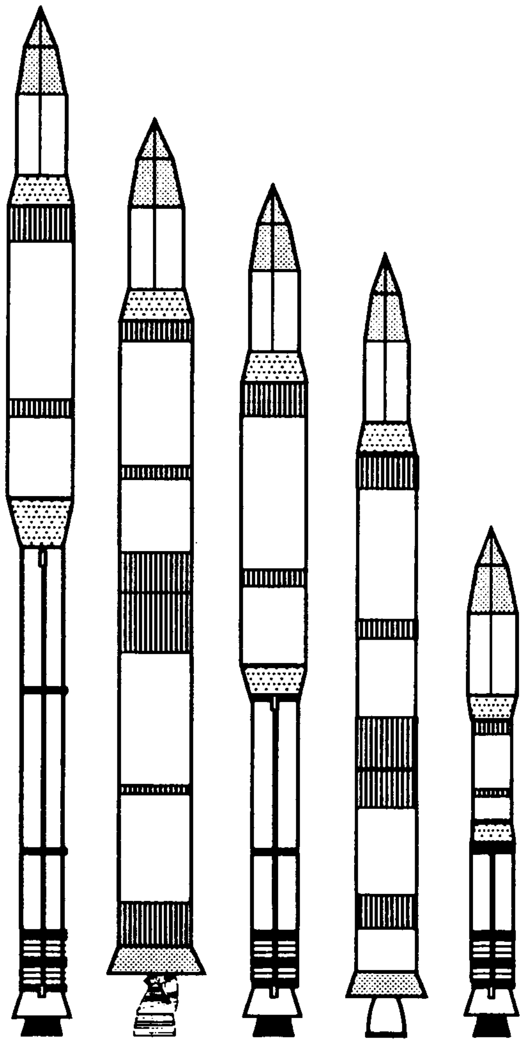
- Liquid, hybrid, solid two-stage concept definition and assessment
- Russian propulsion preliminary assessment
- Mixed fleet ground operations assessments

J. B. McCurry
205-722-4509



3. Vehicle Assessment Summary

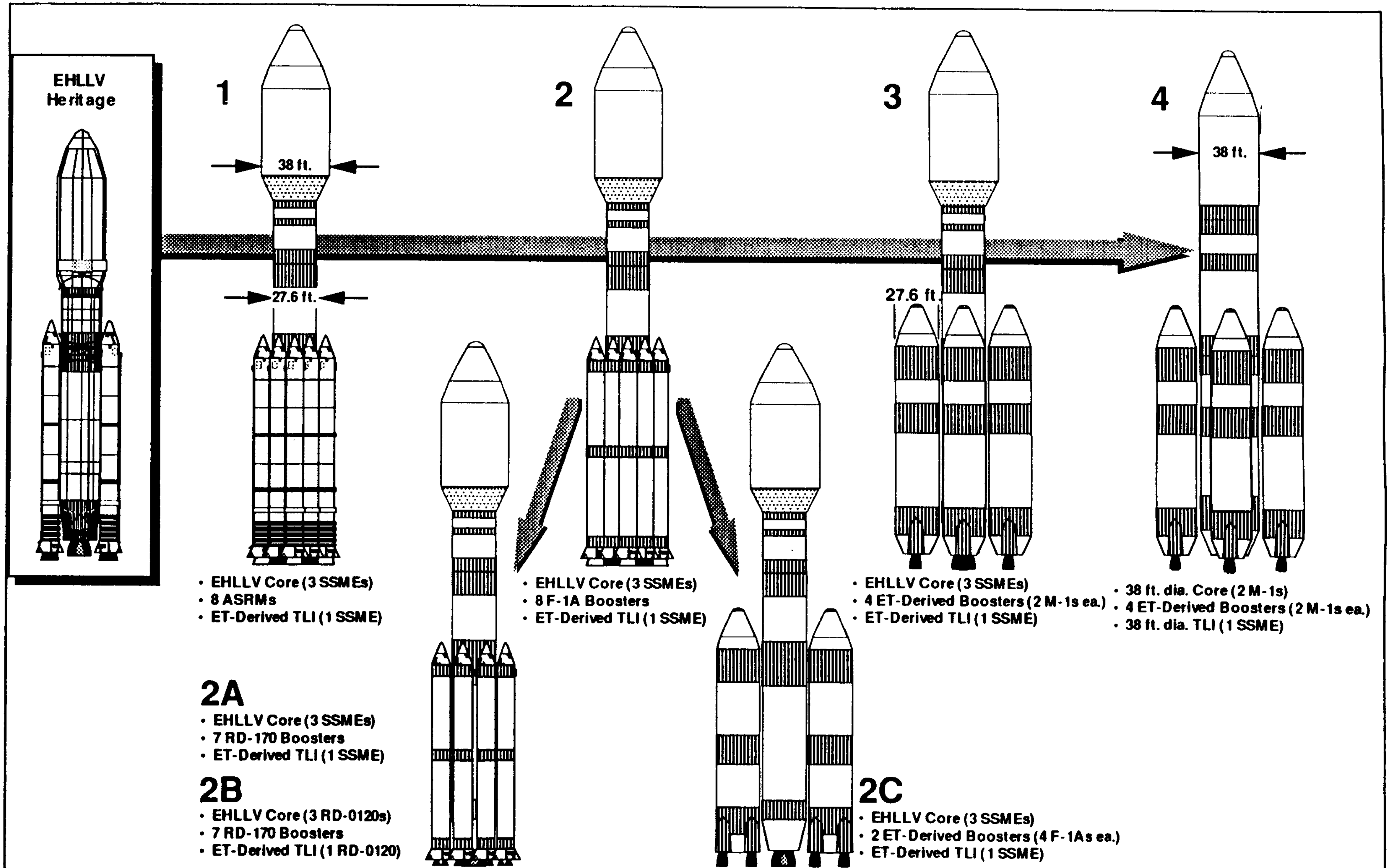
J. B. McCurry
205-722-4509



First Lunar Outpost

J. B. McCurry
205-722-4509

Lunar Mission, Parallel-Burn Configuration Options

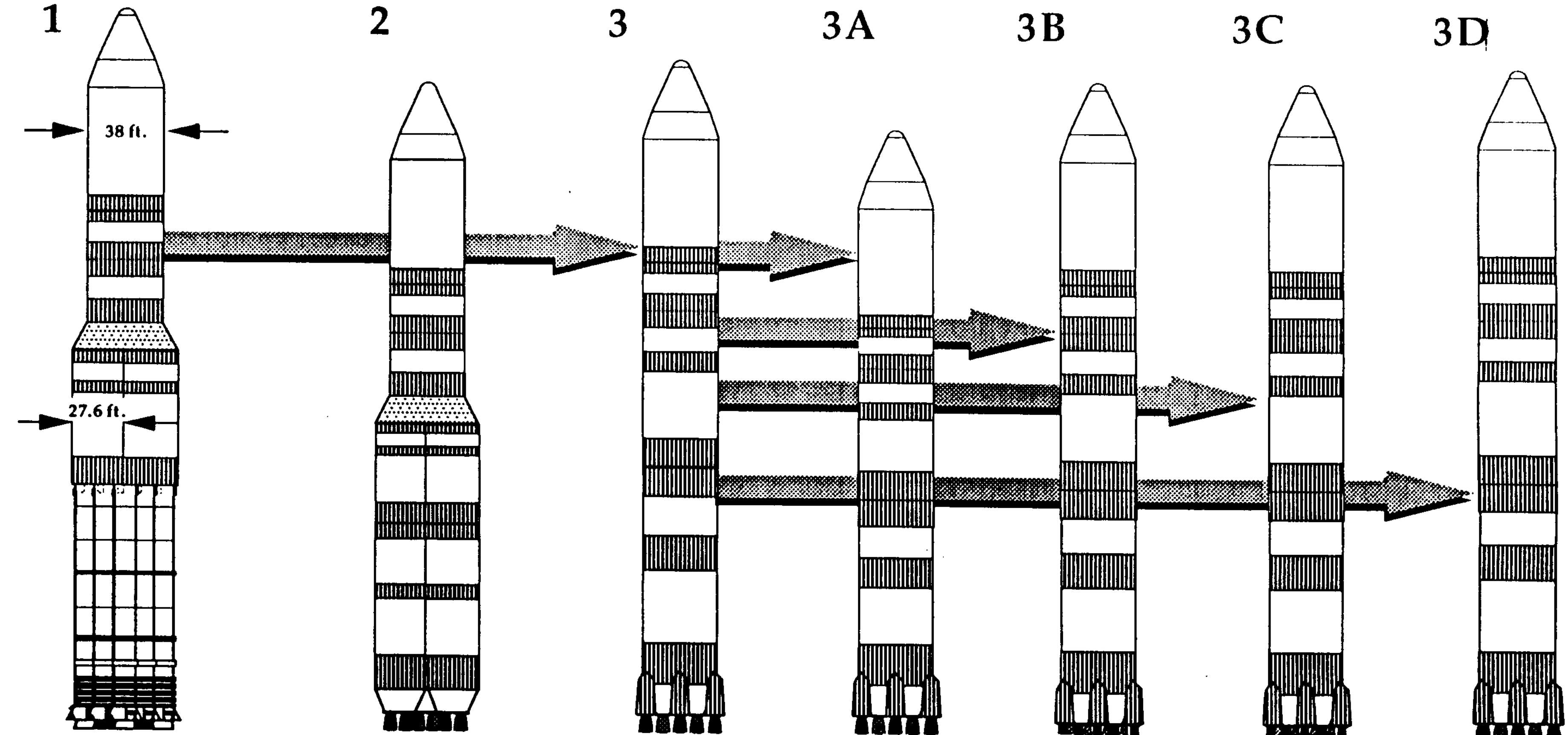


J. B. McCurry
205-722-4509

ATSS TEAM
HEAVY LIFT LAUNCH VEHICLE

ATSS TA-2 CONTRACT STATUS: MAY 1993

Lunar Mission
Series-Burn Configuration Options



1
- TBD ASRMs First Stage
- 3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
- 38 ft. Dia. 3rd Stage (1 SSME)

2
- 3 ET-Derived Tanks First Stage (3 F-1As ea.)
- 3 ET-Derived Tanks 2nd Stage (2 SSMEs ea.)
- 38 ft. Dia. 3rd Stage (1 SSME)

3
- 38 ft. Dia. 1st Stage (7 F-1As)
- 38 ft. Dia. 2nd Stage (5 SSMEs)
- 38 ft. Dia. TLI Stage (1 SSME)

3A
- 38 ft. Dia. 1st Stage Multi-Cell (7 F-1As)
- 38 ft. Dia. 2nd Stage Multi-Cell (5 SSMEs)
- 38 ft. Dia. TLI Stage Multi-Cell (1 SSME)

3B
- 38 ft. Dia. 1st Stage (7 RD-170s)
- 38 ft. Dia. 2nd Stage (5 SSMEs)
- 38 ft. Dia. TLI Stage (1 SSME)

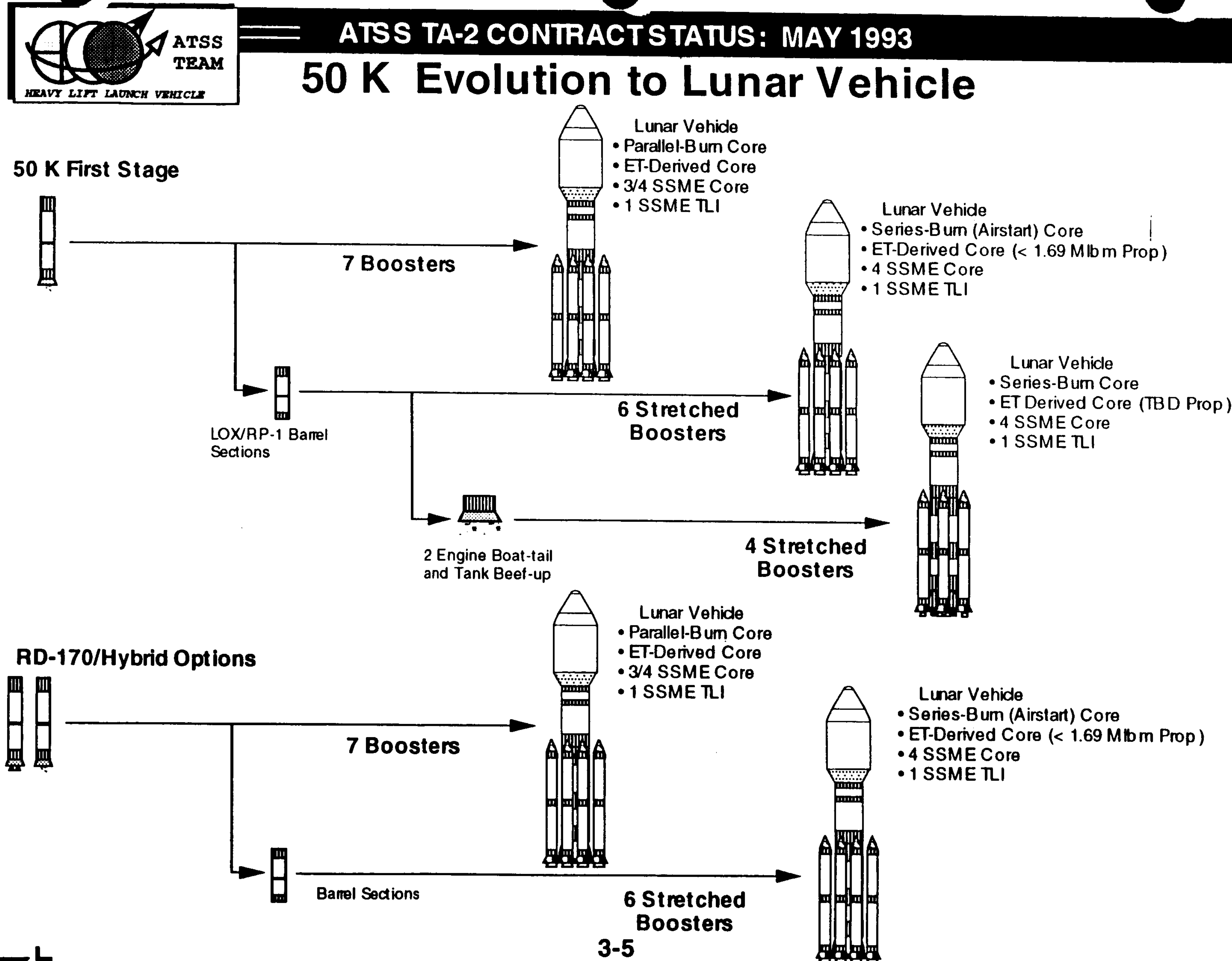
3C
- 38 ft. Dia. 1st Stage (7 RD-170s)
- 38 ft. Dia. 2nd Stage (5 RD-0120s)
- 38 ft. Dia. TLI Stage (1 RD-0120)

3D
- 38 ft. Dia. 1st Stage (TBD M-1s)
- 38 ft. Dia. 2nd Stage (TBD M-1s)
- 38 ft. Dia. TLI Stage (1 SSME)

Lockheed, Aerojet, ECON, Pratt & Whitney

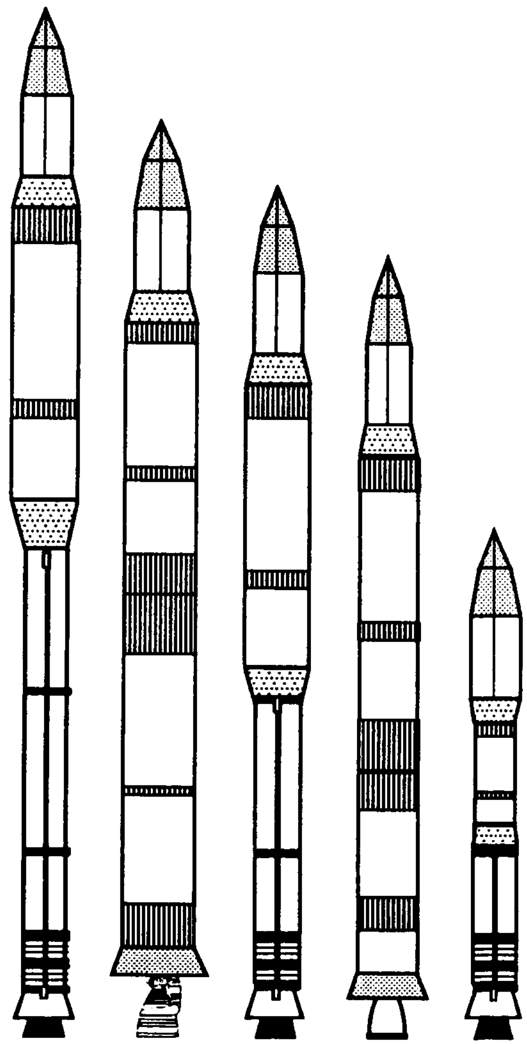
J. B. McCurry
205-722-4509

50 K Evolution to Lunar Vehicle



Lockheed, Aerojet, ECON, Pratt & Whitney

J. B McCurry
205-722-4509



50/80K Vehicle Sizing and Performance Assessment

J. B. McCurry
205-722-4509

Vehicle Configuration Matrix

First Stage/Second Stage Options		
Liquid/Liquid *	Hybrid/Liquid *	Solid/Liquid
F-1A/LCSSME F-1A/J-2S F-1A/SSME F-1A/RD-0120 F-1A/Vulcain STME/LCSSME STME/STME STME/RD-0120 STME/Vulcain M-1A/LCSSME M-1A/RD-0120 M-1A/Vulcain RD-170/LCSSME RD-170/J-2S RD-170/RD-0120 RD-170/Vulcain LCSSME/LCSSME LCSSME/RD-0120 LCSSME/Vulcain	Staged Combustion Hybrid/LCSSME Staged Combustion Hybrid/Rubber STME Staged Combustion Hybrid/J-2S Staged Combustion Hybrid/Vulcain Staged Combustion Hybrid/RD-0120 Classical Hybrid/LCSSME Classical Hybrid/Rubber STME Classical Hybrid/J-2S Classical Hybrid/Vulcain Classical Hybrid/RD-0120	3 Segment ASRM/LCSSME 3 Segment ASRM/LCSSME 2 Segment ASRM/J-2S 2 Segment ASRM/LCSSME 2 Segment ASRM/SSME 1 Segment ASRM/Centaur

Note: * Configurations sized for 50 Klbm payload

J. B McCurry
205-722-4509

Vehicle Configuration Payload

Liquid/Liquid *	Payload (lbm) **
F-1A/LCSSME	48,249
F-1A/J-2S	54,893
F-1A/SSME	51,098
F-1A/RD-0120	48,599
F-1A/Vulcain	49,155
STME/LCSSME	48,321
STME/STME	48,034
STME/RD-0120	50,186
STME/Vulcain	49,986
M-1A/LCSSME	47,992
M-1A/RD-0120	49,471
M-1A/Vulcain	48,993
RD-170/LCSSME	49,878
RD-170/J-2S	50,166
RD-170/RD-0120	48,598
RD-170/Vulcain	50,598
LCSSME/LCSSME	48,222
LCSSME/RD-0120	49,339
LCSSME/Vulcain	49,071

Note: * First Stage/Second Stage Propulsion Options
** Payloads verified by 3-DOF trajectory analysis

J. B McCurry
205-722-4509

Vehicle Configuration Payload (Concluded)

Hybrid/Liquid *	Payload (lbm) **
Staged Combustion Hybrid/LCSSME	51,773
Staged Combustion Hybrid/Rubber STME	54,836
Staged Combustion Hybrid/J-2S	50,610
Staged Combustion Hybrid/Vulcain	50,072
Staged Combustion Hybrid/RD-0120	51,354
Classical Hybrid/LCSSME	51,663
Classical Hybrid/Rubber STME	54,987
Classical Hybrid/J-2S	50,559
Classical Hybrid/Vulcain	49,962
Classical Hybrid/RD-0120	51,265

Solid/Liquid *	Payload (lbm) **
3 Segment ASRM/LCSSME	65,000
3 Segment ASRM/SSME	82,100
2 Segment ASRM/J-2S	43,600
2 Segment ASRM/LCSSME	49,300
2 Segment ASRM/SSME	56,600
1 Segment ASRM/Centaur	6,900

Note: * First Stage/Second Stage Propulsion Options
** Payloads verified by 3-DOF trajectory analysis

J. B. McCurry
205-722-4509

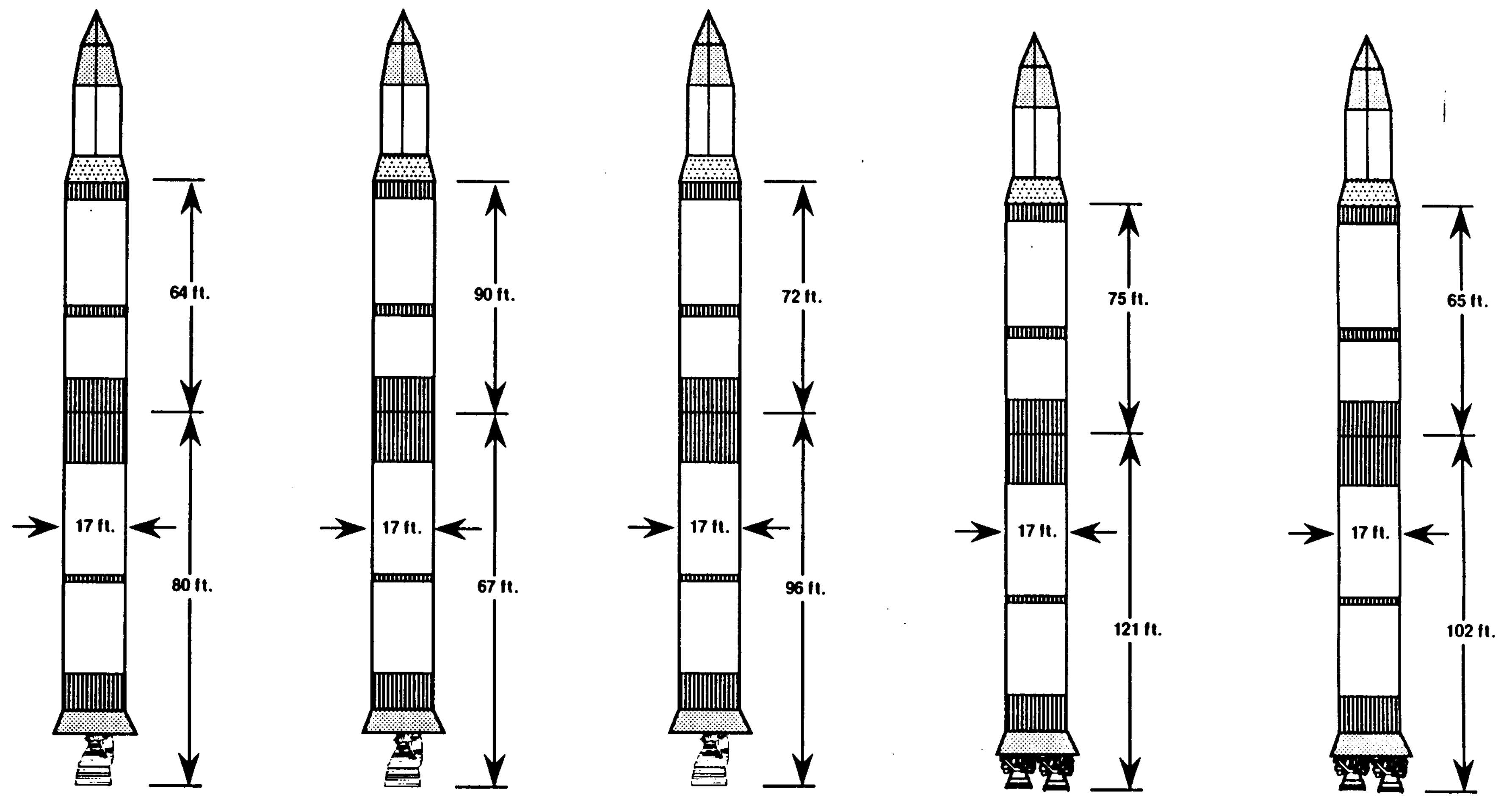
Engine Specifications

	M-1A	F-1A	STME	SSME (104% RPL)	RD-170
Sea Level Thrust (lbf)	1,300,000	1,800,000	551,430	390,000	1,632,000
Vacuum Thrust (lbf)	1,562,000	2,020,700	650,000	488,800	1,777,000
Sea Level Specific Impulse (sec)	344.5	269.7	364	364.8	309
Vacuum Specific Impulse (sec)	414.0	303.1	428.5	452.9	337
Chamber Pressure (psia)	1,000	1,161	2,250	3,110	3,560
Mixture Ratio	5.0	2.27	6.0	6.0	2.6
Area Ratio	20	16	45	77.5	36.87
Engine Mass (lbm)	20,200	19,000	9,974	6,990	21,510
Engine Length (ft)	19.08	18.36	13	14	13.12
Engine Diameter (ft)	12.58	11.96	12.1	8	12.20
Propellant	O2/H2	O2/RP-1	O2/H2	O2/H2	O2/Syn10

	RD-0120	Vulcain	J-2S	LCSSME (Altitude)	LCSSME (Sea Level)
Sea Level Thrust (lbf)	352,746	---	197,000	---	506,000
Vacuum Thrust (lbf)	440,925	230,000	265,000	326,600	600,000
Sea Level Specific Impulse (sec)	364	---	320	---	371
Vacuum Specific Impulse (sec)	455	431.6	436	451.9	440
Chamber Pressure (psia)	3,000	1,450	1,200	2,075	2,075
Mixture Ratio	6.0	5.2	5.5	6.0	6.0
Area Ratio	85.7	45	40	77.5	42
Engine Mass (lbm)	7,607	2,860	3,800	7,053	7,300
Engine Length (ft)	14.93	9.62	11.08	14	14
Engine Diameter (ft)	7.94	5.77	6.71	8	8
Propellant	O2/H2	O2/H2	O2/H2	O2/H2	O2/H2

J. B. McCurry
205-722-4509

Examples of All-Liquid 50K Concepts



Length/Diameter= 8.5 **F-1A/LCSSME**

Length/Diameter= 9.2 **F-1A/SSME**

Length/Diameter= 9.9 **F-1A/J-2S**

Length/Diameter= 11.5 **STME/STME**

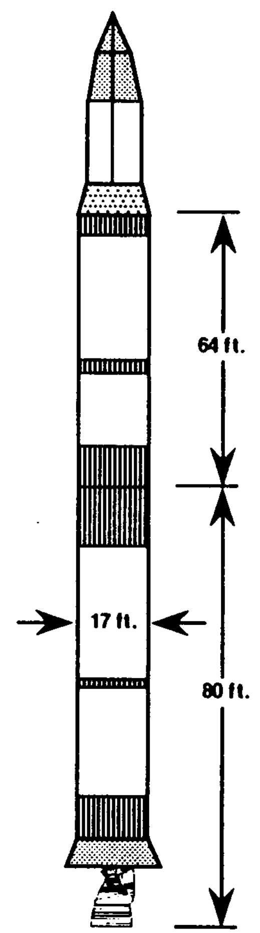
Length/Diameter= 9.8 **LCSSME/LCSSME**

(Vehicles not to scale)

Lockheed, Aerojet, ECON, Pratt & Whitney

J. B. McCurry
205-722-4509

50 k Vehicle, F-1A/ LCSSME



Length/Diameter= 8.5

Payload:	48,249 lbm (21.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 996,632 lbm

First Stage:

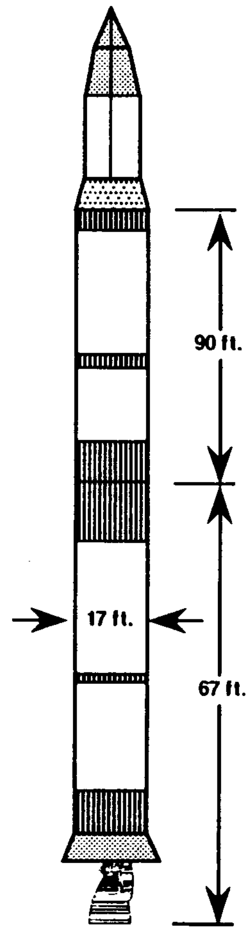
Inert Mass:	69,190 lbm
Usable Propellant:	665,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	81.8 %

Second Stage:

Inert Mass:	24,193 lbm
Usable Propellant:	190,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.236 g

J. B McCurry
205-722-4509

50 k Vehicle, F-1A/ SSME



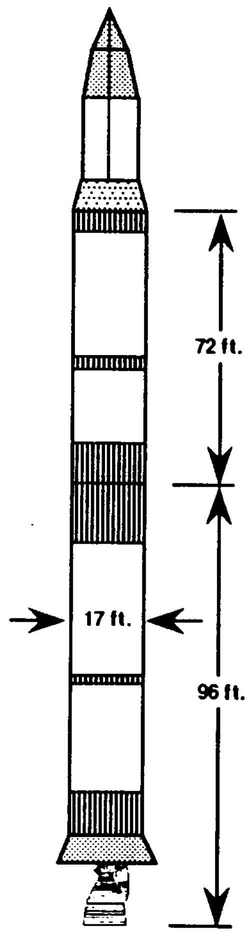
Length/Diameter= 9.2

Payload:	51,098 lbm (23.2 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	940,454 lbm
First Stage:	
Inert Mass:	64,841 lbm
Usable Propellant:	478,000 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	77.0 %
Second Stage:	
Inert Mass:	31,515 lbm
Usable Propellant:	315,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	SSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.233 g

J. B. McCurry
205-722-4509

50 k Vehicle, F-1A/ J-2S



Length/Diameter= 9.9

Payload:	54,893 lbm (24.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,177,785 lbm

First Stage:

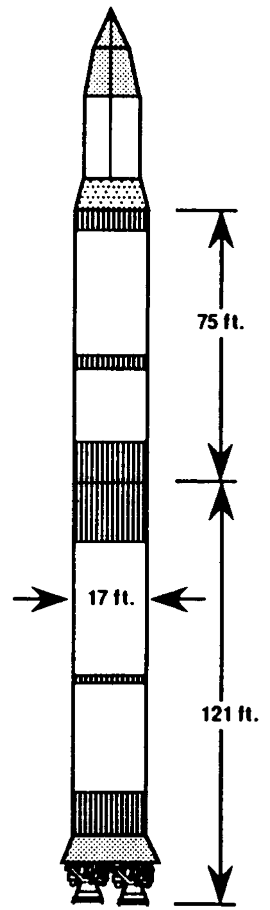
Inert Mass:	68,458 lbm
Usable Propellant:	819,635 lbm
Propellant Type:	LOX/RP-1
Engine Type/No.:	F-1A/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Throttle Setting:	96.1 %

Second Stage:

Inert Mass:	24,881 lbm
Usable Propellant:	210,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	17.0 ft
Thrust/Weight:	0.930 g

J. B. McCurry
205-722-4509

50 k Vehicle, STME/ STME



Length/Diameter= 11.5

Payload:	48,034 lbm (21.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW:	865,286 lbm

First Stage:

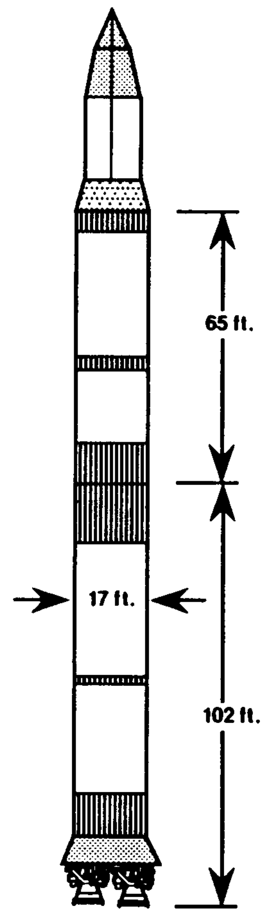
Inert Mass:	73,559 lbm
Usable Propellant:	460,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/2
Diameter:	17.0 ft
Thrust/Weight:	1.272 g
Throttle Setting:	100.0 %

Second Stage:

Inert Mass:	33,693 lbm
Usable Propellant:	250,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	STME/1
Diameter:	17.0 ft
Thrust/Weight:	1.422 g
Throttle Setting	73.0 %

J. B McCurry
205-722-4509

50 k Vehicle, LCSSME/ LCSSME



Length/Diameter= 9.8

Payload:	48,222 lbm (21.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 692,379 lbm

First Stage:

Inert Mass:	59,920 lbm
Usable Propellant:	365,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/2
Diameter:	17.0 ft
Thrust/Weight:	1.458 g
Throttle Setting:	100.0 %

Second Stage:

Inert Mass:	24,237 lbm
Usable Propellant:	195,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	1.213 g

J. B. McCurry
205-722-4509

Hybrid Booster Concepts

Performance and Sizing Groundrules and Assumptions

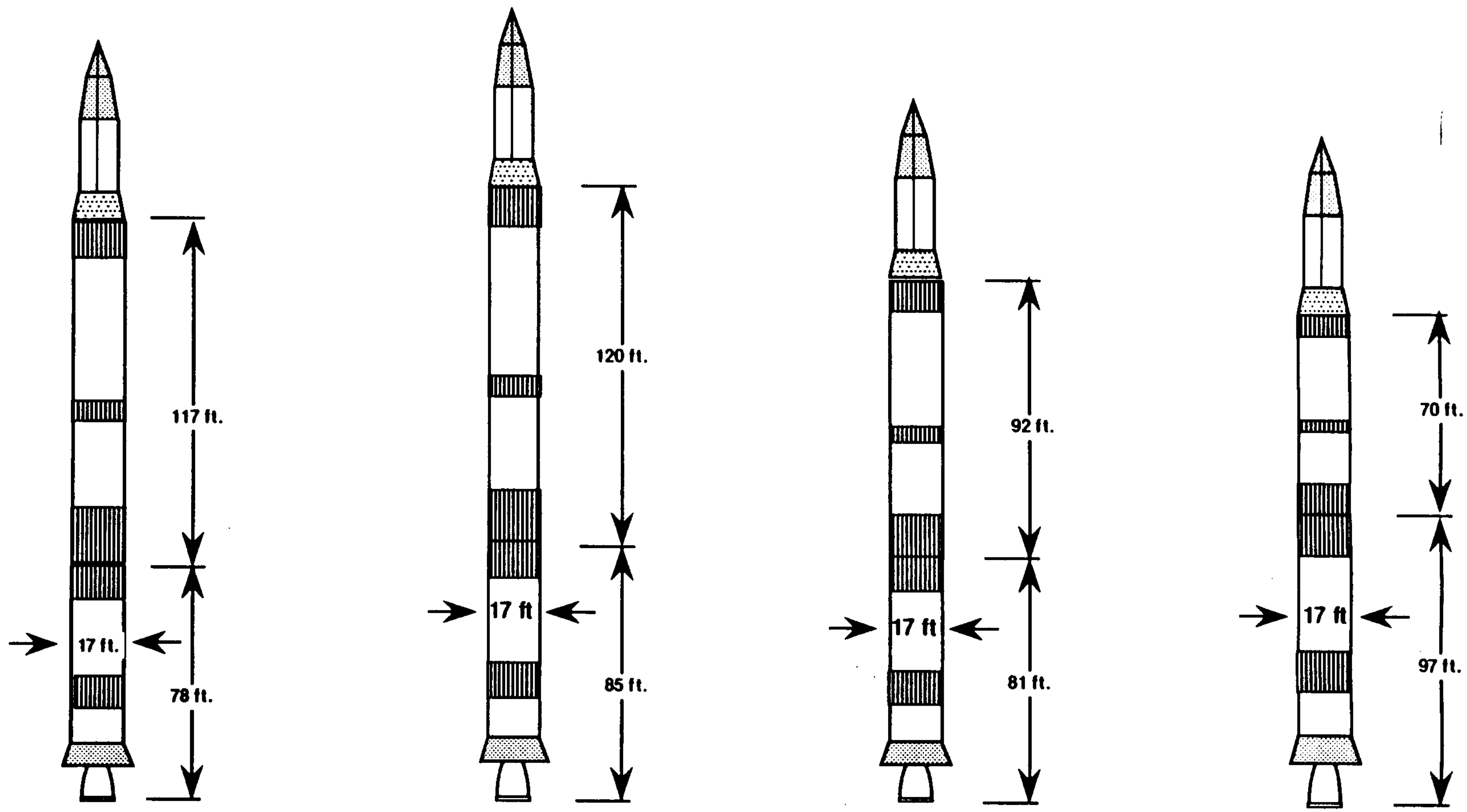
- Staged Combustion Concepts Only
  - -- Oxidizer Type: Ammonium Perchlorate (AP)
  - -- Fuel Grain Oxidizer Content: overall (Booster) MR is 2.5; the 2.5 is split 1.9 as LOX and the remaining 0.6 is AP within the grain

- Staged Combustion and Classical Concepts:
  - -- Grain Ignition Method: redundant, forward end, tri-ethyl aluminum
  - -- Igniter Weight: 50 lbf
  - -- Thrust Termination Method: termination of LOX flow
  - -- LOX Tank Pressurization Method: autogenous, warm GOX, turbine exhaust
  - -- LOX Tank Ullage Pressure: 60 psia
  - -- Motor Chamber Pressure: 1700 psia
  - -- LOX Injector Inlet Pressure: 2000 psia
  - -- Thrust Chamber Cooling Method: regenerative
  - -- TVC Method: electro-mechanical actuators
  - -- Minimum Throttle Setting: 75 % Rated Power Level

- Residuals
  - -- 2 % of solid propellant load for staged combustion
  - -- 10 % of solid propellant load for classical combustion

J. B. McCurry
205-722-4509

Examples of Hybrid/Liquid 50K Concepts



Length/Diameter= 11.5
Staged Comb./ Rubber STME

Length/Diameter= 12.0
Classical/ Rubber STME

Length/Diameter= 10.1
Classical/ LCSSME

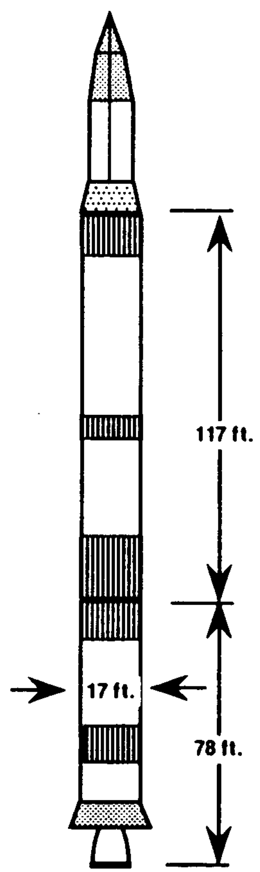
Length/Diameter= 9.8
Classical/ Vulcain

(Vehicles not to scale)

Lockheed, Aerojet, ECON, Pratt & Whitney

J. B. McCurry
205-722-4509

50 k Vehicle, Staged Combustion Hybrid/Rubber STME



Length/Diameter= 11.5

Payload:	54,836 lbm (24.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,302,605 lbm

First Stage:

Inert Mass:	95,402 lbm
Usable Propellant:	620,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.443 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	100.0 %

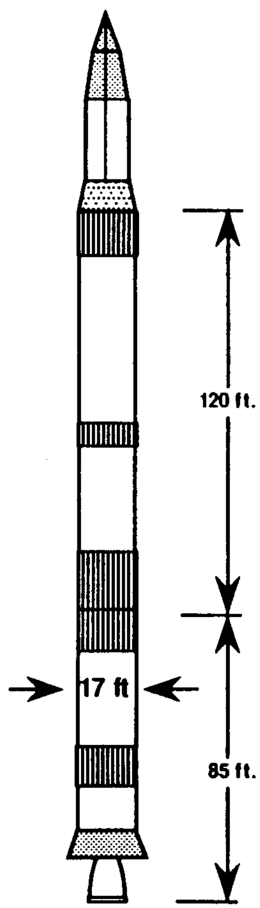
Second Stage:

Inert Mass:	36,367 lbm
Usable Propellant:	446,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	17.0 ft
Vacuum Thrust	425,894 lbf
Thrust/Weight:	0.800 g

Lockheed, Aerojet, ECON, Pratt & Whitney

J. B. McCurry
205-722-4509

50 k Vehicle, Classical Hybrid/Rubber STME



Length/Diameter= 12.0

Payload:	53,733 lbm (24.4 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,314,534 lbm

First Stage:

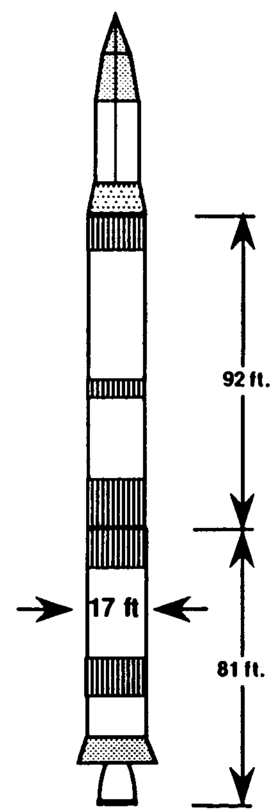
Inert Mass:	115,675 lbm
Usable Propellant:	650,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.386 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	37,126 lbm
Usable Propellant:	458,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	17.0 ft
Vacuum Thrust	436,101 lbf
Thrust/Weight:	0.800 g

J. B. McCurry
205-722-4509

50 k Vehicle, Classical Hybrid/LCSSME



Length/Diameter= 10.1

Payload:	50, 499 lbm (22.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,114,359 lbm

First Stage:

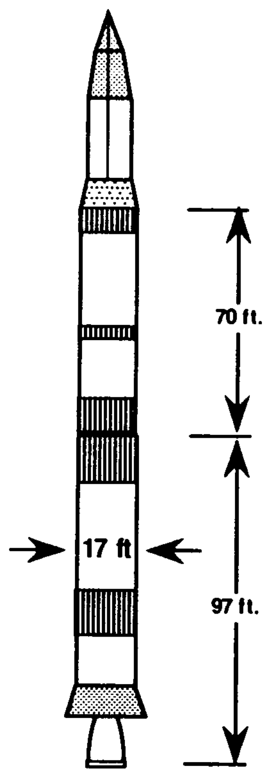
Inert Mass:	108,583 lbm
Usable Propellant:	600,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,630,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	30,277 lbm
Usable Propellant:	325,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft
Thrust/Weight:	0.806 g

J. B. McCurry
205-722-4509

50 k Vehicle, Classical Hybrid/Vulcain



Length/Diameter= 9.8

Payload:	47,887 lbm (21.7 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,207,921 lbm

First Stage:

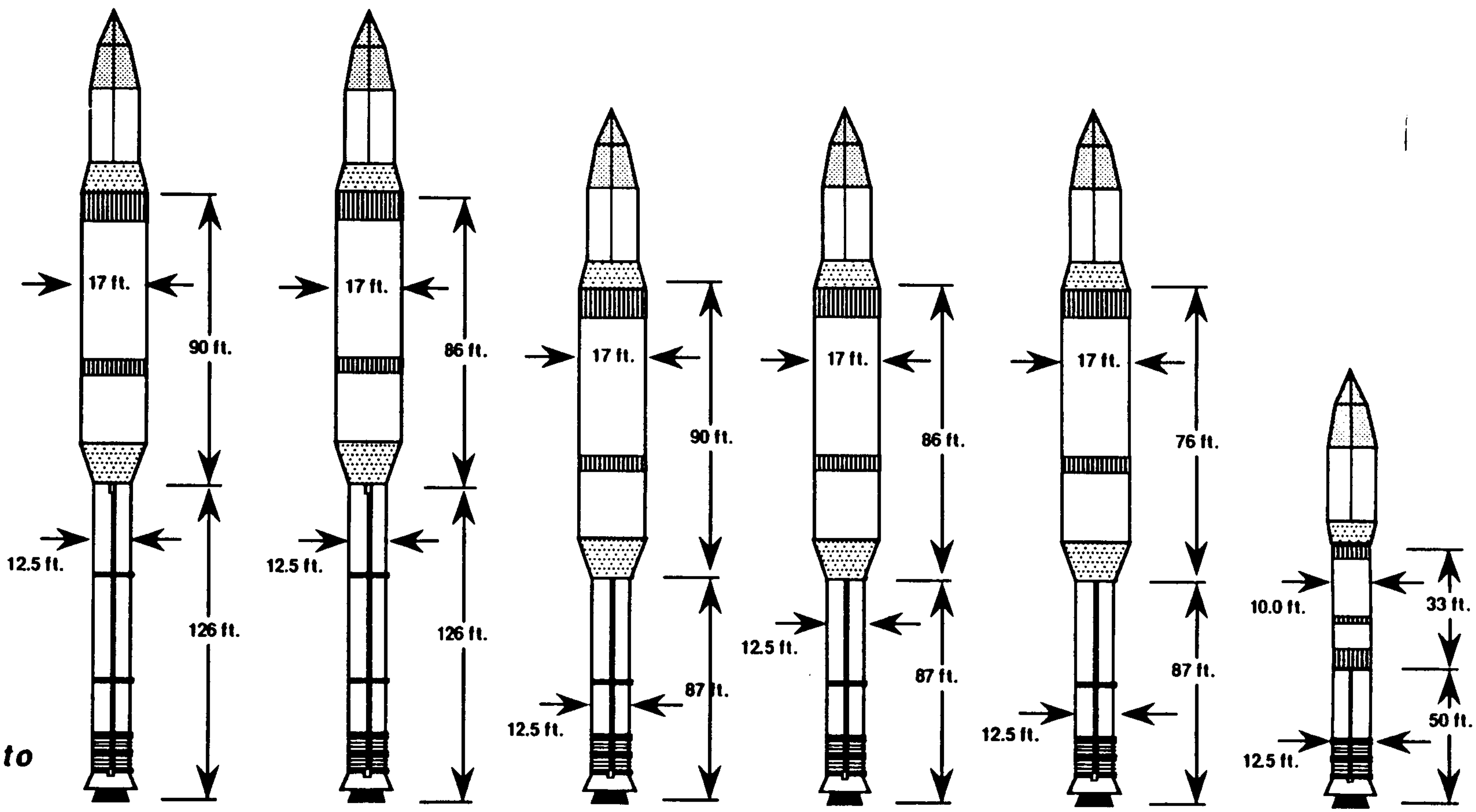
Inert Mass:	129,623 lbm
Usable Propellant:	790,000 lbm
Propellant Type:	LOX/HTDP
Engine Type/No.:	Classical Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.475 g
Sea Level Thrust	1,785,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	20,411 lbm
Usable Propellant:	220,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Vulcain/1
Diameter:	17.0 ft
Thrust/Weight:	0.792 g

J. B. McCurry
205-722-4509

ASRM-Derived Launch Vehicle Family



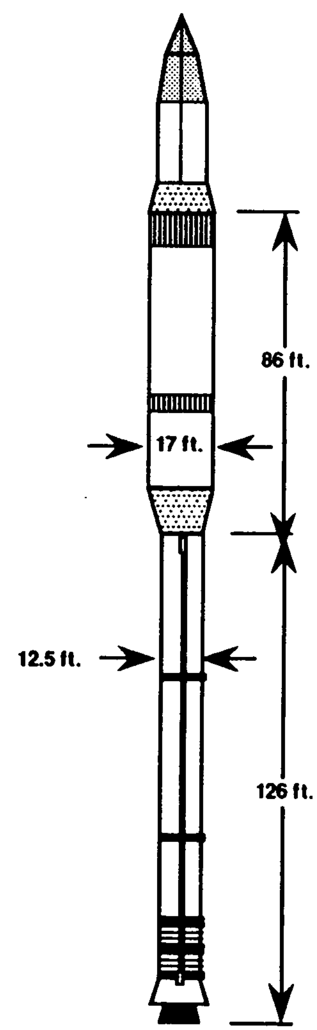
Vehicles not to Scale

1 st Stage	3 Seg ASRM	3 Seg ASRM	2 Seg ASRM	2 Seg ASRM	2 Seg ASRM	1 Seg ASRM
2 nd Stage	1 SSME	1 LCSSME	1 SSME	1 LCSSME	1 J-2s	Centaur
Payload*	82,100 lbm	65,000 lbm	56,600 lbm	49,300 lbm	43,600 lbm	6,900 lbm

Note:
- Direct insertion into 15 x 220 nm, 28.5 deg. orbit

J. B. McCurry
205-722-4509

50 k Vehicle, 3 Segment ASRM/LCSSME



Payload:	65,000 lbm (29.5 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,736,481 lbf

First Stage:

Inert Mass:	179,947 lbm**
Usable Propellant:	1,214,401 lbm
Propellant Type:	HTPB
Engine Type/No.:	3 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.740 g
Sea Level Thrust	3,020,812 lbf

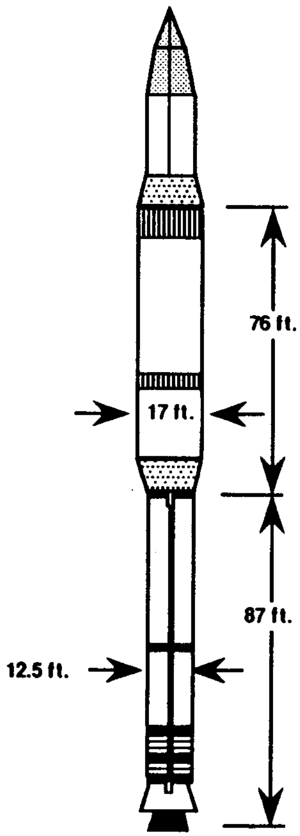
Second Stage:

Inert Mass:	27,133 lbm
Usable Propellant:	250,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	LCSSME/1
Diameter:	17.0 ft*
Thrust/Weight:	0.955 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for best total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 42,300 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses

J. B McCurry
205-722-4509

50 k Vehicle, 2 Segment ASRM/J-2S



Payload:	43,600 lbm (19.8 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,293,267 lbf

First Stage:

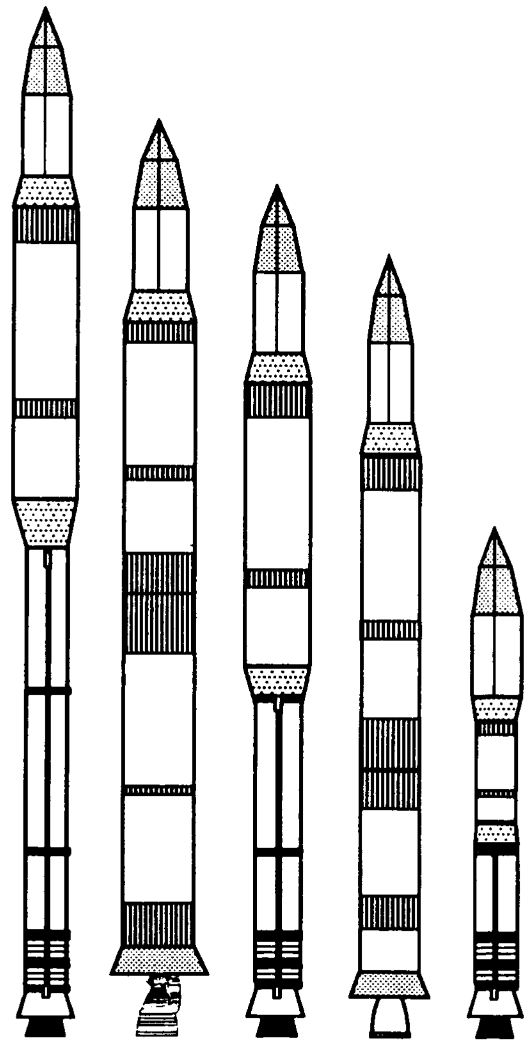
Inert Mass:	158,788 lbm**
Usable Propellant:	807,212 lbm
Propellant Type:	HTPB
Engine Type/No.:	2 Segment ASRM/1
Diameter:	12.5 ft
Thrust/Weight:	1.552 g
Sea Level Thrust	2,007,933 lbf***

Second Stage:

Inert Mass:	23,667 lbm
Usable Propellant:	260,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	J-2S/1
Diameter:	12.5 ft *
Thrust/Weight:	0.998 g
Throttle Setting	100.0 %

Note:
* Diameter not optimized for <u>best</u> total vehicle L/D, inert weight, etc.; current diameter results in acceptable L/D
** First stage contains 40,500 lbm in excess of motor mass, this excess mass is used for extra booster stiffness and interstage masses
*** Thrust profile assumed to be a ratio of segment propellant loads

J. B McCurry
205-722-4509



50/80K Vehicle Cost Assessment

Lockheed, Aerojet, ECON, Pratt & Whitney

J. B McCurry
205-722-4509

Groundrules & Assumptions

- All costs presented in FY 92 $s
- NASA Code B new start escalation table used to normalize $s
- Current estimated costs include DDT&E and production
- ECON's weight-based cost estimating relationships utilized with various complexity factors
  -- Cost algorithms have been calibrated against MSFC Engineering Cost Group over the last several years
- Subsystem weights based on mass properties supplied by LMSC's vehicle sizing tools
- Weights included 10% contingency allocated to subsystems
- Mission model for 50K vehicle based on SSP/PLS mixed fleet model supplied by G. Austin/MSFC-PT01
  -- Total of 101 50K vehicle flights over 2003-2010 time horizon
- With exception of engines, all subsystems assumed 2 equivalent test articles

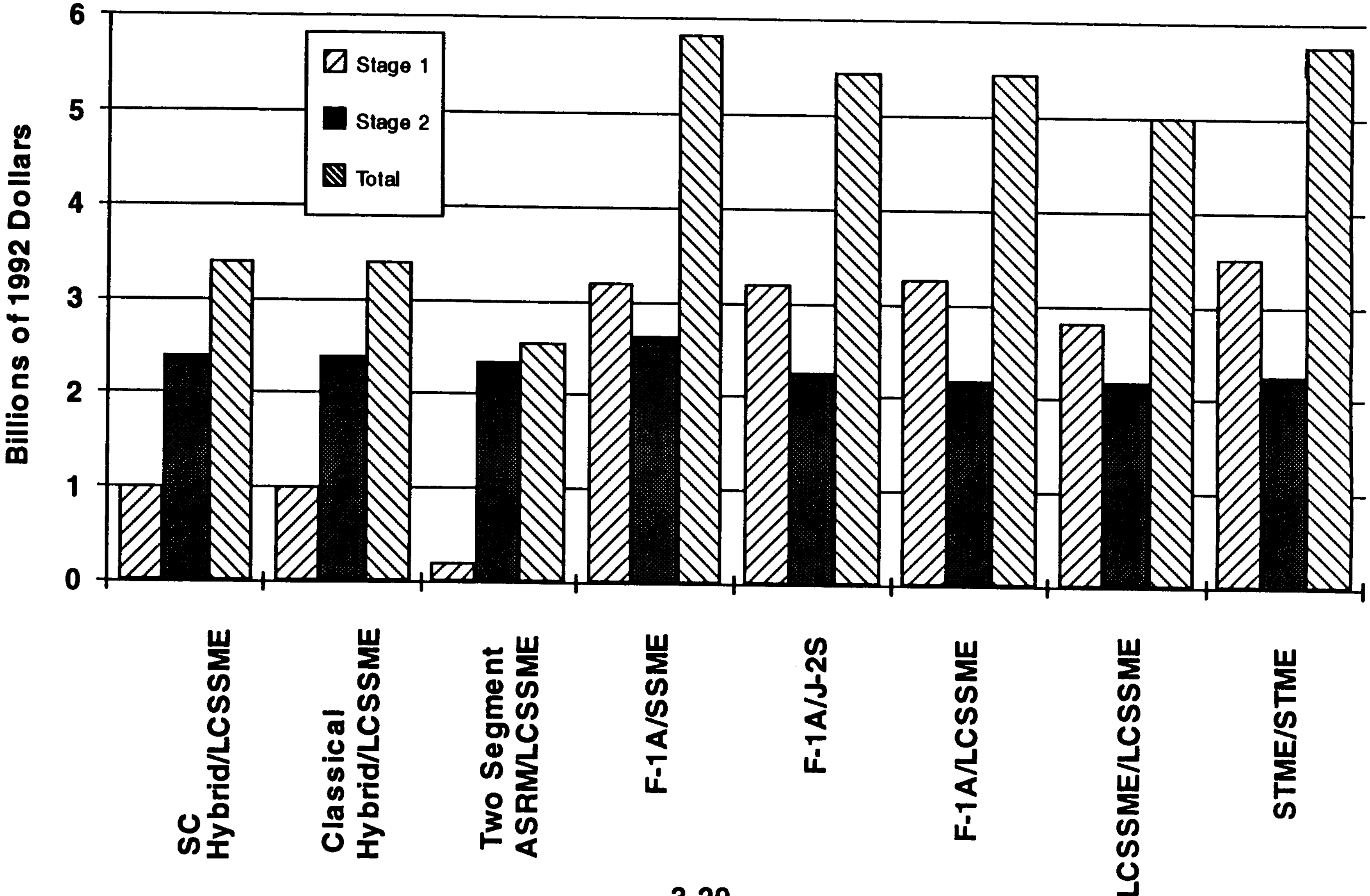
J. B. McCurry
205-722-4509

Groundrules & Assumptions (Concluded)

- All main propulsion cost data were throughputs for the estimate, no independent estimates of production costs were conducted
  - --Hybrid and ASRM data supplied by Aerojet
  - --SSME and LCSSME data supplied by Rocketdyne
  - --STME, F-1A, and J-2S data supplied by MSFC Engineering Cost Group
- No schedule impact assessed in costing
- State-of-the-art ranking assumed to be new drawings with known point-of-departure
  - -- Engines assume most drawings exist
- Specification level set at manned space due to PLS Mission
- No government "wraps" included (40% typically used)

J. B. McCurry
205-722-4509

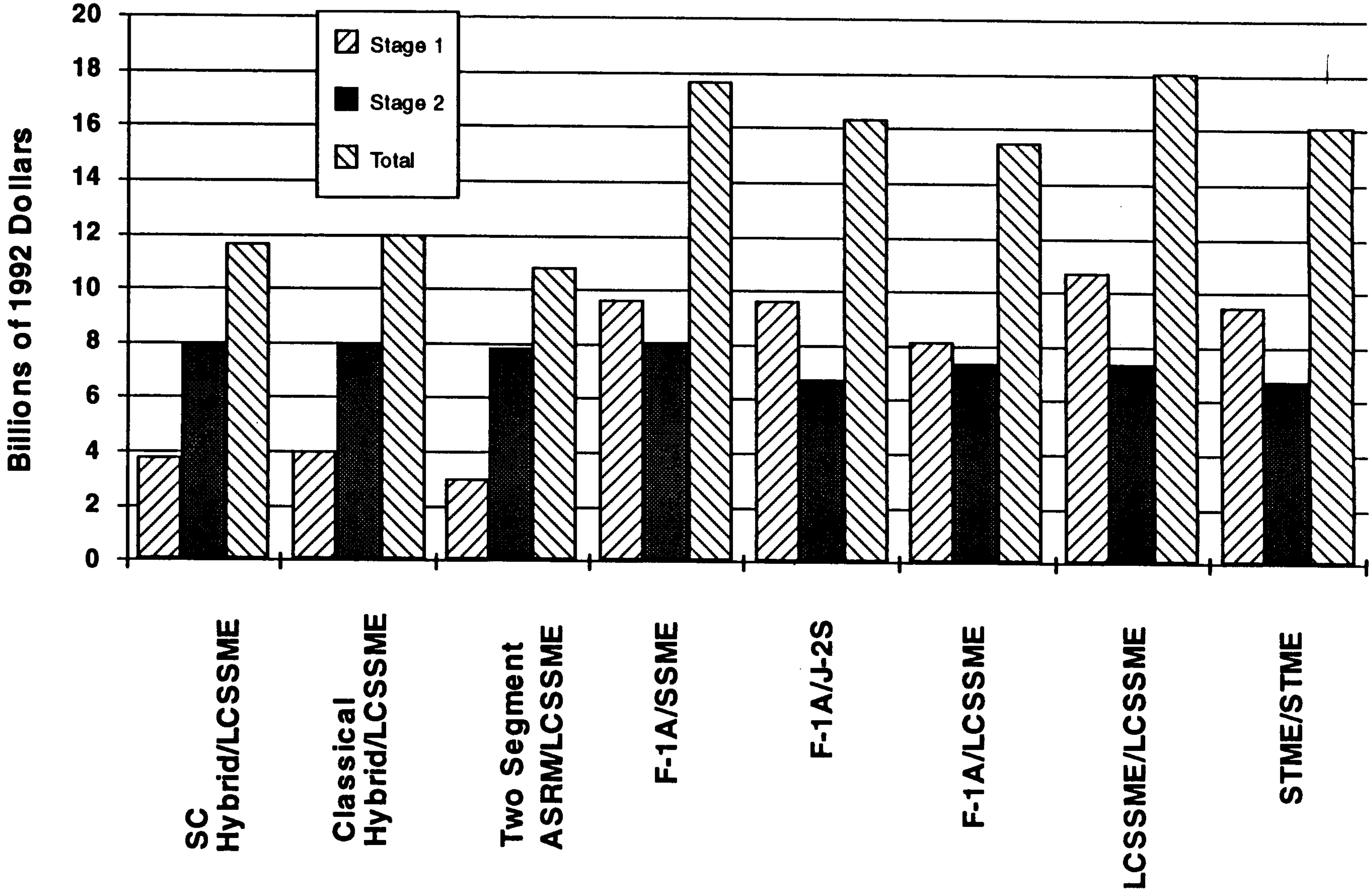
Program Development Costs by Stage



J. B. McCurry
205-722-4509

ATSS TA-2 CONTRACT STATUS: MAY 1993

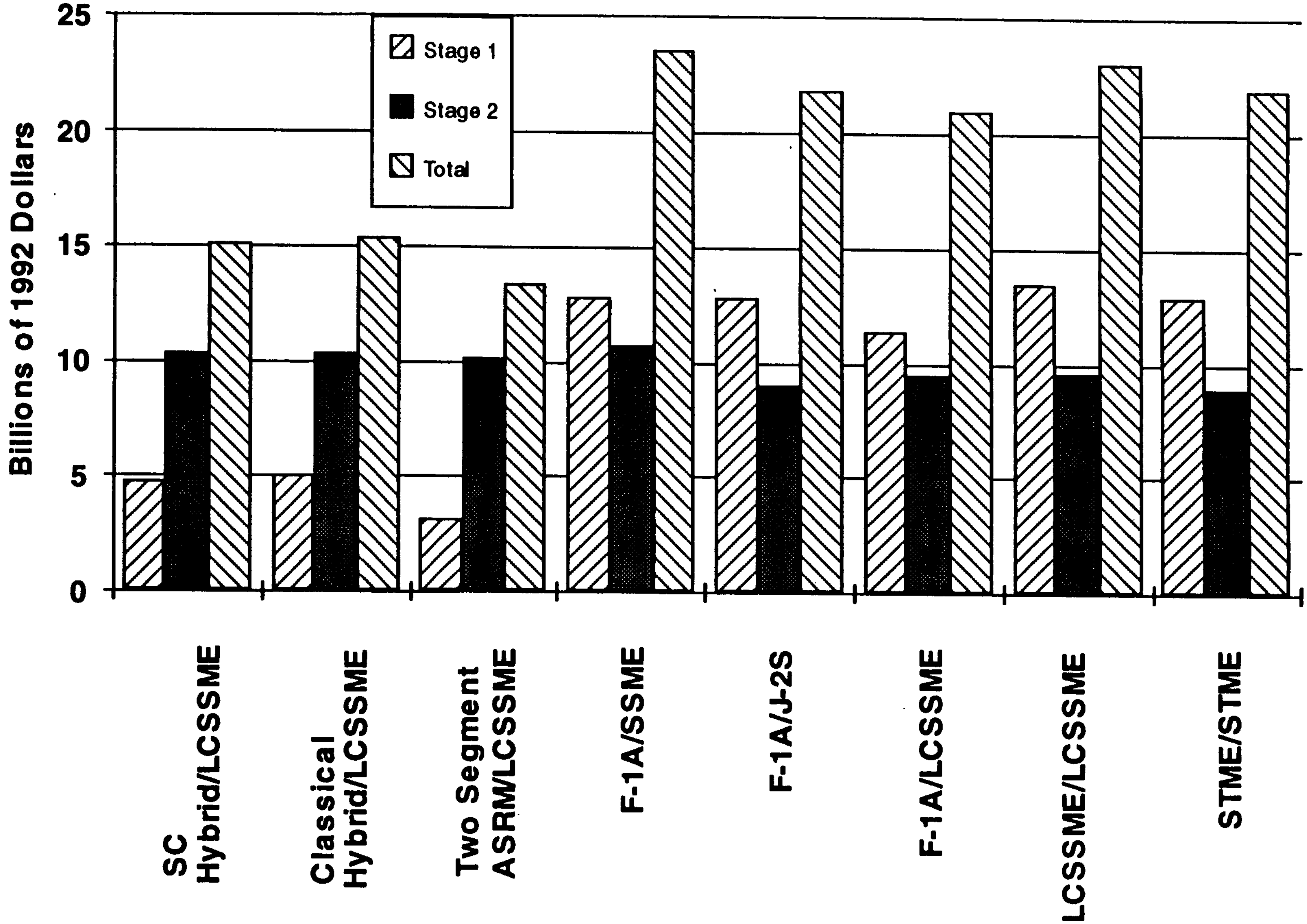
Program Production Costs by Stage



Lockheed, Aerojet, ECON, Pratt & Whitney

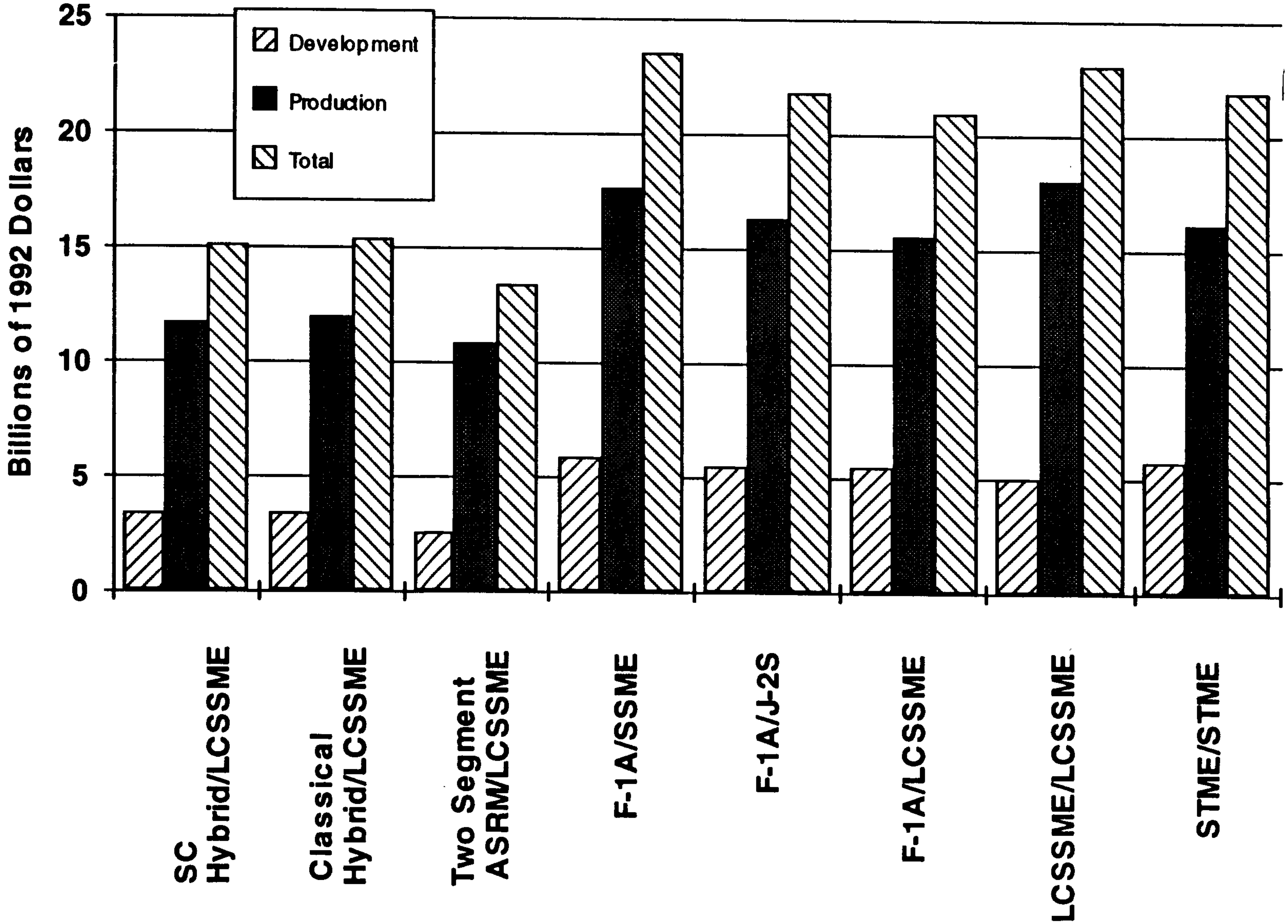
J. B. McCurry
205-722-4509

Program Total Costs by Stage



J. B. McCurry
205-722-4509

Program Costs by Phase



J. B. McCurry
205-722-4509

Conclusions

- For similar 50K designs, the main propulsion system is the primary cost discriminator

- As all main propulsion costs were provided by several differing sources, the groundrules and assumptions behind the estimates are uncertain

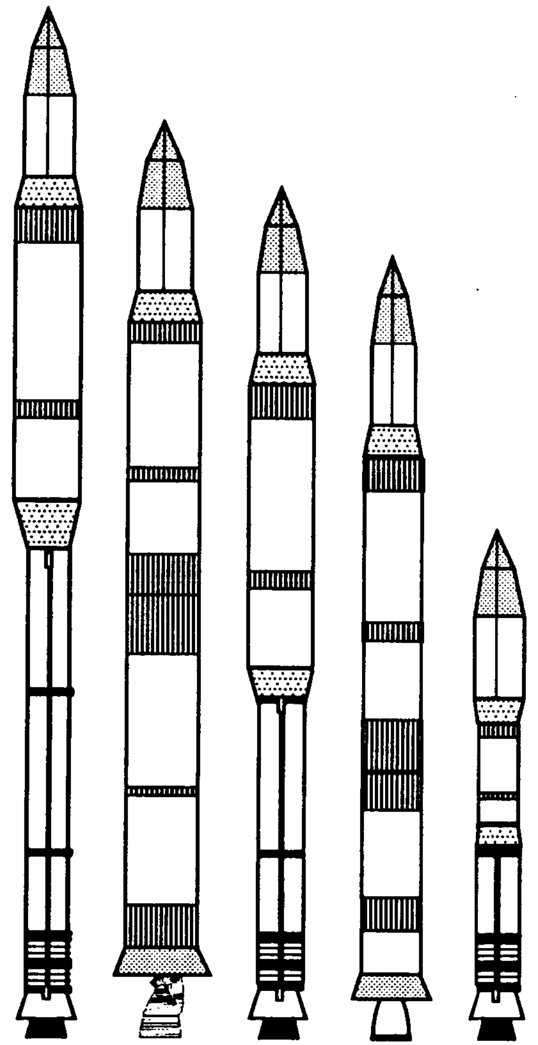
- Since the groundrules and assumptions of the primary cost discriminator (engine cost) may differ significantly, no direct comparisons can be made between the estimates
  -- This condition points out the need for consistent propulsion cost estimation methods

- In general:
  -- Solid and hybrid stages are cheaper than an equivalent liquid stage

  -- Using a single large engine in place of multiple smaller ones <u>will</u> result in lower stage unit cost

No other conclusions can be drawn from the estimates

J. B. McCurry
205-722-4509



4. Ground Operations Assessment

J. B. McCurry
205-722-4509

Major Ground Operations Assessment Accomplishments

- Integrated vehicle health management (VHM) requirements definition
- Integrated Logistics Support Plan (ILSP)
- ATSS operations index evaluation for 50K launch vehicles
- Lunar HLLV operations assessment (RD-170 vs. F-1A)
- Launch site assessment for lunar HLLV Dual Launch concept
- Lunar HLLV mixed fleet ground operations assessments
- Access to Space Option 2 ground operations assessment
- Vehicle configuration CAD support
- Concurrent engineering vehicle design assessment support

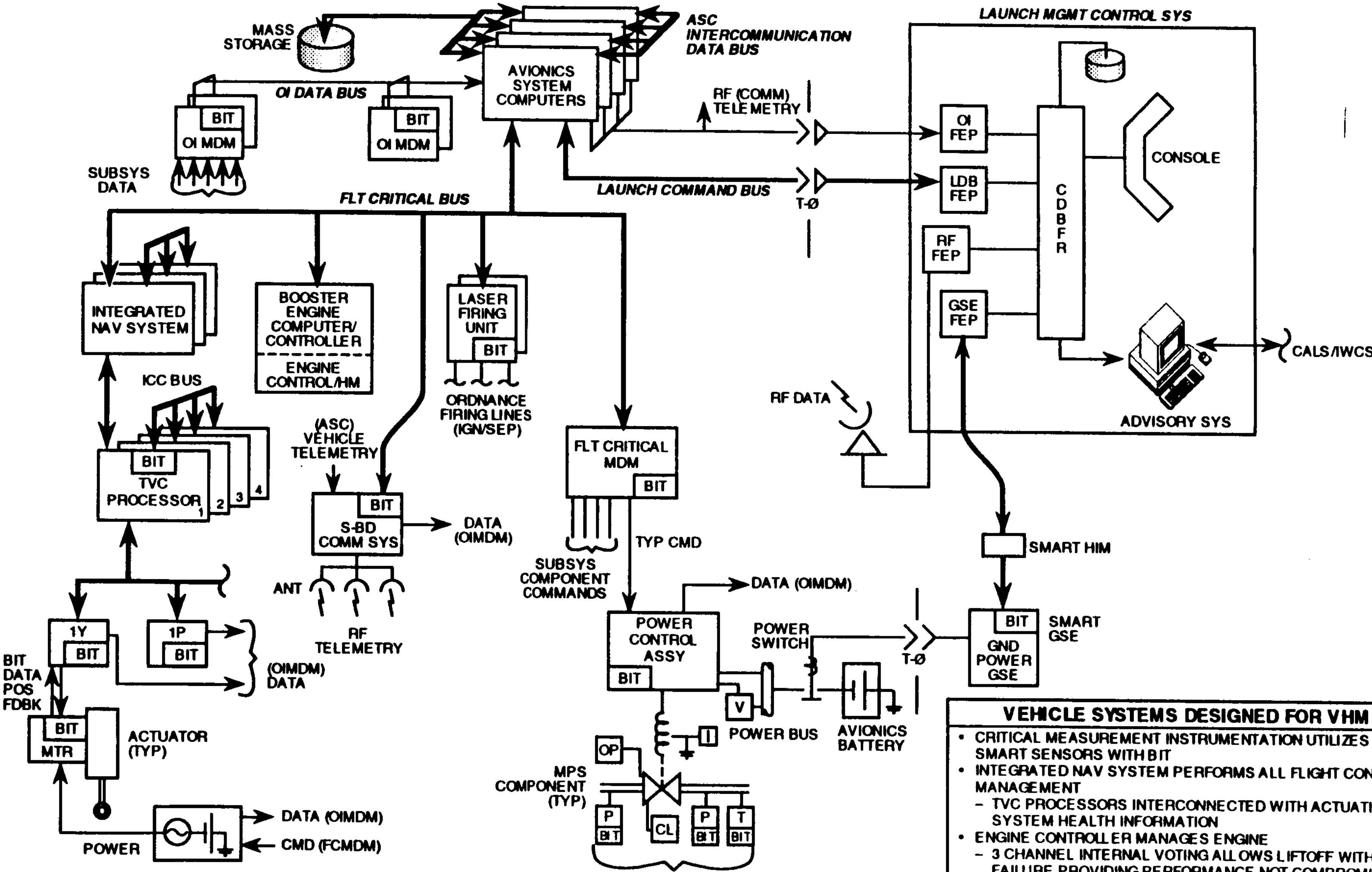
J. B McCurry
205-722-4509

Major Ground Operations Assessment Accomplishments (Concluded)

- FLO Operations Concept Document
- FLO landing and recovery concepts
- ATSS Operations Concept Document (outline)
- HLLV Technical Interchange Meetings (TIMs)
- Vehicle Health Management Workshops

J. B. McCurry
205-722-4509

VHM Requirements Definition



VEHICLE SYSTEMS DESIGNED FOR VHM
• CRITICAL MEASUREMENT INSTRUMENTATION UTILIZES SMART SENSORS WITH BIT • INTEGRATED NAV SYSTEM PERFORMS ALL FLIGHT CONTROL MANAGEMENT – TVC PROCESSORS INTERCONNECTED WITH ACTUATION SYSTEM HEALTH INFORMATION • ENGINE CONTROLLER MANAGES ENGINE – 3 CHANNEL INTERNAL VOTING ALLOWS LIFTOFF WITH FAILURE PROVIDING PERFORMANCE NOT COMPROMISED • ASC PERFORMS SYSTEM MANAGEMENT – INTERFACE TO GND CONTROL (LMCS) – SUBSYSTEMS OPERATIONS AND CONTROL – VHM BASED ON SENSOR, SUBSYSTEM PERFORMANCE, BIT, TREND DATA AND SOFTWARE

VEHICLE HEALTH MANAGEMENT SYSTEM ARCHITECTURE

Lockheed, Aerojet, ECON, Pratt & Whitney

J. B. McCurry
205-722-4809

Operability Assessment of 50/80K Vehicles

- LSOC utilized a Ground Operations Index model to assess first-order ground operations figures of merit for the 35 50/80K two-stage configurations generated by LMSC

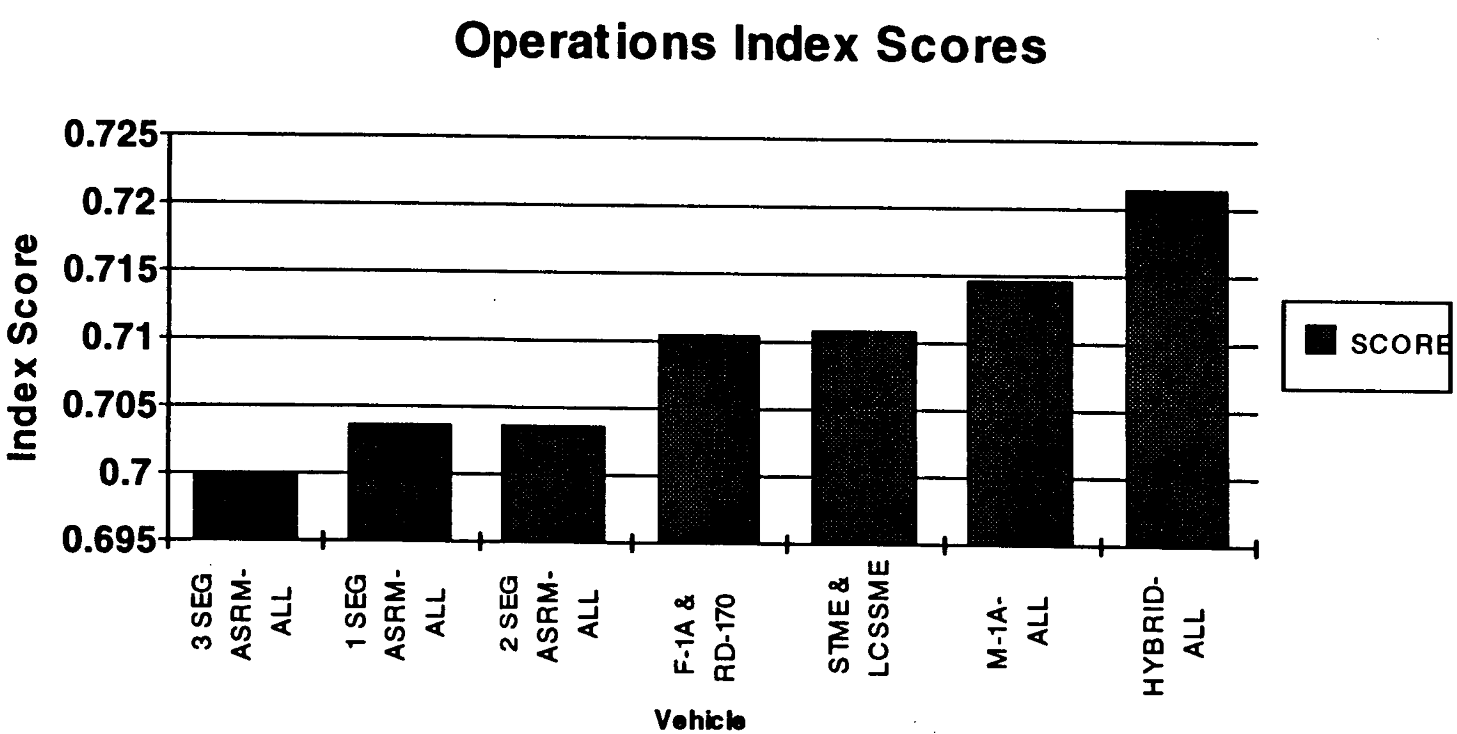
- The model is most useful in providing a relative operability ranking among launch vehicle candidates when detailed configuration definition is limited or not available

- Due to the relatively high degree of second stage commonality between the 35 candidate configurations, the primary discriminator became the first and second stage engine selection (and inherent complexity)

- The configuration figure-of-merit score is a weighted sum of the scores for a series of operability complexity factor utility parameters (number of stage elements, manned/unmanned,processing concept, number of fluids, etc.)

J. B. McCurry
205-722-4509

Operability Assessment of 50/80K Vehicles



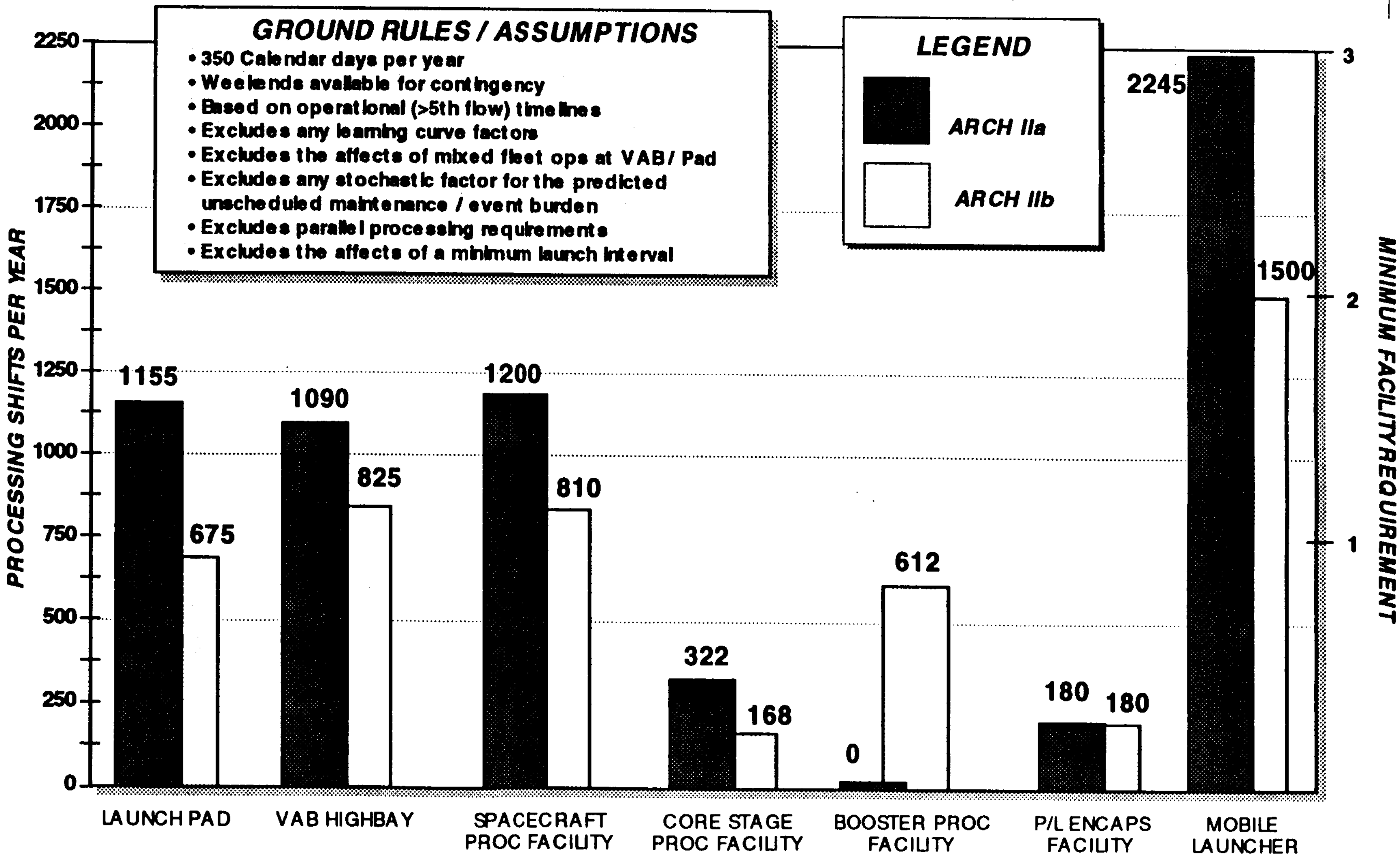
Note:
A higher score indicates better operability

NAME	SCORE
3 SEG ASRM-ALL	0.6997
1 SEG ASRM-ALL	0.7035
2 SEG ASRM-ALL	0.7035
F-1A & RD-170	0.7103
STME & LCSSME	0.7108
M-1A-ALL	0.7146
HYBRID-ALL	0.7214

J. B. McCurry
205-722-4509

Access to Space Option 2 – Ground Operations Assessment

KSC FACILITY UTILIZATION
(5 workdays/week, 3 shifts/day)

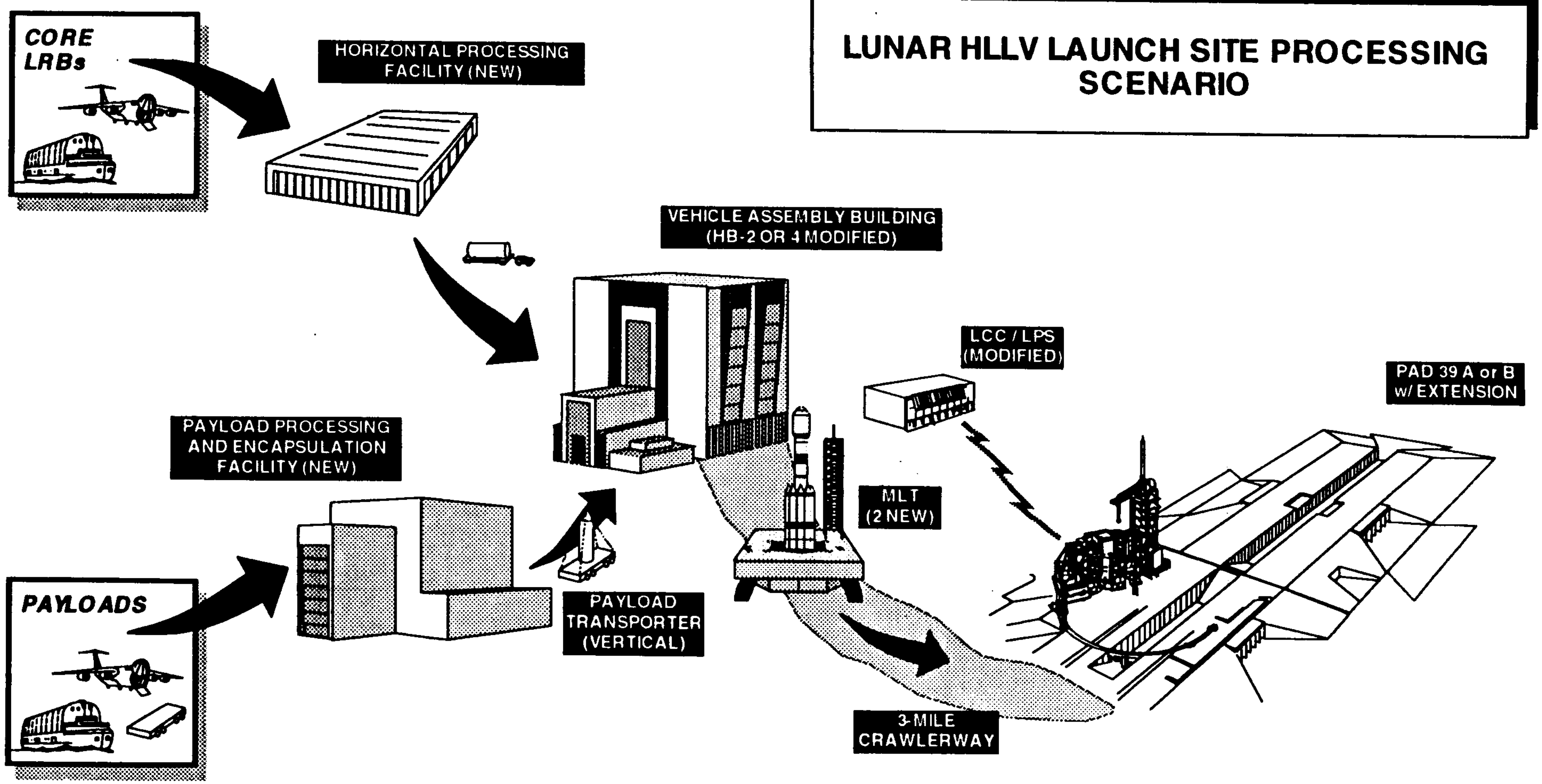


Lockheed, Aerojet, ECON, Pratt & Whitney

J. B. McCurry
205-722-4609

Lunar HLLV Operations Assessment– (RD-170 vs F-1A Engines)

LUNAR HLLV LAUNCH SITE PROCESSING SCENARIO

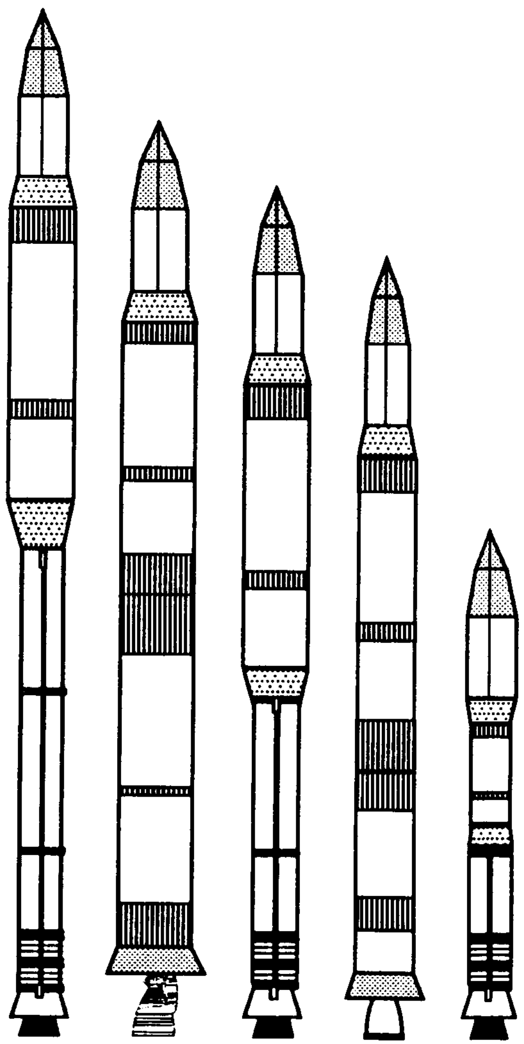


J. B. McCurry
205-722-4509

Integrated Logistics Support Plan (ILSP)

- Logistics Support System
  - -Operational Logistics Support Plans
  - -Supportability Effectiveness System
  - -Logistics Information Management System
  - -Logistics Management Responsibility Transfer
- Maintenance
  - -Maintenance Concepts
  - -Logistics Engineering Analysis
  - -On line Maintenance
  - -Off line Maintenance
  - -Maintenance Verification
- Supply Support
  - -Sparing Concepts
  - -Inventory Management
  - -Commodities
- Transportation/Handling/Storage
- Training

J. B. McCurry
205-722-4509



5. Russian Propulsion Assessment

J. B. McCurry
205-722-4509

Russian Propulsion Assessment

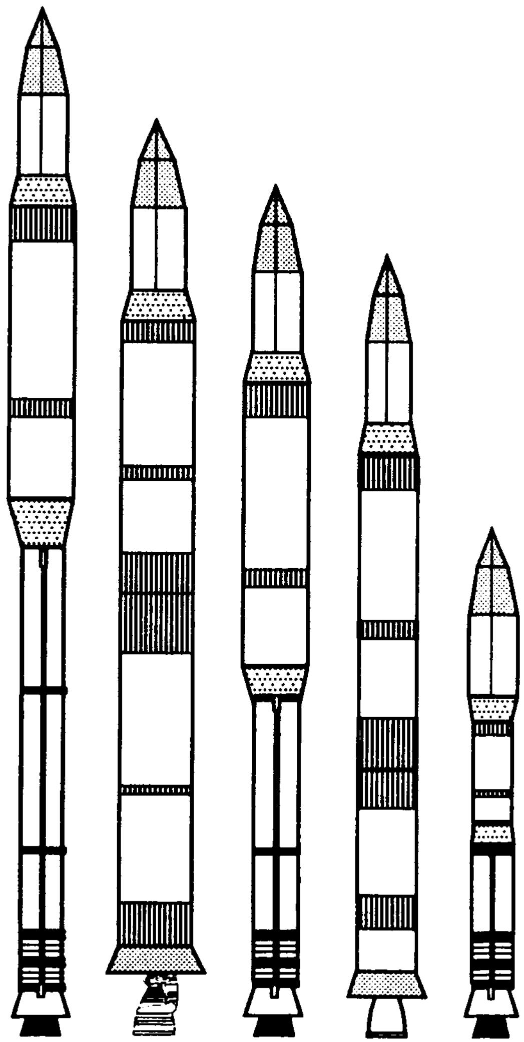
- Subcontract awarded to Pratt & Whitney via TA-2 to utilize their exclusive domestic (US) marketing agreement with NPO Energomash to obtain performance, technology, and programmatic cost data on the RD-170 (LOX/kerosene) and RD-701 (LOX/kerosene/LH_2) engines
  - -- RD-170 flies on Zenit and Energia launch vehicles
  - -- RD-701 design 80% complete to "release drawing" level

- Initial Pratt/NPOE effort (TA-2 Change Order 1) to formally provide:
  - -- Demonstrated RD-170 and theoretical RD-701 performance data
  - -- Preliminary assessment of RD-170 production in CIS versus US
  - -- Preliminary assessment of RD-170 test requirements for US site

- Initial activity completed 5/14/93 with final presentation 5/20/93

- Subsequent change order (TA-2 Change Order 2) has tasked Pratt/NPOE to provide:
  - -- Assessment of technologies associated with RD-701 based upon further understanding of RD-170 technologies
  - -- Data supporting MSFC development of RD-170 test plan

J. B. McCurry
205-722-4509



6. Near-Term Activities

J. B. McCurry
205-722-4509

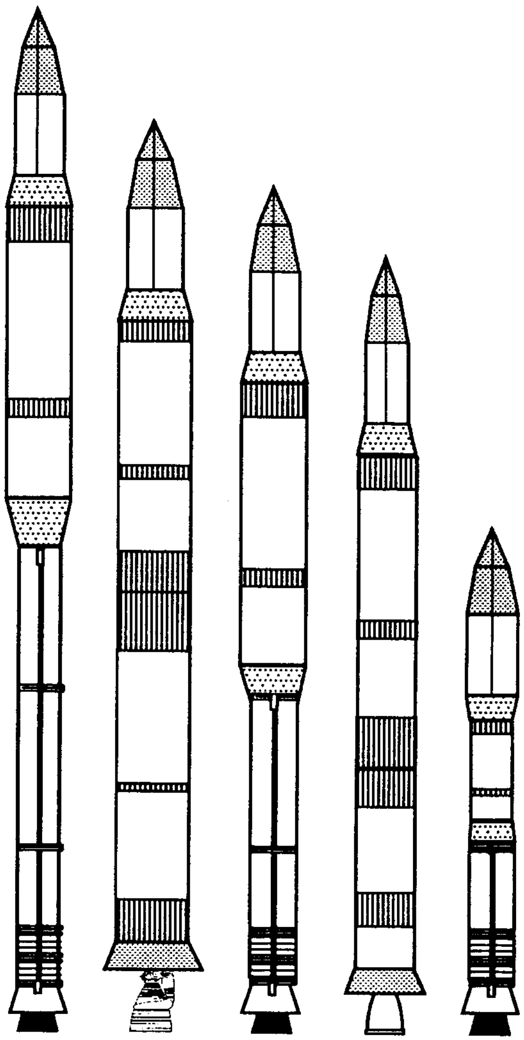
TA-2 Near-Term Activities

- Continued launch vehicle assessment support to Access-to-Space panels 2 & 3 follow-on efforts as directed by cognizant MSFC management
  - -- Resources used to complement MSFC and KSC in-house efforts
  - -- Have received praise for timely support of initial panel activities

- Further identification of domestic technology development options that enable or significantly enhance next transportation system options

- Further definition of RD-170 and RD-701 technologies and associated development requirements

J. B McCurry
205-722-4509



Advanced Transportation System Studies

TA-2 Contract Status

12 August 1993

Advanced Development
Heavy Lift Launch Vehicle Concepts

OBJECTIVES

Perform Heavy Lift Launch Vehicle concept definition and analysis to assist NASA in the identification of future launch vehicle requirements, including:

- Vehicle sizing and performance analysis
- Subsystem concept definition
- Propulsion definition (foreign & domestic)
- Ground operations and facilities analysis
- DDT&E and production cost estimation

BENEFITS

- Identification of candidate launch vehicle requirements
- Identification of vehicle evolution/growth paths
- Identification of technology development requirements & priorities
- Identification of ground operations & facilities requirements
- Identification of minimum DDT&E and minimum recurring cost vehicle concepts

ACCOMPLISHMENTS

FIRST LUNAR OUTPOST (FLO)

- NLS-derived parallel-burn HLLV concepts
- ET-derived & clean-sheet series-burn HLLV concepts
- Ground op.s assessments, all concepts
- Domestic & foreign propulsion assessments
- Launch vehicle technology development priorities

ACCESS TO SPACE

- 50/80K series-burn two-stage concepts (35)
  --Liquid/liquid;"classical" hybrid/liquid; "staged combustion" hybrid/liquid; ASRM-derived/liquid
- Methane vs. RP-1 sizing trade study
- First-order ground op.s assessments
- Russian RD-170/RD-701 data packages
- RD-170 CIS/domestic production assessment
- DDT&E, production costs of 50K config.s
- Integrated Vehicle Health Management requirements
- Integrated Logistics Support Plan
- Operations index evaluation of all 50/80K concepts

PROGRAMMATICS

- Project Manager: Gary W. Johnson;MSFC PT21; Tel. No. 205-544-0636
- Contractor: James B. McCurry; Lockheed Missiles & Space Company; Huntsville, Al; Tel. No. 205-722-4509

Contract Major Events Chronology

Jan.-Sept. 1992: First Lunar Outpost (FLO) Support

- HLLV design goals identification and ranking via concurrent engineering process for min. DDT&E cost , min. recurring cost, and min. risk scenarios

- HLLV configuration identification and sizing for min. DDT&E scenario

- First-order HLLV launch site evaluation and ground op.s assessments of mixed fleet architectures supporting Red/Blue teams and SSF-assembly "Super Red Team"

Oct.1992 -Mar. 1993: FLO Support

- Early HLLV derived parallel-burn and series-burn HLLV configuration and ground op.s assessments and liquid and hybrid 50+K two-stage concept definition and assessment for evolution into FLO HLLV strap-on boosters

- Alternative HLLV structural/manufacturing design concept assessments

- Ground op.s assessments of mixed fleet architectures supporting SSF-assembly "Super Red Team" and multiple-booster FLO HLLV configurations (8 F-1A boosters vs. 7 RD-170 boosters)

- Integrated HLLV vehicle health management requirements identification supporting enhanced manufacturing and operability

April-May 1993: Access-to-Space Support

- Liquid, hybrid, solid two-stage concept definition and assessment

- Russian propulsion preliminary assessment

- Mixed fleet ground operations assessments

Access-to-Space Propulsion Configuration Matrix

First Stage/Second Stage Options		
Liquid/Liquid *	Hybrid/Liquid *	Solid/Liquid
F-1A/LCSSME F-1A/J-2S F-1A/SSME F-1A/RD-0120 F-1A/Vulcain STME/LCSSME STME/STME STME/RD-0120 STME/Vulcain M-1A/LCSSME M-1A/RD-0120 M-1A/Vulcain RD-170/LCSSME RD-170/J-2S RD-170/RD-0120 RD-170/Vulcain LCSSME/LCSSME LCSSME/RD-0120 LCSSME/Vulcain	Staged Combustion Hybrid/LCSSME Staged Combustion Hybrid/Rubber STME Staged Combustion Hybrid/J-2S Staged Combustion Hybrid/Vulcain Staged Combustion Hybrid/RD-0120 Classical Hybrid/LCSSME Classical Hybrid/Rubber STME Classical Hybrid/J-2S Classical Hybrid/Vulcain Classical Hybrid/RD-0120	3 Segment ASRM/LCSSME 3 Segment ASRM/LCSSME 2 Segment ASRM/J-2S 2 Segment ASRM/LCSSME 2 Segment ASRM/SSME 1 Segment ASRM/Centaur

Note: * Configurations sized for 50 Klbm payload

Vehicle Configuration Payload

Liquid/Liquid *	Payload (lbm) **
F-1A/LCSSME	48,249
F-1A/J-2S	54,893
F-1A/SSME	51,098
F-1A/RD-0120	48,599
F-1A/Vulcain	49,155
STME/LCSSME	48,321
STME/STME	48,034
STME/RD-0120	50,186
STME/Vulcain	49,986
M-1A/LCSSME	47,992
M-1A/RD-0120	49,471
M-1A/Vulcain	48,993
RD-170/LCSSME	49,878
RD-170/J-2S	50,166
RD-170/RD-0120	48,598
RD-170/Vulcain	50,598
LCSSME/LCSSME	48,222
LCSSME/RD-0120	49,339
LCSSME/Vulcain	49,071

Note: * First Stage/Second Stage Propulsion Options
** Payloads verified by 3-DOF trajectory analysis

Vehicle Configuration Payload (Concluded)

Hybrid/Liquid *	Payload (lbm) **
Staged Combustion Hybrid/LCSSME	51,773
Staged Combustion Hybrid/Rubber STME	54,836
Staged Combustion Hybrid/J-2S	50,610
Staged Combustion Hybrid/Vulcain	50,072
Staged Combustion Hybrid/RD-0120	51,354
Classical Hybrid/LCSSME	51,663
Classical Hybrid/Rubber STME	54,987
Classical Hybrid/J-2S	50,559
Classical Hybrid/Vulcain	49,962
Classical Hybrid/RD-0120	51,265

Solid/Liquid *	Payload (lbm) **
3 Segment ASRM/LCSSME	65,000
3 Segment ASRM/SSME	82,100
2 Segment ASRM/J-2S	43,600
2 Segment ASRM/LCSSME	49,300
2 Segment ASRM/SSME	56,600
1 Segment ASRM/Centaur	6,900

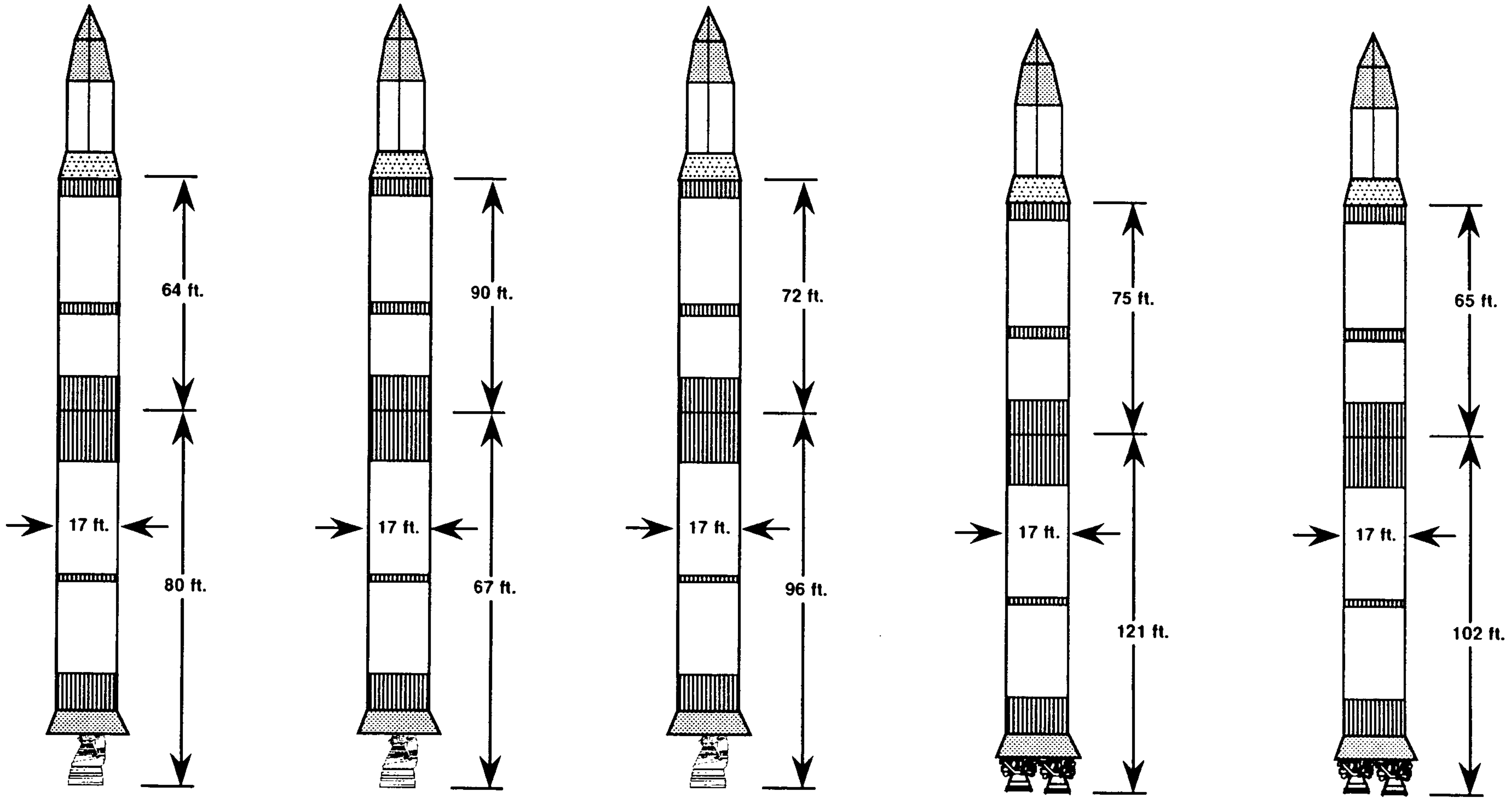
Note: * First Stage/Second Stage Propulsion Options
** Payloads verified by 3-DOF trajectory analysis

Engine Specifications

	M-1A	F-1A	STME	SSME (104% RPL)	RD-170
Sea Level Thrust (lbf)	1,300,000	1,800,000	551,430	390,000	1,632,000
Vacuum Thrust (lbf)	1,562,000	2,020,700	650,000	488,800	1,777,000
Sea Level Specific Impulse (sec)	344.5	269.7	364	364.8	309
Vacuum Specific Impulse (sec)	414.0	303.1	428.5	452.9	337
Chamber Pressure (psia)	1,000	1,161	2,250	3,110	3,560
Mixture Ratio	5.0	2.27	6.0	6.0	2.6
Area Ratio	20	16	45	77.5	36.87
Engine Mass (lbm)	20,200	19,000	9,974	6,990	21,510
Engine Length (ft)	19.08	18.36	13	14	13.12
Engine Diameter (ft)	12.58	11.96	12.1	8	12.20
Propellant	O2/H2	O2/RP-1	O2/H2	O2/H2	O2/Syn10

	RD-0120	Vulcain	J-2S	LCSSME (Altitude)	LCSSME (Sea Level)
Sea Level Thrust (lbf)	352,746	---	197,000	---	506,000
Vacuum Thrust (lbf)	440,925	230,000	265,000	326,600	600,000
Sea Level Specific Impulse (sec)	364	---	320	---	371
Vacuum Specific Impulse (sec)	455	431.6	436	451.9	440
Chamber Pressure (psia)	3,000	1,450	1,200	2,075	2,075
Mixture Ratio	6.0	5.2	5.5	6.0	6.0
Area Ratio	85.7	45	40	77.5	42
Engine Mass (lbm)	7,607	2,860	3,800	7,053	7,300
Engine Length (ft)	14.93	9.62	11.08	14	14
Engine Diameter (ft)	7.94	5.77	6.71	8	8
Propellant	O2/H2	O2/H2	O2/H2	O2/H2	O2/H2

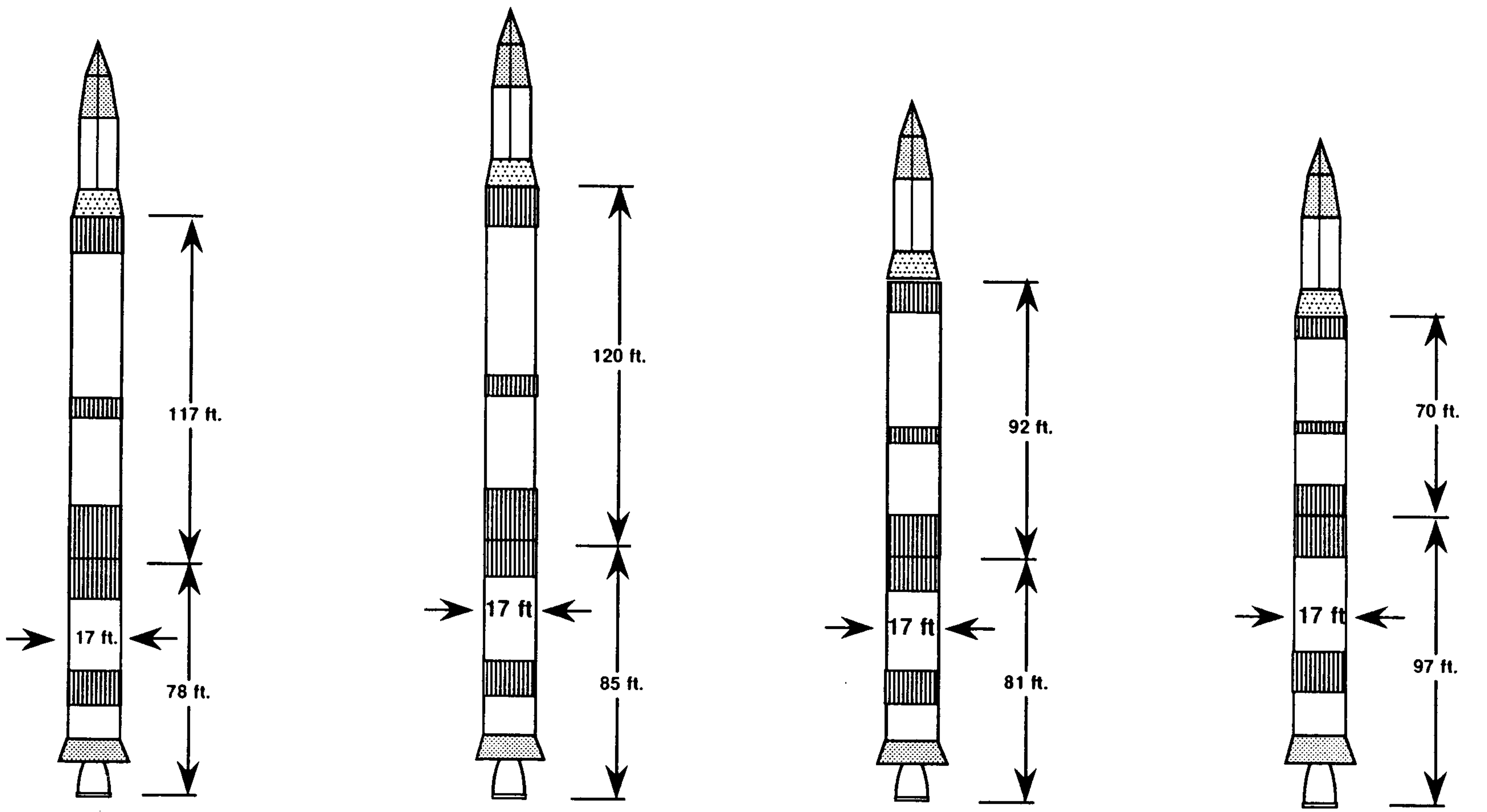
Examples of All-Liquid 50K Concepts



Length/Diameter= 8.5	Length/Diameter= 9.2	Length/Diameter= 9.9	Length/Diameter= 11.5	Length/Diameter= 9.8
F-1A/LCSSME	**F-1A/SSME**	**F-1A/J-2S**	**STME/STME**	**LCSSME/LCSSME**

(Vehicles not to scale)

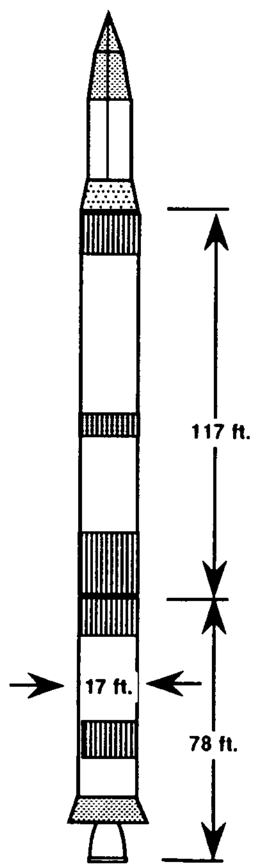
Examples of Hybrid/Liquid 50K Concepts



Length/Diameter= 11.5	Length/Diameter= 12.0	Length/Diameter= 10.1	Length/Diameter= 9.8
Staged Comb./ Rubber STME	**Classical/ Rubber STME**	**Classical/ LCSSME**	**Classical/ Vulcain**

(Vehicles not to scale)

50 k Vehicle, Staged Combustion Hybrid/Rubber STME



Length/Diameter= 11.5

Payload:	54,836 lbm (24.9 t)
Final Position:	15x220 NM Orbit, i= 28.5 deg

GLOW: 1,302,605 lbm

First Stage:

Inert Mass:	95,402 lbm
Usable Propellant:	620,000 lbm
Propellant Type:	LOX/PEBC
Engine Type/No.:	Staged Combustion Hybrid/1
Diameter:	17.0 ft
Thrust/Weight:	1.443 g
Sea Level Thrust	1,800,000 lbf
Throttle Setting	100.0 %

Second Stage:

Inert Mass:	36,367 lbm
Usable Propellant:	446,000 lbm
Propellant Type:	LOX/LH2
Engine Type/No.:	Rubber STME/1
Diameter:	17.0 ft
Vacuum Thrust	425,894 lbf
Thrust/Weight:	0.800 g

Russian Propulsion Assessment

- Subcontract awarded to Pratt & Whitney via TA-2 to utilize their exclusive domestic (US) marketing agreement with NPO Energomash to obtain performance, technology, and programmatic cost data on the RD-170 (LOX/kerosene) and RD-701 (LOX/kerosene/LH_2) engines
  - -- RD-170 flies on Zenit and Energia launch vehicles
  - -- RD-701 design 80% complete to "release drawing" level

- Initial Pratt/NPOE effort (TA-2 Change Order 1) to formally provide:
  - -- Demonstrated RD-170 and theoretical RD-701 performance data
  - -- Preliminary assessment of RD-170 production in CIS versus US
  - -- Preliminary assessment of RD-170 test requirements for US site

- Initial activity completed 5/14/93 with final presentation 5/20/93

- Subsequent change order (TA-2 Change Order 2) has tasked Pratt/NPOE to provide:
  - -- Assessment of technologies associated with RD-701 based upon further understanding of RD-170 technologies
  - -- Data supporting MSFC development of RD-170 test plan

Major Contract Deliverables

- ET-derived First Lunar Outpost (FLO) parallel burn launch vehicle concepts
- FLO launch site first-order assessment
- FLO single-launch vs. dual launch ground operations assessment
- FLO launch vehicle lift-off tower-clear/drift dispersion requirements assessment
- FLO/Shuttle/ELV mixed fleet ground operations assessment
- Early Heavy Lift derived parallel burn and series burn HLLV configuration & ground operations assessments
- Alternative HLLV structural design concept assessment
- 50/80K HLLV concept definition & sizing assessment (liquid, hybrid, solid options)
- Preliminary Russian booster propulsion technology assessment (RD-170 & RD-701)
- Integrated "cradle-to-grave" vehicle health management requirements assessment
- Integrated Logistics Support Plan draft

Major TA-2 Presentations Given in 1994

Qualitative Assessment of Tripropellant Main Propulsion versus Bipropellant Main Propulsion for Single Stage to Orbit Vehicles

J. B. McCurry
205-722-4509
2/7/94

SSTO Operability Pros/Cons of Bipropellant vs. Tripropellant

Effect on Recurring Ops Cost if Tripropellant Vehicle

- Lockheed, LSOC, and Aerojet brainstormed operability issues associated with tripropellant (LOX/LH_2/Hydrocarbon) main propulsion versus LOX/LH_2 bipropellant
  --"Does tripropellant hurt/help recurring operations costs?"

- Generic main propulsion concepts were assumed (engine/cycle independent)

- Operational issues associated with cryogenic third propellant versus non-cryogenic were considered

J. B. McCurry
205-722-4509
2/7/94

SSTO Operability Pros/Cons of Bipropellant vs. Tripropellant

Major Assumptions :

- Recurring operations costs are *the* dominant driver for SSTO viability
  - -- Outweigh relative differences in vehicle unit costs for different design concepts due to small fleet size (regardless of historical trend by U.S. Govt. to consider production cost a "sunk cost")

- All SSTO vehicle designs will meet the mission payload requirement

- For typical range of SSTO vehicle sizes within a class (VTVL or VTHL) the development of a "new" main engine will allow the same number of engines to be used, regardless of vehicle size (within a particular class)
  - -- Same number of engines for biprop. or triprop. solutions presuming rubber engines (thrust level & Pc sized as needed)
  - -- True for aerospike or bell nozzle concepts

Assessment Question :

- Does tripropellant concept help to reduce recurring operations costs as compared to a bipropellant concept?
  - -- If not, tripropellant engines should not be invested in; use technology development funds to obtain more reliable bipropellant engine

J. B. McCurry
205-722-4509
2/7/94

SSTO Operability Pros/Cons of Bipropellant vs. Tripropellant
Effect on Recurring Ops Cost if Tripropellant Vehicle

Pros	Cons	Neutral
• Less stand-off structure (for non-integral propellant tank designs) allows less structural maintenance • Less TPS allows less body TPS refurb. & repair • Use of noncryogenic third propellant facilitates prop. loading timeline vs. cryogenic third prop.	• ~50% increase in main prop. feed & press. parts count, increasing processing test & checkout by 50% • Increased parts count increases likelihood of unscheduled maintenance • Increased unscheduled maintenance increases logistics burden (spares) • "New" nature of triprop. propulsion increases likelihood of infant mortality failures in propulsion components • Increased complexity increases processing learning curve • Decreased vehicle size and increased parts count increases maintenance accessibility difficulty (if not considered in the design)	• Smaller vehicle but not a driver for SSTO class -- Same number of engines to process as biprop. for new "rubber engine" -- Processing not affected by vehicle size (up to a point)

SSTO Operability Pros/Cons of Bipropellant vs. Tripropellant
Effect on Recurring Ops Cost if Tripropellant Vehicle (concl)

Pros	Cons	Neutral
	• Increased ground checkout and launch software will increase sustaining software maintenance • Third propellant is an additional commodity to buy, transport, store and load at launch pad • 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 • No capability to verify 90% engine health onpad in both modes prior to liftoff • Higher flight performance reserve for 3 propellants • Use of cryogenic third prop. complicates prop. loading timeline • Fuel mode optimization complicates nominal/abort flight design	

SSTO Operability Pros/Cons of Bipropellant vs. Tripropellant
Effect on Non-Recurring Ops Cost if Tripropellant Vehicle

Pros	Cons	Neutral
• Smaller vehicle will require less primary structure and associated TPS materials • Smaller vehicle will require less stand-off structure, for non-integral prop. tank designs	• Increased propulsion complexity will require more ground checkout and launch software • Increased propulsion complexity will require more flight ops software • Environmental hazard mitigation for hydrocarbons will require spill pond, water sample wells, and possibly a waste water treatment facility • Increased propulsion complexity will require more extensive engine qualification and certification program • Additional hazardous gas detection hardware onboard • Additional propellant tankage with associated tank insulation	

SSTO Operability Pros/Cons of Bipropellant vs. Tripropellant

Conclusions :

- Tripropellant inherently more complicated
- Tripropellant main propulsion will adversely affect recurring launch vehicle maintenance and ground processing
- Automated main propulsion feed subsystem test and check-out will help to mitigate hands-on processing but inherently higher parts count will result in higher unscheduled maintenance than bipropellant
- A significant reduction in number of required engines would be needed for tripropellant option to achieve parts count reduction over bipropellant option
- Recurring operations cost approximately 50% higher for tripropellant
- DDT&E and vehicle per unit cost also likely higher for tripropellant

Recommendation :

- Apply scarce DDT&E funds to pursue simpler, robust, operable bipropellant engine concept for SSTO

J. B. McCurry
205-722-4509
2/7/94

Advanced Transportation System Studies
TA-2 Contract
Status

Assessment
of
Single Stage to Orbit Concepts

Given
at the
Marshall Space Flight Center
February 11, 1994

Lockheed

Agenda

1. Assessment Introduction	J. McCurry
2. SSTO Design Groundrules	J. McCurry
3. Operations Issues	G. Letchworth
4. VTOL/VTHL Pros & Cons	G. Letchworth
5. Design Results	K. Holden
6. Simulation Results	K. Sagis
7. Technology Requirements Summary	J. McCurry
8. Conclusions	J. McCurry

Assessment Introduction

- Why SSTO?
- TA-2 Assessment Purpose & Approach
- Past Figures of Merit
- Today's Figures of Merit
- Design Process
- Operability and Integration Design Goals
- Configuration Overview

J. B. McCurry
205-722-4509

Why is the concept of Single Stage to Orbit (SSTO) even considered?

- Classical rocket sizing equations based on rocket equation indicate that the combination of multiple stage elements (usually 2-3) "best" accomplishes a given mission total ΔV requirement
  - "Best" becomes a function of the figures of merit used

- Historic use of "subsidies" by governments to develop new launch vehicles has masked the influence of economic forcing functions and diluted their ability to incentivize efficiency

Mission Cost = recurring fixed cost (infrastructure, W.O.D.B.) + cost due to size (materials, manufacturing)
+ cost due to technology/design (complexity, integr., degree of reuse, refurb., test & checkout, etc.) + DDT&E amortization

+ . . .

- SSTO validity must balance the benefit of fewest number of stage elements with performance efficiency and design complexity needed to accomplish mission requirements

J. B. McCurry
205-722-4509

TA-2 Assessment Purpose

Identify first order design sensitivities for some SSTO configurations not assessed by Access to Space Option 3 team ("help fill in some other squares")

- Outer moldline considerations
- Major structural element layout
- Propellant combination
- Main propulsion selection

J. B. McCurry
205-722-4509

TA-2 Assessment Approach

- Option 3 team's groundrules, assumptions, mission requirements, and types of technologies were used for "apples-to-apples" comparisons
  - Applied equally to each configuration type

- Lockheed's SSTO sizing tool also had to be calibrated against known sizing methods used by Option 3 team (LaRC personnel)
  - Option 3's final baseline configuration used (LaRC's "wing-body" integral tank tripropellant design)
  - Common sizing methodology used for each configuration type
  - Some Option 3 design assumptions remain to be identified

- Time & money have run out prior to completion of engine/propellant options and assessment of enhancing technology sensitivities

J. B. McCurry
205-722-4509

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

J. B. McCurry
205-722-4509

Today's Figures of Merit

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

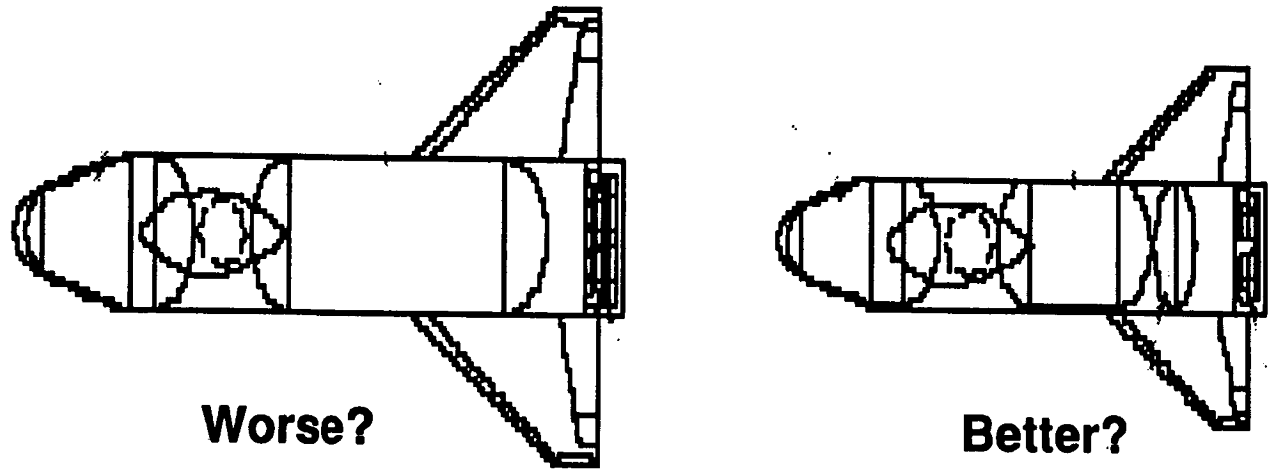
J. B. McCurry
205-722-4509

SSTO Design Process

- 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
  - Structural materials
  - Outer moldline shape
  - Operations scenarios
  - Propulsion system
  - Major subsystem layout
  - TPS types
- Size the vehicles, simulate ascent/entry trajectories
  - Performance
  - First order loads
  - Aerodynamic heating
  - Flight mechanics
- Resize as needed

J. B. McCurry
205-722-4509

"Things aren't necessarily what they seem!"



A smaller vehicle may or may not:

- **Decrease number of engines**
- **Enhance design complexity**
- **Enhance vehicle unit cost**
- **Enhance operations cost**
  - **Weight-based cost estimating relationships can be misleading**
  - **Op.s cost is strong function of "a priori" program requirements**

J. B. McCurry
205-722-4509

Operability & Integration Design Goals

During the conceptual design of any new launch vehicle, an integrated approach must be taken in which design goals for each functional subsystem are 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
- Construction methods
- Primary structure materials
- Intertank/interstage design
- Stage thrust structure design
- Propellant feed subsystem design
- Main stage propulsion type

In addition, there are subsystem-independent design goals.

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

Subsystem Independent Design Goals (Concluded)

- 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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

Typical Subsystem Functional Work Breakdown Structure

- Active Thermal Control
  - Supporting any subsystem need
- Avionics
  - Guidance, navigation, control, data processing, communications & tracking, instrumentation, caution & warning (manned missions), electrical power control & distribution, range safety
- Crew Escape (manned missions)
- Electrical Power
  - Generation, auxiliary power unit
- Environmental Control & Life Support (manned missions)
  - Atmospheric revitalization, airlock, water & waste management, EVA support, smoke detection, fire suppression
- Main Engine

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

Typical Subsystem Functional Work Breakdown Structure (Concluded)

- Main Engine Propellant Pressurization and Feed
  - Propellant tanks, feed lines, pressurization subsystem, valves, prevalves
- Mechanical
  - Latches, actuators, doors, aerosurfaces, landing/deceleration, payload deploy/retrieval, pyrotechnics, payload bay door(s)
- Orbital Maneuvering
- Passive Thermal Control
- Reaction Control
  - Primary, vernier
- Structure
  - Primary, secondary, purge, vent, drain

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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)

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

Orbital Maneuvering

- Size for ~1000 fps ΔV 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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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

J. B. McCurry
205-722-4509

Operability & Integration Design Goals
(Continued)

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 ΔV for ascent governed by method of roll control and desired rates (which is an ascent performance tradeoff)
- Size ΔV capability for sum of on-orbit and entry requirements to ~100 fps

J. B. McCurry
205-722-4509

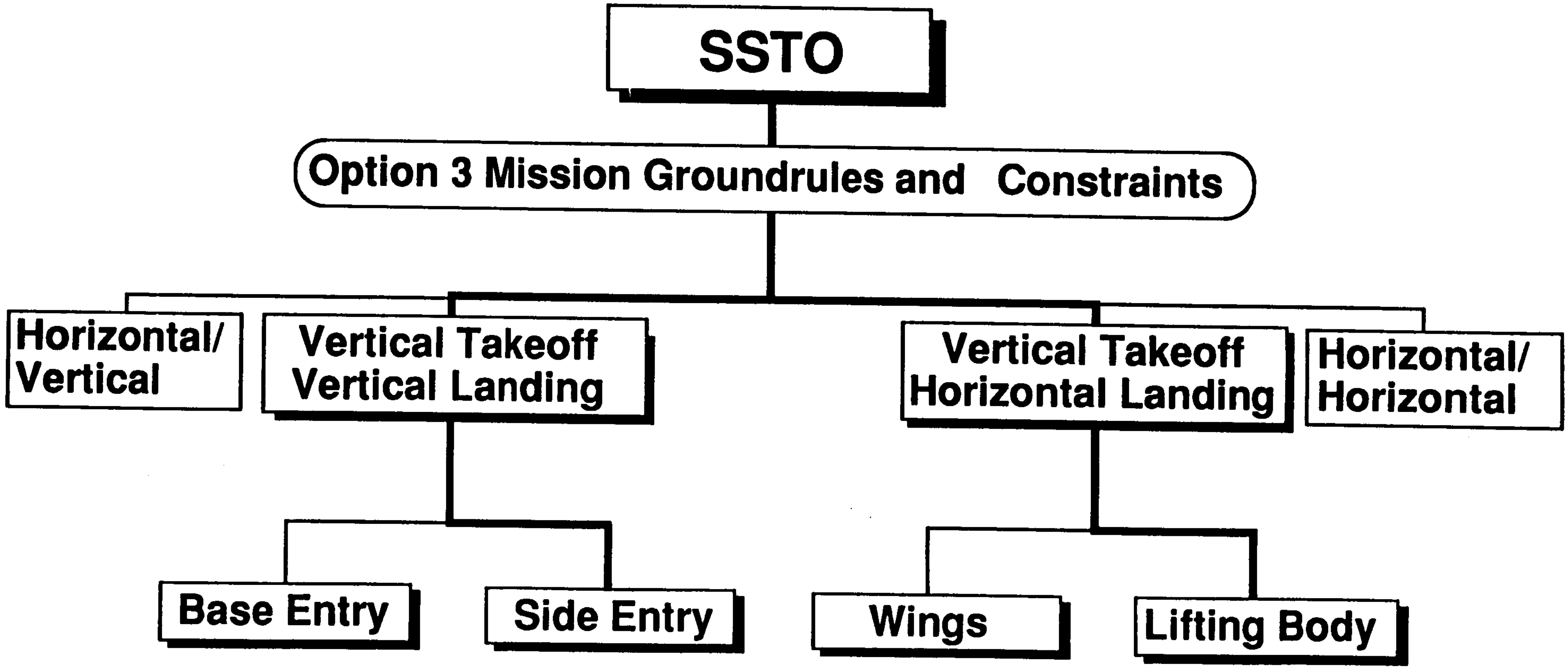
Operability & Integration Design Goals
(Concluded)

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

J. B. McCurry
205-722-4509

Single Stage to Orbit Vehicle Design Path



Note: Bold indicates path taken on ATSS TA-2 contract

J. B. McCurry
205-722-4509

S-II and S-IVB stages. The J-2S uses liquid oxygen (LOX) as the oxidizer and liquid hydrogen (LH_2) as the fuel. The addition of expendable nozzle extensions and the strengthening of turbomachinery, turbine exhaust gas manifolds, and thrust chamber forward manifolds, as well as enlargement of main valve actuators produces a greater expansion ratio (40:1 versus 27.5:1) and higher specific impulse (436 seconds vs. 425 seconds) for the J-2S as compared to the J-2. The vacuum thrust of the J-2S engine is 1,178,773 N (265 Klbf). The J-2S engine has independently driven pumps for both liquid oxygen and liquid hydrogen, a gas generator to supply hot gas to two turbines functioning in series, pneumatic and electrical control interlocks, altitude restart capability, and a propellant management, or utilization monitoring, system. The J-2S has no throttle-down capability from its 100 percent RPL value.

STME

The STME is a 2.89 MN (650 Klbf) vacuum thrust engine with a designed specific impulse of 428.5 seconds, as currently baselined by the National Launch System (NLS) program. The engine is in the preliminary design phase, and consists of a LOX/LH_2 turbopump powerhead with a standard fuel-rich gas generator cycle. The combustion chamber is regeneratively cooled, and the nozzle uses both regenerative and film cooling. The STME is being designed for a 75 percent RPL minimum thrust level and will utilize a single-step throttle-down capability. The STME, while not being designed for reuse, is to be designed with robust operating margins and will have the inherent capability for multiple engine starts to support flight certification and multiple launch attempts after an on-pad abort shut-down. It is assumed that the STME development schedule will become compatible with SEI requirements.

SSME

The SSME, modified for second stage altitude start and on-orbit restart capability, will develop 2.09 MN (470 Klbf) of vacuum thrust operating at 100 percent RPL, and will not be throttled during any burn. The requirement for a vacuum start of the SSME will require modifications to the engine start sequence due to the reduced liquid oxygen (LOX) inlet pressure and zero ambient pressure, as well as modifications to the LOX feed system for the auto-spark igniter. The reference NLS-derived configuration requires the SSME to burn twice: once as a suborbital burn and once as the TLI burn. It is assumed that pre-lift-off thermal conditioning and inert gas purges will be performed on the SSME via T-0 umbilicals. Thermal conditioning and purges may be required for the suborbital burn, but further analyses must be performed to confirm that assessment. Conditioning and purges will likely be required for the TLI burn, given the possibility of 1-3 hours of on-orbit dwell time between the suborbital burn and the TLI burn. The purges also would ensure that there would not be any ice build-up in the engine after the first burn.

3.2.2 Avionics Subsystems

The avionics suite is assumed to be centralized in the TLI stage and based on the reference NLS Cycle "O" design. The accuracy requirements of the inertial navigation system are assumed to be the same as those baselined in the NLS Level II System Requirements Document (Version 6.0, Section 3.1.4.2.1):

- Apogee-- ±0.9 Km (0.5 nm)
- Perigee-- ±0.9 Km (0.5 nm)
- Inclination--±0.05 deg

Figure 3.2.2-1 illustrates the basic HLLV avionics architecture concept, shown here for the NLS-derived vehicle. Table 3.2.2-1 summarizes the location and quantities of the various avionics components on the HLLV elements. The SEI HLLV does not need a "Launch with Faults" string because there is no surge requirement and the annual flight rate is low. Active mission times for the vehicle elements are assumed to be less than 10 minutes for the booster, less than 30 minutes for the core, and 6-8 hours for the TLI stage. This concept assumes that each engine has one internally redundant engine controller with data bus interfacing included. It also assumes each engine has two electromechanical actuator (EMA) controllers. The avionics design also assumes that the HLLV is a throw-away vehicle and does not require autonomous or crew-controlled rendezvous and docking capabilities. The avionics is located in the TLI stage in order to control the boosters, core, and TLI stage during their respective flight phases. Also, the instrument unit concept is more applicable to the SEI vehicle since it has the same configuration each time and weeks can be taken for the vehicle integration processing without adversely affecting the mission. The selected architecture is a voting three-string system for the core and TLI stage with control avionics on the TLI, and dual-string avionics on the boosters. The design captures the maximum number of faults and produces the highest reliability for the least cost for short duration mission vehicles. Avionics masses include cables, EMAs, and engine controllers. This basic design will work for both unmanned and manned vehicles. However, an emergency detection function will need to be added for the manned vehicle. This function could be performed by the TLI computers using the standard vehicle health management (VHM) suite.

Power for all stages is provided by silver-zinc batteries because of the short duration mission. Each element has its own power supply and power distribution control to minimize noise, voltage drop, and cable mass. Communications for both launch and on-orbit phases of the mission are provided by the TLI stage.

HLLV Avionics

<u>TLI Stage (Central Avionics)</u>

- **Data Management**
  - Flight Computers (3)
  - Remote Data Units (12)
- **Communications**
  - Transponders (2)
  - Amplifiers (2)
  - Antennae (4)
- **Guidance and Navigation**
  - Inertial Measurement Unit
  - Navigation Update System (3)
- **Electrical Power**
  - Energy Source (Batteries) (8)
  - Power Distribution Unit (3)
- **Range Safety**
  - Receiver (2)
  - Distributor
  - Antennae (4)
- **Ordnance**

<u>Core and Booster (Distributed Elements)</u>

- **Electrical Power**
  - Energy Source (Batteries)
  - Power Distribution Unit
- **Controllers**
  - Engine
  - Thrust Vector Control
- **Range Safety**
- **Interfaces With Central Avionics**
  - Data Management
  - Sensors and Instrumentation

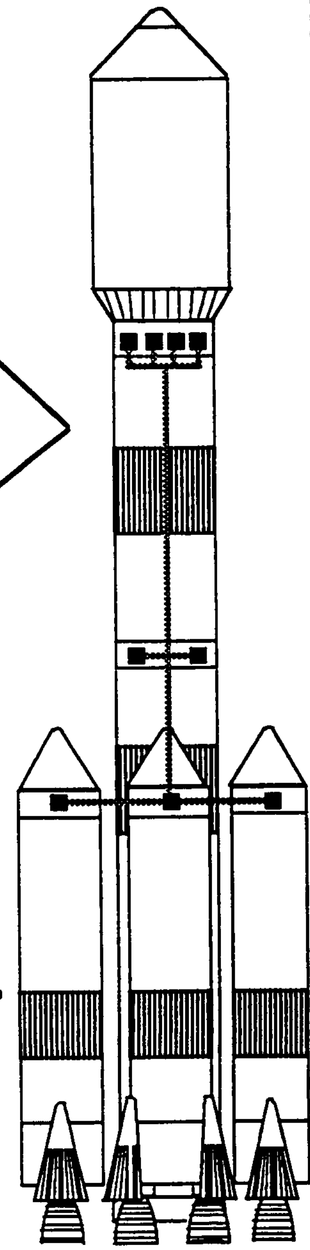


Figure 3.2-1 HLLV Avionics Architecture Concept

Table 3.2.2-1 Avionics Components List

SEI AVIONICS COMPONENT QUANTITY DISTRIBUTION

AVIONICS CATEGORY	4 BOOSTERS	CORE VEHICLE	SECOND STAGE	TLI STAGE
<u>DATA MANAGEMENT</u>				
• Remote Unit	8	8	2	12
• Flight Computer				3
• Interface Assembly (Ground Interface)		1		1
<u>COMMUNICATIONS</u>				
• Antenna				4
• S-Band Transponder				2
• S-Band Amplifier				2
• Operational Flight Instrumentation				1
<u>POWER</u>				
• Primary Batteries	8	4	2	8
• Distribution Subsystem	8	2	2	3
• Umbilical	4	4	2	2
• Protection Node	4	1	1	1
<u>GUIDANCE, NAVIGATION, & CONTROL</u>				
• Rate Gyros	4			1
• Inertial Measurement Unit				1
• Global Positioning System Unit & Preamplifier				3
• Global Positioning System Antenna				4
• Video Display Equipment				3

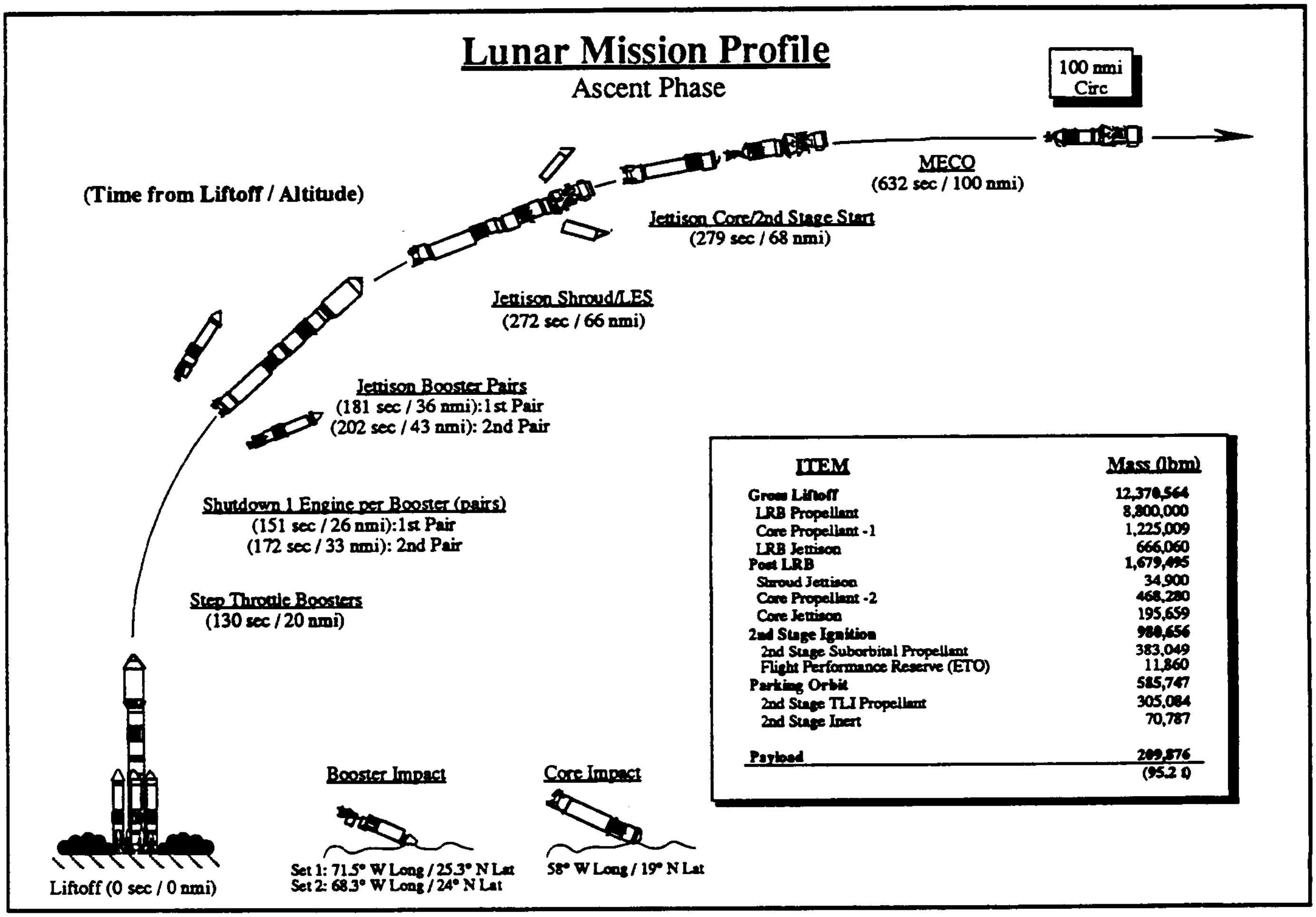
4. Reference Launch Vehicle Options

4.1 NLS Derived

The following sections present the results of the definition and assessment of the reference NLS derived launch vehicle configuration.

4.1.1 Mission Profile

Figure 4.1.1-1 illustrates the mission profile during ascent. The profile is the same for both piloted and cargo flights except that a launch escape system (LES) is included and jettisoned at shroud separation in manned missions. The on-orbit mission profile is shown in Figure 2.2-1.



ITEM	Mass (lbm)
Gross Liftoff	12,370,564
LRB Propellant	8,800,000
Core Propellant -1	1,225,009
LRB Jettison	666,060
Post LRB	1,679,495
Shroud Jettison	34,900
Core Propellant -2	468,280
Core Jettison	195,659
2nd Stage Ignition	980,656
2nd Stage Suborbital Propellant	383,049
Flight Performance Reserve (ETO)	11,860
Parking Orbit	585,747
2nd Stage TLI Propellant	305,084
2nd Stage Inert	70,787
Payload	209,876
	(95.2 t)

Figure 4.1.1-1 NLS-Derived Ascent Mission Profile

Core and booster main engines are ignited at T-0 and attain 100 percent RPL thrust prior to liftoff. The thrust-to-weight ratio at liftoff is 1.34. The vehicle flies at an angle of attack of 0 degrees through the region of maximum dynamic pressure (30 - 135 seconds mission elapsed time) to minimize structural loading. Maximum dynamic pressure during ascent is 43.1 K N/m^2 (900 psf). Booster engines are step-throttled to 75 percent RPL when acceleration first reaches 4 Gs. One engine on each of two boosters is also shut-down during two subsequent occurrences of attaining 4 Gs. Each booster pair is jettisoned when the propellant is depleted to the minimum reserve level. The payload shroud is jettisoned from the launch vehicle at a geodetic altitude of 122 Km (400,000 ft). At that altitude, the aerodynamic and aeroheating effects of the atmosphere are negligible. The vehicle then suborbitally ignites the TLI stage to inject into a 185 Km (100 nm) circular orbit. Booster and core impact points were calculated using an average ballistic coefficient. The vehicle has a launch azimuth range of 72 degrees to 108 degrees. The azimuth range provides any-day launch availability, weather permitting, with a day-dependent window of approximately 4 hours, while providing for capsule abort options. The same azimuth range was used in the Apollo Saturn V program. A ten percent mass contingency was included in the sizing and mass properties assessments of each vehicle element, with the exception of the F-1A engine, which presumed that use of a 1960-based technology would provide a similar conservatism.

4.1.2 Reference Vehicle

4.1.2.1 Vehicle Description and Performance Summary

The NLS family combines state-of-the-art technology (e.g., propulsion and avionics) while maximizing the use of current infrastructure (e.g., manufacturing, launch facilities, etc.). The core tankage of the NLS-1 (HLLV) and NLS-2 (50 Klbm payload) vehicles is derived from the Space Shuttle External Tank (ET). All core elements utilize the STME for main propulsion: the HLLV uses four STMEs, the NLS-2 vehicle uses six STMEs, and the NLS-3 (20 Klbm payload) vehicle uses one STME. The HLLV also uses two Advanced Solid Rocket Motors (ASRMs) for boost stage propulsion. The goal is to develop a robust, low-cost system that will meet NASA, Department of Defense (DoD), and commercial payload needs into the next century. The NLS-derived lunar vehicle further develops this theme by utilizing the NLS HLLV core stage, ET diameter TLI stage, and an SSME for an upper stage engine as the basic elements of the lunar launch system. Figure 4.1.2.1-1 illustrates this concept. The goal is to define a system that fulfills both the goals of SEI and the nation's other Earth-to-orbit needs such as Space Shuttle off-loading, DoD, and commercial payloads, etc. The configuration is composed of two core stage elements and four strap-on booster elements. The lunar configuration consists of the basic NLS core stage, a new TLI stage, and four boosters.

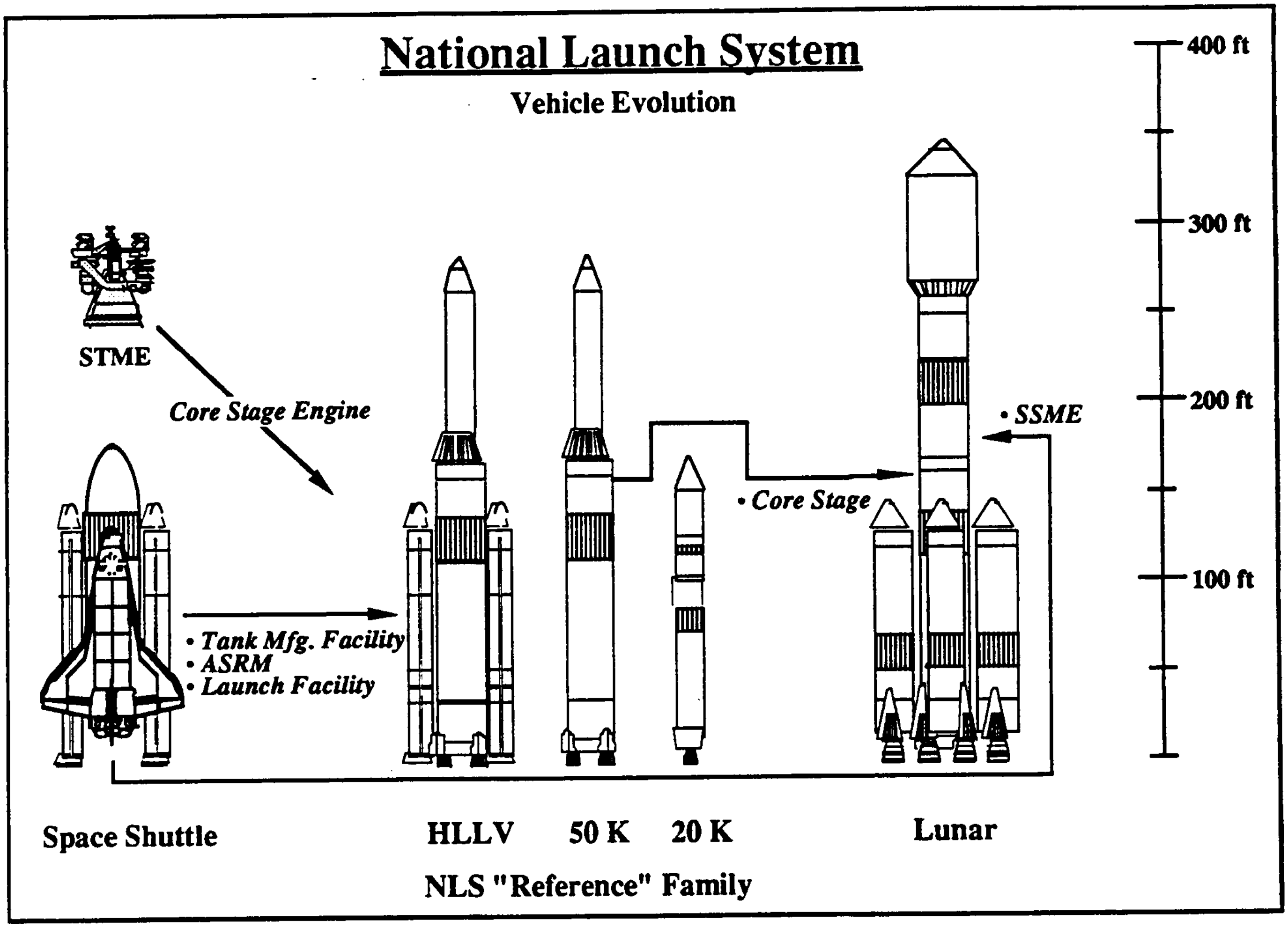
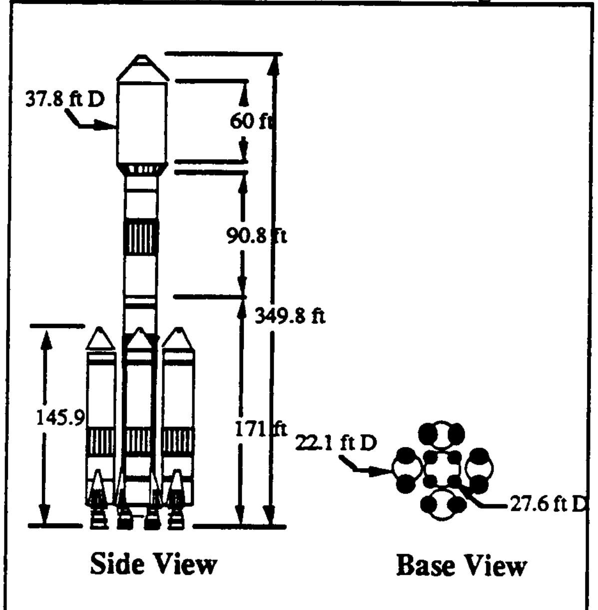


Figure 4.1.2.1-1 Evolution of NLS-Derived Concept

Based on the FLO requirements, constraints, technology assumptions, and selected configuration approach, numerous launch vehicle sizing optimization analyses were performed which led to the reference NLS-derived HLLV for the lunar mission. See Section 7.1 for information on other design options that were considered. Figure 4.1.2.1-2 summarizes the reference lunar mission configuration performance and major element mass properties.

Lunar NLS Derived HLLV w/ 4 LOx/RP Boosters

Single Launch - Cargo



Payload:	210 klb (95 t) / 586 klb
Final Position:	TLI / LEO Cutoff
GLOW:	12.4 Mlb
Engine Out:	None
CORE:	
Inert Mass:	195.7 klb
Propellant Mass:	1.69 Mlb
Propellant Type:	LOX/LH2
Engine Type/#:	STME/4
Vac /SL Thrust (Ea):	650/551 klb
Vac /SL ISP:	428.5/365 s
Engine Exit Dia:	92 in
Length:	171 ft
Diameter:	27.6 ft
Reusability:	None
BOOSTER:	
Number/Type:	4/New
Inert Mass:	166.5 klb
Propellant Mass:	2.2 Mlb
Propellant Type:	LOX/RP
Engine Type/#:	F-1A/2
Vac /SL Thrust (Ea):	2.02/1.8 Mlb
Vac ISP:	303.1 s
Engine Exit Dia:	144 in
Length:	146 ft
Diameter:	22.1 ft
Reusability:	None
TLI Stage:	
Inert Mass:	70.8 klb
Propellant Mass:	700 klb
Propellant Type:	LOX/LH2
Engine Type/#:	SSME/1
Vac Thrust (Ea):	489.9 klb
Vac ISP:	452.4 s
Engine Exit Dia:	90.5 in
Length:	90.8 ft
Diameter:	27.6 ft

Structure:	Al 2219
Shroud - Usable Volume:	33 x 60 ft
Mass:	28,240 lb

Notes:
- Core/Boosters are Ignited on Pad w/Booster Holddown
- ET prop. capacity (1.69 Mlb) based on a 5 ft stretch
- 108° Launch Azimuth
- F-1A & STME are 75% Step Throttlable
- Use Throttle/Booster Engine Shutdown for Loads
- Max G = 4.0 / Max q = 900 psf

Figure 4.1.2.1-2 NLS-Derived Reference Configuration Specifications Summary

4.1.2.2 Booster Element

The booster diameter of 6.7 m (22.1 ft), and length of 44.4 m (145.9 ft) including aerodynamic nose cone, were derived by constraining the booster length to match the NLS Core Stage attach point locations, which are the same as those for the Space Shuttle ET/SRBs. The LOX/RP boosters are configured so that loads are transmitted through a thrust beam in the core intertank into the booster forward adapter, which is also similar to the current Space Shuttle design. Aerodynamic fairings have been placed on the aft skirt of each booster to protect the engines from ascent loads. Core and booster LOX tanks have been placed forward of the fuel tanks to move the vehicle center-of-gravity forward and therefore improve aerodynamic stability and control. Figure 4.1.2.2-1 shows the internal layout and dimensions of the boosters with respect to the core vehicle and TLI stage. The thrust vector control (TVC) subsystem was chosen to be the same as that used on the Saturn V S-IC stage, in which RP-1 fuel is bled off of a high pressure discharge port on the F-1A fuel turbopump and used to power hydraulic actuators. A more detailed trade study remains to be performed on the use of alternative TVC concepts such as electrohydrostatic (EHA) and electromechanical actuators (EMAs).

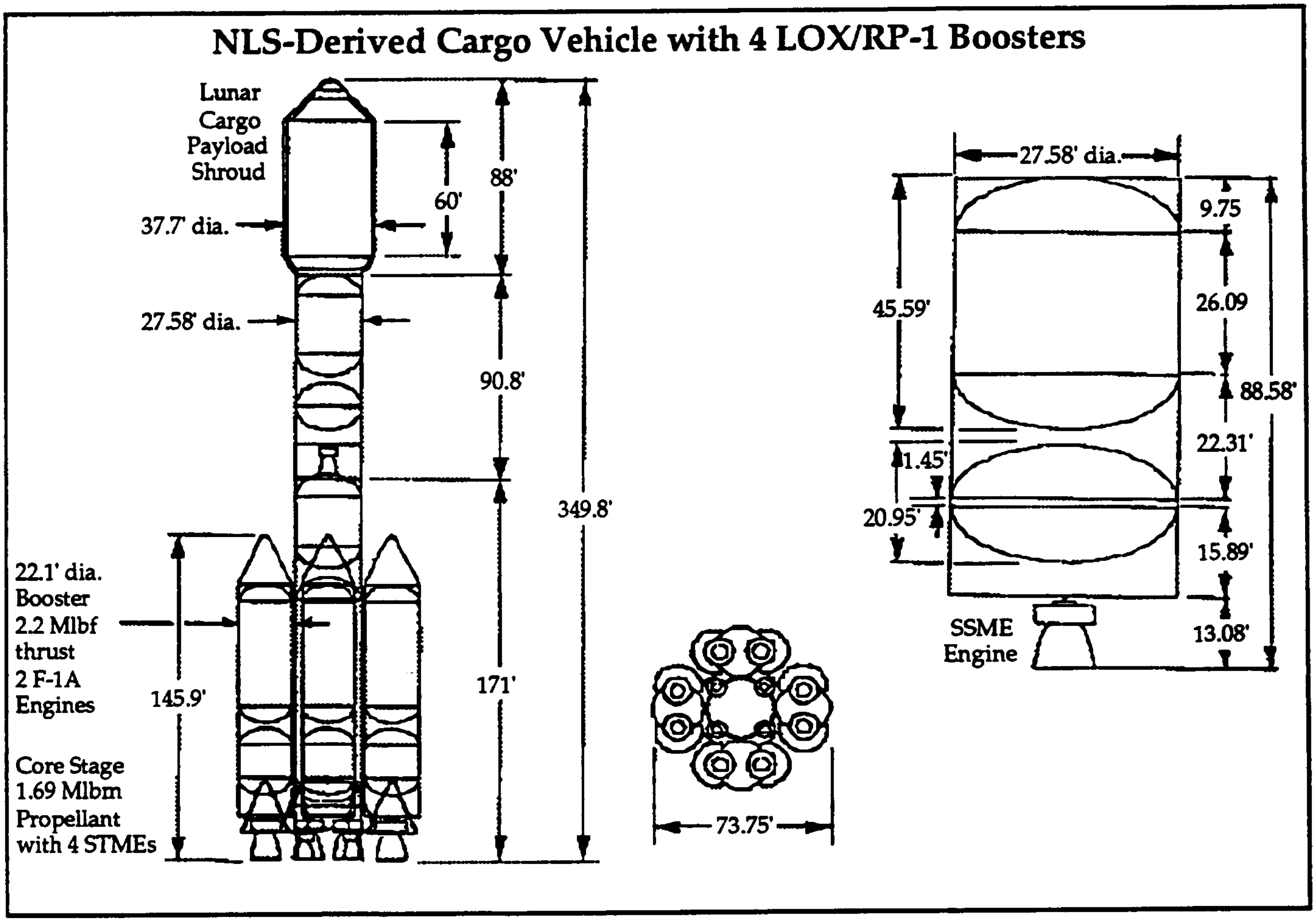


Figure 4.1.2.2-1 NLS-Derived Cargo Vehicle Internal Layout and Dimensions

The LOX/RP-1 booster mass summary for the NLS-derived HLLV is shown in Figure 4.1.2.2-2. These masses were derived from the Saturn V S-1C stage and the Space Shuttle SRB. The stage masses were derived from the S-1C stage with updates for only two F-1A engines and the reduced diameter and usable propellant capacity of 998 t (2.2 Mlbm). The attachment and separation system masses were scaled from the Space Shuttle Solid Rocket Booster (SRB). The total dry mass is shown for only one booster as the other three boosters are identical. The unusable residuals were added to the dry mass to give the minimum burnout mass for one booster.

NLS LUNAR HLLV SATURN V-DERIVED LOX/RP1 BOOSTER WITH 2 F-1A ENGINES (All Values Shown as Pounds Mass)	
FWD STRUCTURES AND NOSE CONE	3,630
ATTACH AND SEPARATION	2,327
LOX TANK	18,027
INTERTANK	5,926
RP-1 TANK	10,836
AFT STRUCTURES	11,149
THRUST STRUCTURE AND HOLD-DOWN	11,465
MAIN ENGINES (2 F-1As)	38,000
BASE HEAT SHIELD	2,590
LOX SYSTEM	11,165
RP-1 SYSTEM	6,308
TVC	4,069
AVIONICS	1,050
CONTINENCY 10% *	8,854
TOTAL DRY MASS	135,396
RESIDUALS	31,262
TOTAL BURNOUT MASS	166,658 lbm
USABLE PROPELLANTS = 2,200,000 lbm STAGE DIAMETER = 265 INCHES	

* Not applied to the engines, which are an existing design

Figure 4.1.2.2-2 NLS-Derived Booster Mass Properties Summary

4.1.2.3 Core Stage I Element

The core stage has the same diameter as the Space Shuttle Program (SSP) External Tank (ET), 8.39 m (27.58 ft), with several additional changes: the ogive LOX tank has been replaced with an elliptical endcap and cylindrical section arrangement, the LH_2 tank has been stretched five feet to accommodate the 408.2 t (900 Klbm) increase in propellant load over the basic 766.6 t (1.69 Mlbm) capacity, the LOX tank stretched a corresponding amount for the 6:1 mixture ratio, and tank structural modifications have been made as necessary (see Section Analysis Section-Structures). The SSP SRB structural attach point height has been retained. This results in an overall core height of 52 m(171 ft). Figure 4.1.2.2-1 shows the internal layout and dimensions of the core stage with respect to the boosters and TLI stage.

The core stage mass summary for the NLS Lunar HLLV configuration is shown in Figure 4.1.2.3-1. The mass properties were derived using the MSFC NLS ET reference masses and updating to account for the reduced loads produced by the HLLV-only configuration, and the increased loads produced by the TLI stage, increased payload mass, increased shroud mass, and the four boosters. The propulsion system masses were revised to accommodate the 2.89 MN vacuum thrust (650 Klbf) STME. The avionics masses were revised for the lunar configuration and mission. A ten percent mass contingency was applied to all systems, including new and modified systems, and is shown as a separate entry in Figure 4.1.2.3-1. The minimum burnout mass for the core stage includes the total dry mass for the core stage and the unusable residuals, which do not include any usable reserves. The total usable propellant capacity is approximately 768 t (1,693 Klbm) for the stage, which utilizes a 8.4 m (331 in) diameter derived from the Space Shuttle External Tank (ET).

NLS LUNAR HLLV
ET DERIVED CORE STAGE WITH 4 650K STMEs
(All Values Shown as Pounds Mass)

INTERSTAGE	11,018
FORWARD STRUCTURES	5,508
LOX TANK	16,104
INTERTANK (INC 2 CROSSBEAMS)	17,304
LH2 TANK	27,703
TPS,HEAT SHIELD,ENVIRONMENTAL CONTROL	6,447
MAIN ENGINES (4 650K STMEs)	35,460
PROPELLANT FEED SYSTEM	10,465
PNEUMATIC SYSTEM	3,229
SUBSYSTEM INSTALLATION STRUCTURE	3,688
THRUST STRUCTURE & MECHANICAL	18,866
TVC	3,129
ATTACH & SEPARATION SYSTEM	2,174
AVIONICS	1,800
CONTINGENCY 10%	16,290
TOTAL DRY MASS	179,185
RESIDUALS	16,474
TOTAL BURNOUT MASS	195,659 lbm

USABLE PROPELLANTS = 1,693,000 lbm   STAGE DIAMETER = 331 INCHES

Figure 4.1.2.3-1 NLS-Derived Core Mass Properties Summary

4.1.2.4 TLI Stage Element

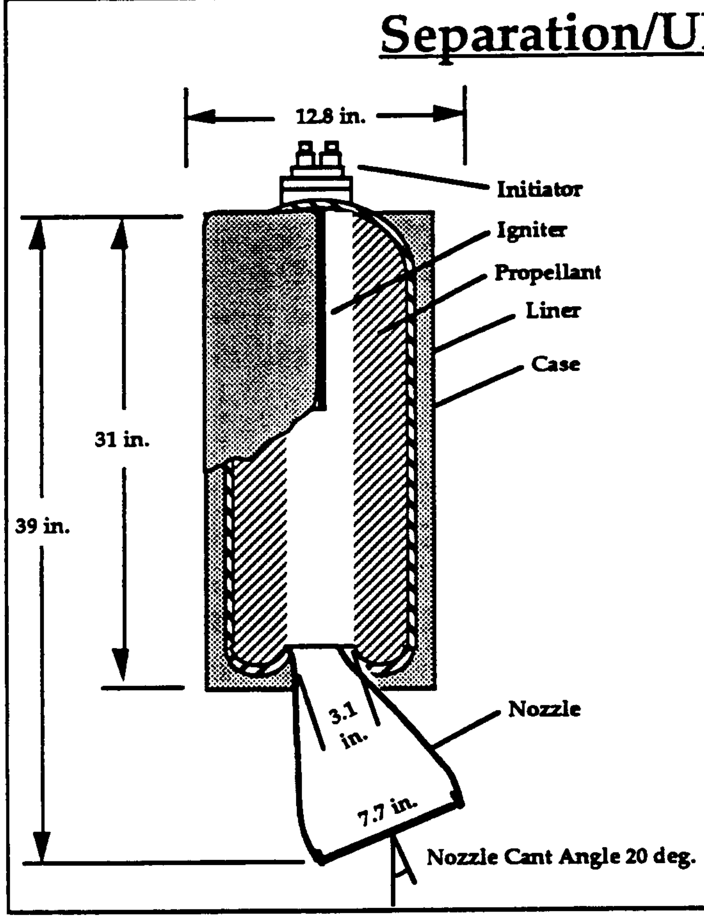
The LOX/LH_2 TLI stage is placed in-line with the core and has a common diameter of 8.39 m (27.58 ft). The stage has a propellant capacity of 317.5 t (700 Klbm), which results in an overall stage length of 26.9 m (88.58 ft). The LOX tank is placed aft for this stage to improve stability during non-atmospheric maneuvers. This results in a lighter overall stage weight since the heavier LOX tank does not have to be supported by the fuel tank. Shorter LOX lines are another benefit. The ET diameter of 8.4 m (27.58 ft) and the S-IVB stage tank arrangement of having the LOX tank aft and the LH_2 tank forward were also utilized. Both separate and common bulkhead tank configurations were studied, but the separate tank design was used as the reference. The TLI stage uses conventional Aluminum 2219 for all structural and tankage components. The intertank and forward and aft skirts use a standard skin-stringer design. The stage design may support the use of existing Shuttle ET tooling. Figure 4.1.2.2-1 shows the internal layout and dimensions of the TLI stage with respect to the boosters and core stage.

A preliminary assessment of TLI propellant tank thermal control requirements resulted in the baseline design of 5 cm (2 in) thick spray-on foam insulation (SOFI) for both the LOX and LH_2, which would result in no more than a 1 percent per hour LH_2 boil-off rate for on-orbit stay times of less than five hours. No technological advances would be required for the reference TLI stage thermal control methodology. A more detailed discussion of the SOFI design may be found in Section 7.1.4.

The reference RCS design consists of ten 445 N (100 lbf) thrust bipropellant engines. The propellant combination is monomethyl hydrazine and nitrogen tetroxide, with a helium diaphragm pressurization system. The resulting engine performance yields a specific impulse of 300 seconds at an inlet pressure of 1,724 Kpascal (250 psia) and a mixture ratio of 1.65:1. The required usable propellant quantity, which include a 25 percent contingency reserve, are included with the unusable residuals which may all be left on the stage at burnout. The total RCS propellant budget is 323 Kg (713 lbm) and the dry system mass is 86 Kg (190 lbm).

Four solid motors, similar to the one shown in Figure 4.1.2.4-1, are used to move the TLI stage away from the spent NLS core during TLI separation. The separation motors will help to provide a small positive acceleration that settles the TLI propellants prior to the suborbital burn, and will also be used for a settling burn prior to the final TLI burn.

The TLI stage mass summary for the NLS Lunar HLLV configuration is shown in Figure 4.1.2.4-2. The mass properties were derived from the NLS reference masses, the Saturn S-IVB stage, and the Space Shuttle Orbiter. The stage masses were estimated to account for the loads produced by the combination of the 317.5 t (700 Klbm) propellant capacity, payload, shroud, and the thrust of one SSME. The Thermal Protection System (TPS) mass allowance was scaled from the S-IVB stage which should be adequate, but a new TPS will have to be developed (see Analysis Section - Thermal). A micro-meteoroid shield mass was estimated which could be integrated with the TPS into one system. The propulsion system masses were derived from the Orbiter propulsion system. The SSME, ancillary systems, auxiliary power system, and hydraulic power equipment could be used with minimum or no modification for the TLI stage. The reaction control system (RCS) masses were included for roll control during powered flight of the single-engine TLI stage and for control during on-orbit stay time. A ten percent mass contingency is included as a separate entry on everything except the SSME and ancillary systems (which are existing hardware). The total dry mass for the TLI stage, RCS propellant, and unusable residuals (which do not include reserves) were added to define the stage burnout mass.



Thrust (lbf) vacuum	34,000
Specific Impulse (sec) vacuum	242.5
Chamber Pressure (psia)	2221
Expansion Ratio	5.8
Mass (lbm)	656
Propellant	HTPB
Action Time (sec)	3.7

Figure 4.1.2.4-1 NLS-Derived TLI Stage Separation/Ullage Motor

NLS LUNAR HLLV ET DERIVED TLI STAGE WITH 1 SSME (All Values Shown as Pounds Mass)	
TRANSITION STRUCTURE	3,800
FORWARD STRUCTURE	2,710
LH_2 TANK	11,344
INTERTANK	10,025
LOX TANK	7,003
AFT STRUCTURE	4,118
TPS, MICROMETEOROID SHIELD, ENGINE PROTECTION	4,315
MAIN ENGINE (1 SSME)	6,956
THRUST STRUCTURE	2,043
PROPELLANT FEED SYSTEM	2,628
ANCILLARY SYSTEMS	1,830
INSTALLATION STRUCTURE ,APU,HYDRAULICS	1,334
RCS (ON-ORBIT & ROLL CONTROL)	897
AVIONICS	2,200
CONTINGENCY 10% *	5,241
TOTAL DRY MASS	66,441
RESIDUALS	3,782
TOTAL BURNOUT MASS	70,223 lbm

USABLE PROPELLANTS = 700,000 lbm   STAGE DIAMETER = 331 INCHES
*Does not include SSME & ancilliary, since existing designs

Figure 4.1.2.4-2 NLS-Derived TLI Stage Mass Properties Summary

4.1.3 Aerodynamics

The distributed aerodynamic loads are presented in the form of dimensionless coefficients that are a function of Mach number and core vehicle body station location (measured as the ratio of body station to body diameter, X/D): dCA/d(X/D) and dCNa/d(X/D). By integrating these coefficients over a range of X/D and multiplying by dynamic pressure and reference area (and by angle of attack for normal force), the load acting at the middle of the selected range was computed. The coefficients were computed at Mach 1.5, corresponding to the typical occurrence of maximum dynamic pressure. These data show the center of pressure (CP) of the core-alone configuration at an X/D of 10.87, though the integrated effect of the boosters was to move the CP aft to the X/D of 5.80. There was a discrepancy between the center of pressure calculated by this method and the value calculated from the total normal and pitching moment coefficients. This was presumed to be due to an incorrect adjustment of the wind tunnel data for overall vehicle length. The length of the wind tunnel model was 10.2 diameters while the NLS-TLI vehicle was 13.2 diameters (as assumed in these calculations). The CP being further forward is more conservative for loads estimation. This underscores the importance of wind tunnel data using accurate models to enable more precise estimation.

4.1.4 Stability and Control

During first stage, the strap-on boosters provide control authority for launch vehicle steering. Prior to booster engine cut-off and separation, control authority is transferred to the core vehicle. Based on past operational experience, the core vehicle would be commanded to maintain an attitude hold during booster separation to minimize any attitude and rate transients, and to allow the core vehicle's guidance software to reconverge onto a new guidance solution. A trade study needs to be performed to determine if the core vehicle will have sufficient control authority during first stage to perform thrust vector control steering. While such a capability will probably require an up-sizing of the core vehicle's thrust vector control (TVC) subsystems, it will also allow a significant dry mass savings by removing all TVC hardware from each booster.

A stability analysis will be performed for the four-booster configuration to determine if tail fins will be required on the boosters or core vehicle to provide directional stability for crosswind, load relief (non-zero angle-of-attack and sideslip), and propulsion dispersion conditions during atmospheric flight. The presence of the boosters will help to move the aerodynamic center of pressure aft on the mated vehicle in both the pitch and yaw planes, but NASA standards for stability and control design require the vehicle to also be stable in the presence of atmospheric and performance dispersions.

4.1.5 Manufacturing Facilities and Tooling

Figure 4.1.5-1 shows the new and modified manufacturing facility tooling components and facility requirements needed to support SSP, NLS, and NLS-derived SEI vehicle component manufacturing.

Requirements for New and Modified Manufacturing Facilities and Tooling

	Shuttle ET	NLS Core	SEI NLS-Derived Core	SEI Booster	SEI TLI Stage
Quantity Manufactured Per Year	8	8	4	16	4

Processing Function/ Vehicle Element	Tooling	Facilities
Domes	Set of 22.1 ft. Dome Tools & TLI Stage	500,000 sq.ft New Area & New Storage Building
Barrels	New Fixture (20-30 ft) & 22.1 ft Diameter	Utilities in New Building
Rings	New Fixtures & Tables	Increased Floor Area
Intertank	New 22.1 ft & TLI Intertank Assembly Fixture	Increased Floor Area Utilities
Major Weld	22.1 ft Diameter and TLI Fixtures	Weld Area Rearrangement in New Building
Clean & Thermal Protection System	New Tank Configurations (Lengths & Diameters)	Cell Length Modifications
Final Assembly	New Positions & Transporters	Cell Length Modifications
Thrust Structure & Nosecone	New Structure	Increased Floor Area
Skirts	New Skirts	Increased Floor Area
Propulsion Module	Not Applicable	New Building
Interstage	2 Modified Interstages	Increased Floor Area
Avionics	Not Applicable	Increased Floor Area
Integrated Assembly & Check-Out	Stack of 2 New Vehicles	Increased Stack, Assembly, & Check-out Areas

Figure 4.1.5-1 NLS-Derived Manufacturing Facilities & Tooling

It was assumed that all assembly would be performed at NASA's Michoud Assembly Facility (MAF). Major manufacturing elements include the ET, baseline NLS core vehicle, SEI core vehicle, SEI boosters, and SEI TLI stage. MAF excess tooling, along with additional new tooling, will give MAF the capability of manufacturing the 40 vehicle elements required to effectively support SSP, NLS, and SEI program requirements. New tooling, increased floor areas, integration cell modifications, new storage buildings, and enlarged assembly and check-out area requirements are primarily driven by flight rate and degree of design changes, such as booster diameter, over the current SSP element designs. SEI manufacturing tools and techniques utilize ET manufacturing technologies and processes. Commonality in propellant tank endcap design between the three prospective mixed fleet programs will facilitate maximum manufacturability.

4.1.6 Schedules

Figures 4.1.6-1, 4.1.6-2, and 4.1.6-3 display preliminary schedules for the development and acquisition phases of an NLS-derived core vehicle, boosters, and TLI stage, respectively. The major features of the schedules are a two-year in-house preliminary definition study, immediately followed by a five-year Phase C/D, beginning in early Fiscal Year (FY) 1995. Initiation of the preliminary definition studies in the last quarter of FY 1992 would be necessary to accommodate a launch in 1999. These schedules also show estimates for long lead item procurement and fabrication requirements for the major NLS-derived HLLV subsystems.

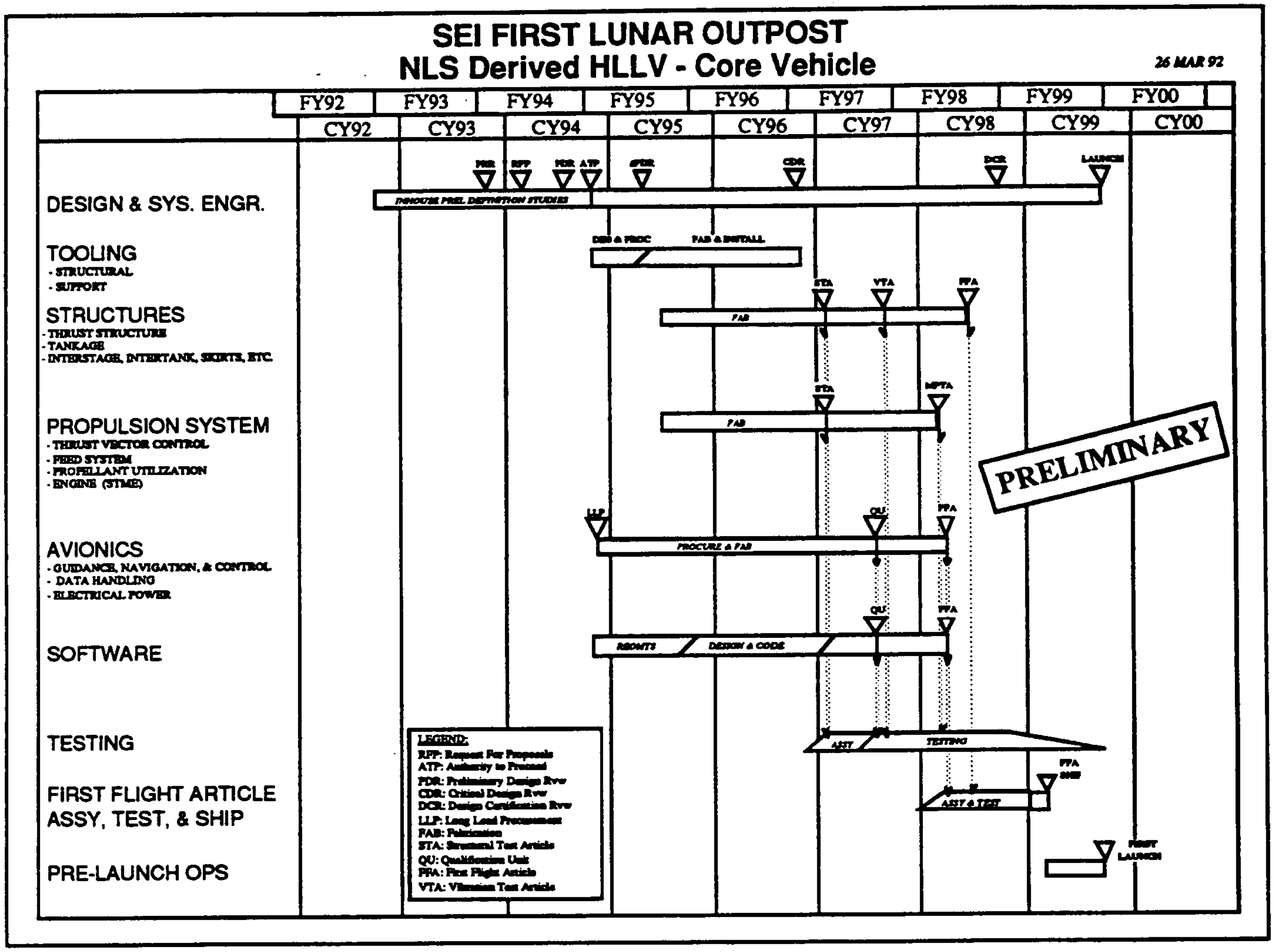


Figure 4.1.6-1 NLS-Derived Core Development & Acquisition Schedule

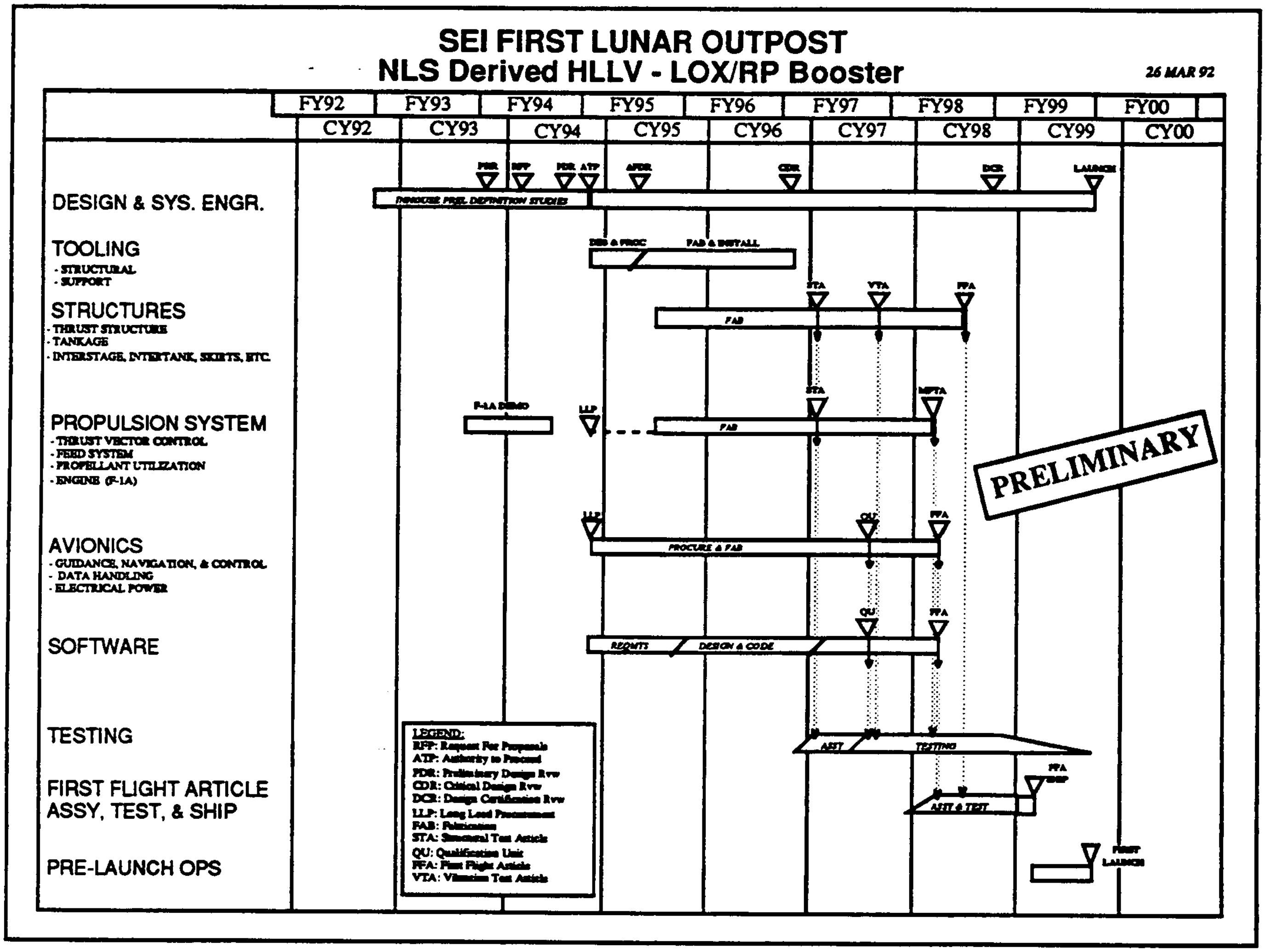


Figure 4.1.6-2 NLS-Derived Booster Development & Acquisition Schedule

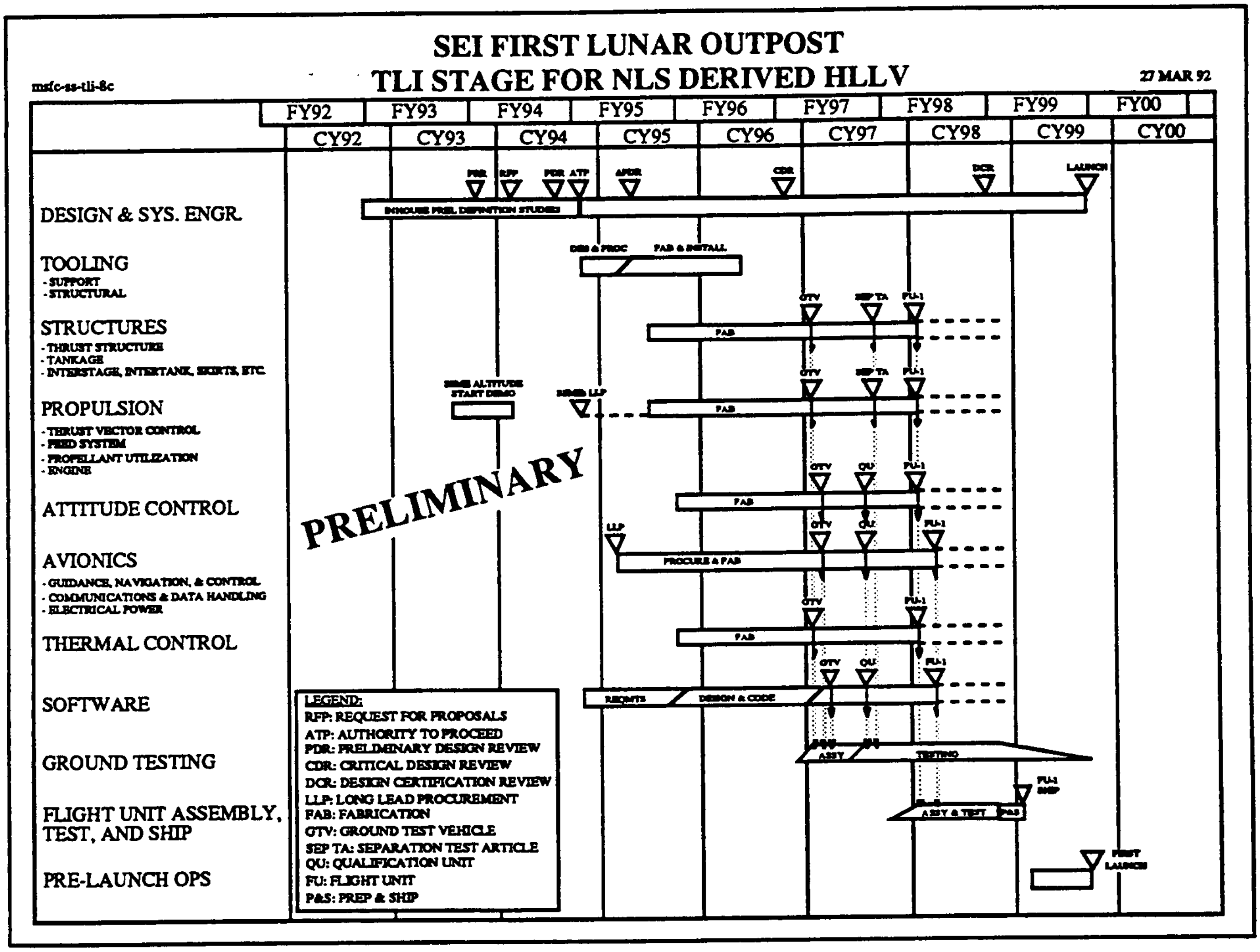


Figure 4.1.6-3 NLS-Derived TLI Stage Development & Acquisition Schedule

4.2 Saturn Derived

The following sections present the results of the definition and assessment of the reference Saturn derived launch vehicle configuration.

4.2.1 Mission Profile

Figure 4.2.1-1 illustrates the mission profile during ascent. The profile is the same for both piloted and cargo flights except that a launch escape system (LES) is included and jettisoned at shroud separation in piloted missions. The on-orbit mission profile is similar to that for the NLS-derived configurations, as shown in Figure 4.1.1-2.

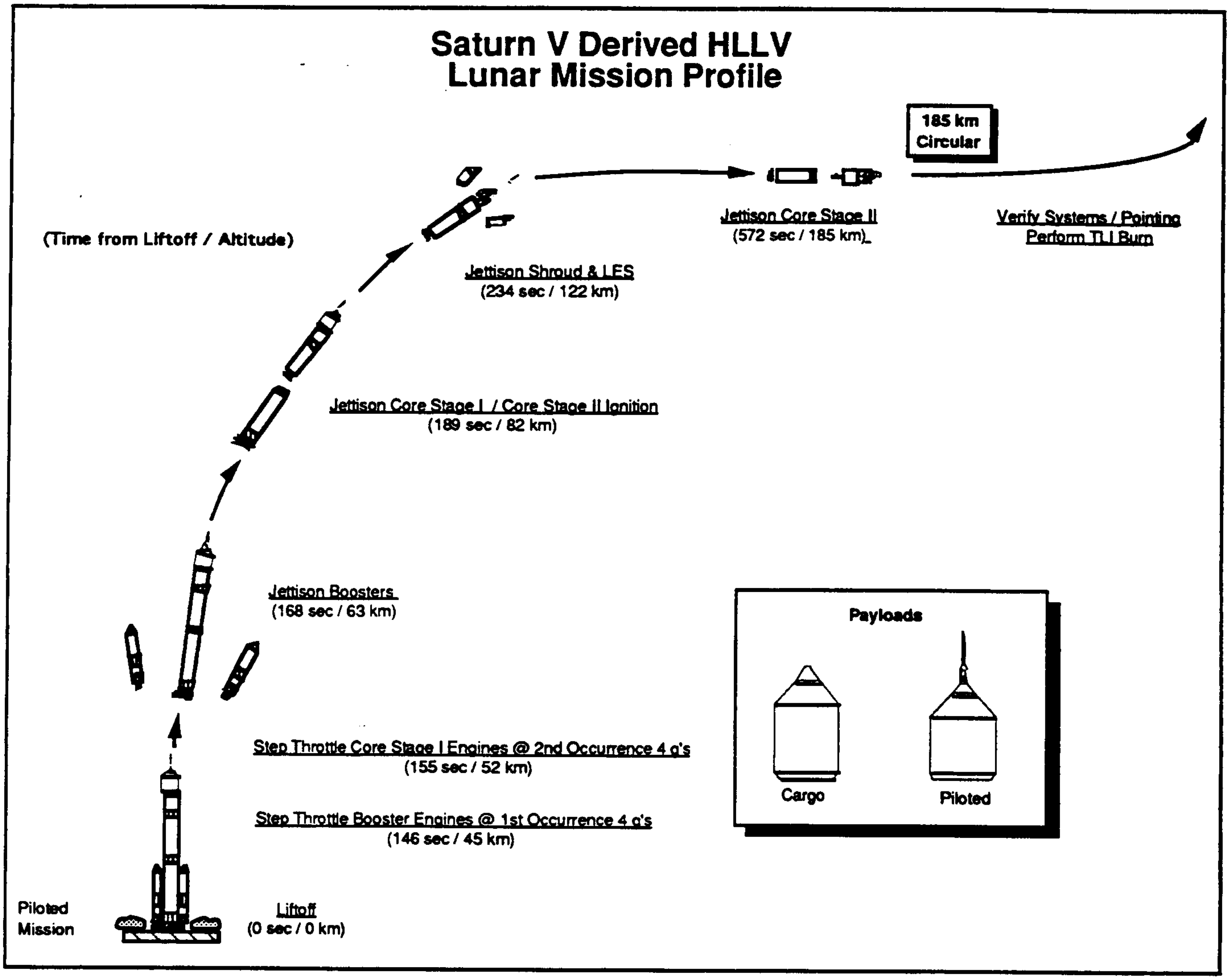


Figure 4.2.1-1 Saturn V-Derived Ascent Mission Profile

Liftoff occurs with the nine booster and core stage F-1A engines operating at 100 percent RPL. A vertical rise maneuver is maintained through tower clearance and is followed by a pitch-over maneuver. From this point, an optimized ascent profile is flown subject to a a 43.1 K N/m^2 (900 psf) maximum dynamic pressure constraint. Ascent acceleration limits are maintained through the use of a throttling sequence with both the boosters and the modified S-IC stage. At the first occurrence of a 4 Gs sensed acceleration level, the booster engines are step-throttled to 75 percent RPL. At the second occurrence of 4 Gs acceleration, the modified S-IC stage engines are step-throttled to 75 percent RPL and maintained for the duration of the burn. At booster propellant depletion, the boosters are jettisoned from the core. The next ascent event occurs when the modified S-IC stage propellant is depleted and the stage is jettisoned. The six J-2S engines of the modified S-II stage are then ignited and operated at full throttle throughout the entire burn sequence. Sensed acceleration never exceeds the established 4 Gs limit during second stage operation. When the vehicle reaches a geodetic altitude of 122 km (400,000 ft), the payload shroud and the launch escape system (LES) are sequentially jettisoned. Insertion of the TLI stage and payload into a 185 km (100 nautical mile) circular orbit is completed using the modified S-II stage, at which point the stage is jettisoned. The TLI stage and payload systems are then checked out in low Earth-orbit. The vehicle is then maneuvered into the proper TLI burn attitude and pointing verification is performed. If no malfunctions are detected, the TLI burn is performed by the TLI stage with its one J-2S engine. If a problem is detected at this point during a piloted mission, the mission is aborted and the crew returned to Earth.

4.2.2 Reference Vehicle

4.2.2.1 Vehicle Description and Performance Summary

The Saturn V-derived launch vehicle option was developed to assess the capability and cost effectiveness of a vehicle employing Saturn V design characteristics, propulsion technology, and proven manufacturing capability. A primary objective of this approach was to minimize vehicle development costs.

A basic Saturn V-derived configuration was selected after consideration of the requirements and constraints, and evaluation of design modularity objectives. The lunar configuration consists of three core stage elements, including stretched S-IC and S-II stages and a new TLI stage, and two booster elements. Based on the FLO requirements, constraints, technology assumptions, and selected configuration approach, numerous launch vehicle sizing optimization analyses were performed. The sizing analyses produced several hundred vehicles from which selection was made of the reference Saturn V-derived, lunar heavy-lift launch vehicle (HLLV) depicted in Figure 4.2.2.1-1. The reference configuration is compared to the Saturn V launch vehicle in Figure 4.2.2.1-2.

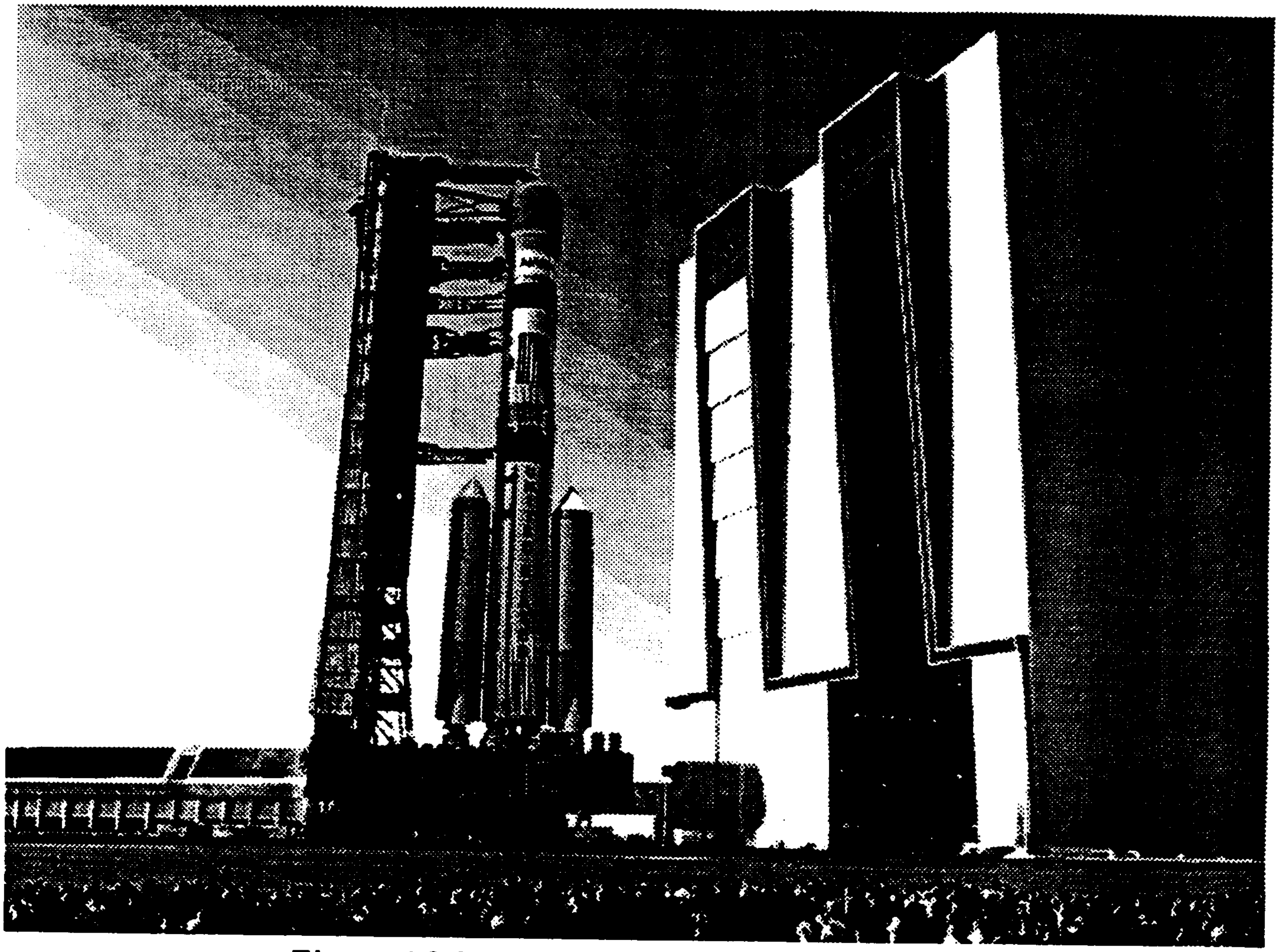


Figure 4.2.2.1-1 Saturn V-Derived Concept

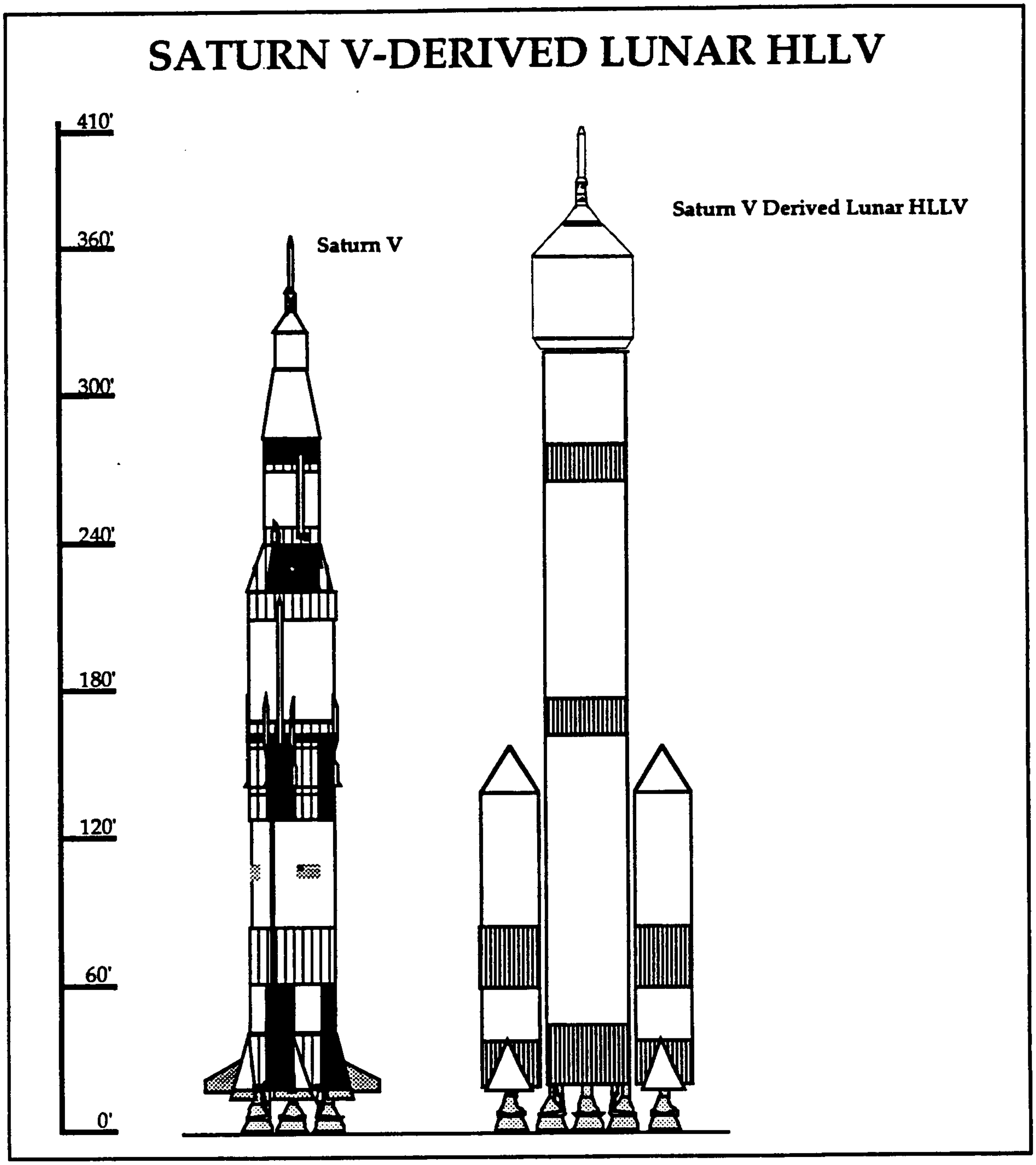


Figure 4.2.2.1-2 Comparison of Saturn V and Saturn V-Derived Vehicles

The reference Saturn-V derived lunar vehicle is illustrated for the piloted configuration in Figure 4.2.2.1-3. The vehicle and stage characteristics were defined through an integrated sizing/optimization process to derive maximum performance capability, subject to the HLLV groundrules and constraints. A 10 percent contingency factor was applied to all stage element dry mass estimates for growth margin. The vehicle was sized to an overall height of 125 m (410 feet), as limited by the desire to utilize the existing VAB facility. Vehicle performance capability was estimated to be 97.6 t (215 Klbm) post-TLI payload, after insertion into a 185 Km (100 nm) circular Earth-orbit. A 72 degree launch azimuth was found to be the most performance constraining. The maximum ascent acceleration limit of 4.0 Gs was satisfied with step throttling control along a 43.1 K N/m^2 (900 psf) maximum dynamic pressure trajectory. The reference vehicle and stage element data presented on the following pages represent the results of subsystem mass properties build-ups and ascent performance analyses for the down-selected configuration. Detailed studies to assess load distributions and ascent stability and control requirements have not been performed.

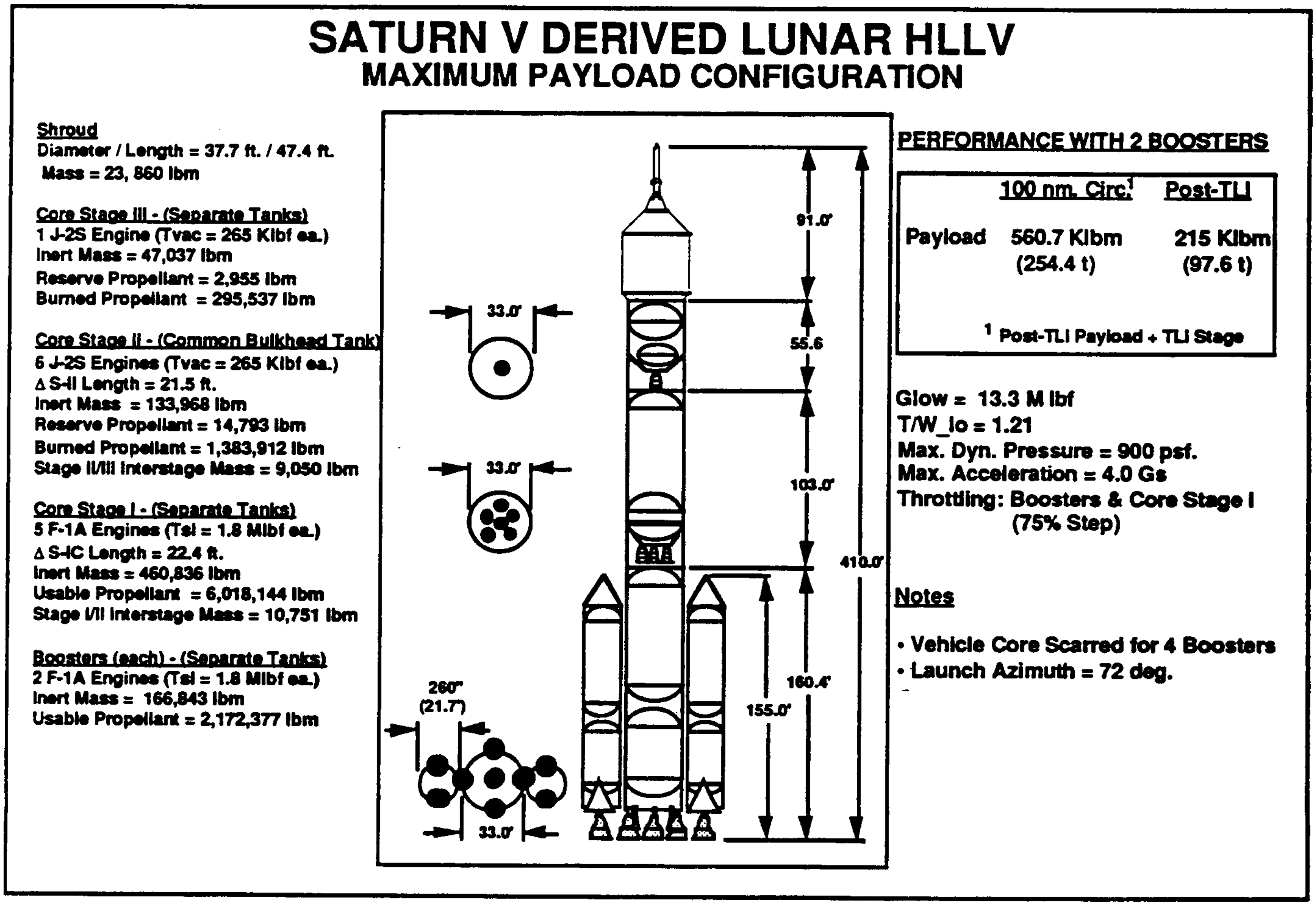


Figure 4.2.2.1-3 Saturn V-Derived Reference Configuration Specification Summary

The configuration consists of two boosters and three core stage elements of diameter 10 m (33 ft): a stretched S-IC, a stretched S-II, and a new TLI stage. The booster elements provide thrust augmentation for a total vehicle lift-off thrust of 72 million N (16.2 Mlbf), yielding a lift-off thrust-to-weight ratio of 1.21. Each booster element provides 16 million N (3.6 Mlbf) sea-level thrust, utilizing two F-1A engines, and had a total fueled mass of 1,061 t (2.34 Mlbm). The boosters are configured with separate RP-1 and LOX tanks providing a total usable propellant mass of 985 t (2.17 Mlbm). The booster diameter is 6.6 m (21.7 feet).

Core stage I is a modified S-IC, stretched 6.8 m (22.4 ft), providing an RP-1/LOX propellant mass of 2,730 t (6.018 Mlbm). Five F-1A engines deliver a total stage sea-level thrust of 40 million N (9 Mlbf). Mass properties estimates were incorporated to reflect the significant stage structural modifications required for the increased thrust loading of the F-1A engines and for the loads imposed by the booster elements onto the S-IC stage. The stage mass has also been scarred with additional structure required for a four-booster Mars configuration. The effects of the stage stretch and structural enhancements increase the stage dry mass approximately 39 percent with respect to the S-IC. The total fueled stage mass is 2,944 t (6.49 Mlbm). Core Stage II is a modified S-II stretched 6.5 m (21 feet), and provides an LH_2/LOX propellant mass of 635 t (1.4 Mlbm). Approximately 6.7 t (14.8 Klbm) of reserve ascent propellants are included. The S-II common bulkhead tank configuration has been retained in the interest of minimizing the overall vehicle height. Six J-2S engines, as compared to five J-2s on the original S-II, provide a total stage vacuum thrust of 7.1 million N (1.59 Mlbf). The increased stage length, additional engine, and structural modifications required for the additional thrust loads result in an increase to the S-II dry mass of approximately 42 percent. The total fueled stage mass is 700 t (1.54 Mlbm). Core stage III is a new stage element which was designed for the TLI maneuver, and was not used suborbitally during ascent. The TLI stage is 17 m (56 feet) in length and utilizes separate LH_2 and LOX tanks that provide a usable propellant mass of 135 t (298 Klbm). A single J-2S engine is used for main propulsion. The estimated total vehicle gross lift-off mass is 6,033 t (13.3 Mlbm).

Ascent trajectory analyses were performed for all candidate vehicle configurations which were down-selected from the sizing/optimization process, in order to verify lunar mission payload capability objectives and to ensure satisfaction of the ascent constraints. Configurations which met all objectives were carried through to final selection. Those configurations not meeting the payload objectives or study groundrules were either refined through subsequent sizing iterations or eliminated from consideration.

The ascent performance analyses conducted during this phase of study were three degree-of-freedom trajectory simulations. Analyses to assess stability and control requirements of the reference vehicles are to be addressed in later phases of study. The trajectory simulations are performed from a Kennedy Space Center (KSC) launch site and utilize a 1963 Patrick Air Force Base atmosphere model. The launch vehicle configuration and mass properties, defined during the sizing process, along with the propulsion system specifications for the F-1A and J-2S engines were used to simulate the vehicle characteristics.

Trajectory simulation event sequences are modeled after the mission profile previously described. After tower clearance, the vehicle pitch-plane steering profile was optimized through iterative trajectory evaluations to define the maximum vehicle payload capability, subject to two primary ascent performance constraints. The constraints to be satisfied are a maximum dynamic pressure level less than or equal to 43.1 K N/m^2 (900 psf) and a 4 Gs maximum acceleration level. No groundrules were imposed for maximum Q-alpha constraints. Lunar HLLV trajectory simulations were performed from a worst case launch azimuth to ensure that payload requirements would be satisfied from any azimuth within the required capability range of 72 to 108 degrees.

Figures 4.2.2.1-4, 4.2.2.1-5, 4.2.2.1-6, and 4.2.2.1-7 present a summary of the geodetic altitude, dynamic pressure, acceleration, and Earth-relative velocity profiles that were generated from the ascent trajectory simulation for the reference lunar HLLV. The optimized trajectory profile achieves a peak dynamic pressure of 43.1 K N/m^2 (900 psf), the upper constraint limit, at approximately 85 seconds into ascent. No engine throttling is required for dynamic pressure control. The maximum acceleration constraint is satisfied during ascent through the use of a dual throttling sequence with the booster and modified S-IC stages. At a trajectory simulation time of approximately 146 seconds, the first occurrence of a 4 Gs acceleration level is encountered and the four F-1A engines of the boosters are simultaneously step-throttled to 75 percent RPL for acceleration control. The booster engines remain at this power setting for the duration of their burn sequence. At a simulation time of approximately 155 seconds, prior to booster staging, a 4 Gs acceleration level is encountered for the second time at which point the five F-1A engines of the modified S-IC stage are simultaneously step-throttled to 75 percent RPL to maintain acceleration control. The five F-1A engines remain at this power setting for the duration of the stage burn. Booster propellant depletion occurs at a simulation time of approximately 168 seconds, at which point the booster staging event occurs. As illustrated in Figure 4.2.2.1-5, the vehicle achieves a 4 Gs acceleration level for a third time, just prior to the booster staging event. Burnout and jettison of the modified S-IC stage occur next, at approximately 189 seconds into ascent. Ignition of the six J-2S engines on the modified S-II stage follows, and all engines are operated at a 100 percent RPL throughout the duration of ascent. At an altitude of 122 km (400,000 ft), which is attained at approximately 234 seconds into ascent, the launch escape system (LES), on piloted missions, and payload shroud are sequentially jettisoned. A performance sensitivity analysis against LES jettison time indicated that the impact of carrying the LES to shroud jettison altitude was not significant. The ascent sequence is complete with shutdown and jettison of the modified S-II stage, when the orbital insertion targets for the 185 km (100 nm) orbit are attained at a simulation time of approximately 572 seconds.

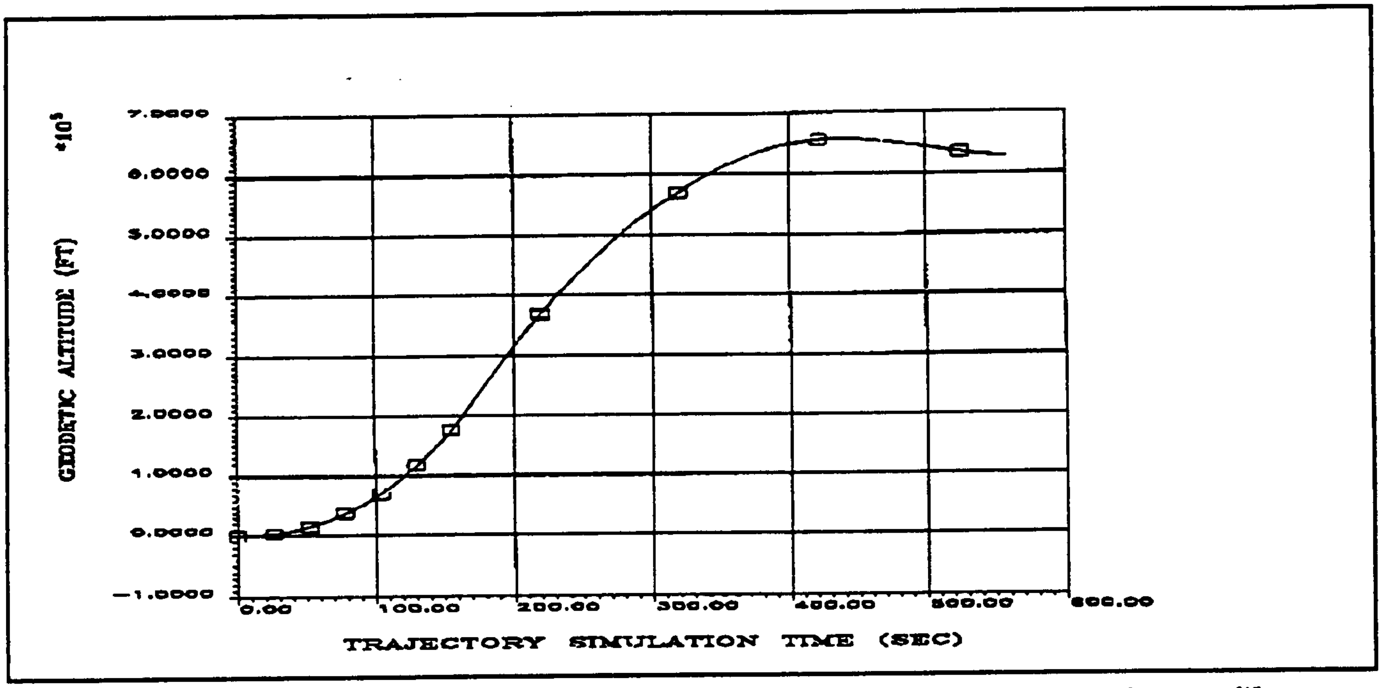


Figure 4.2.2.1-4 Reference Ascent Trajectory Geodetic Altitude Profile

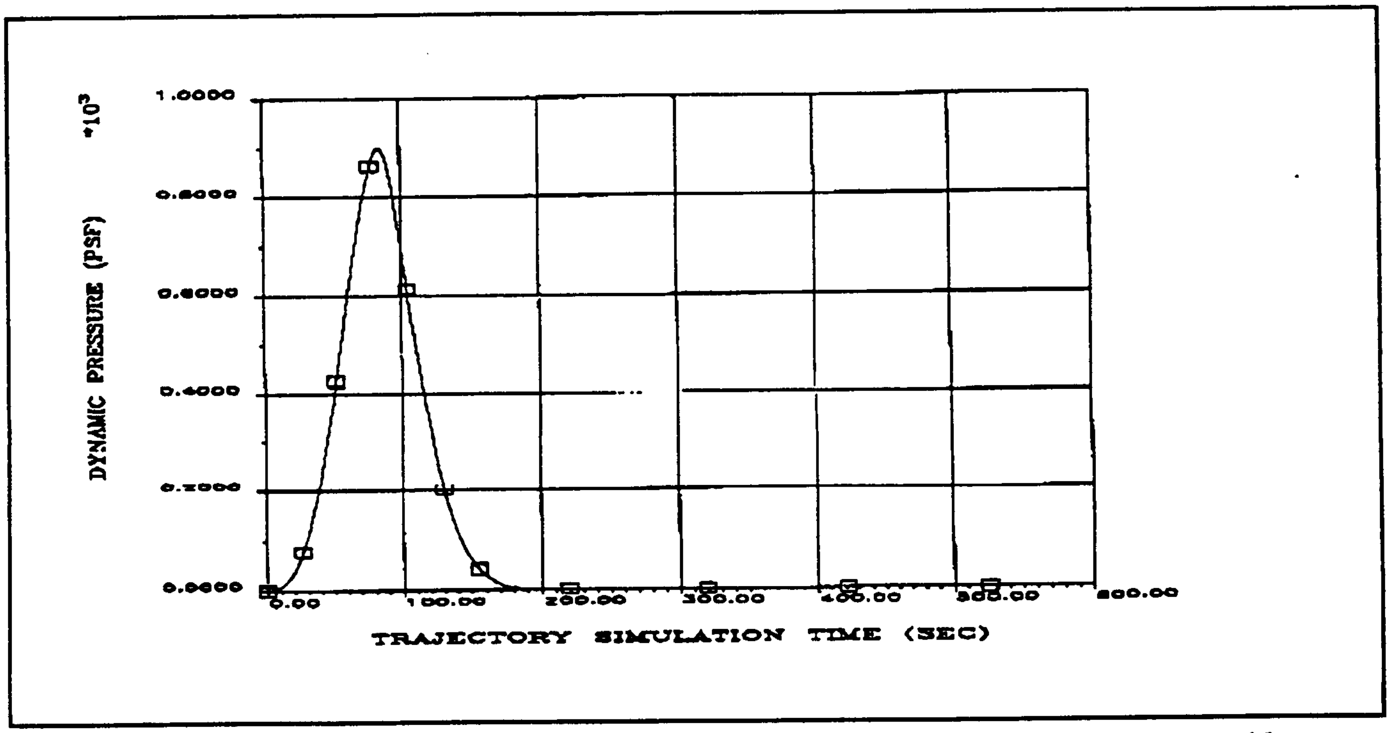


Figure 4.2.2.1-5 Reference Ascent Trajectory Dynamic Pressure Profile

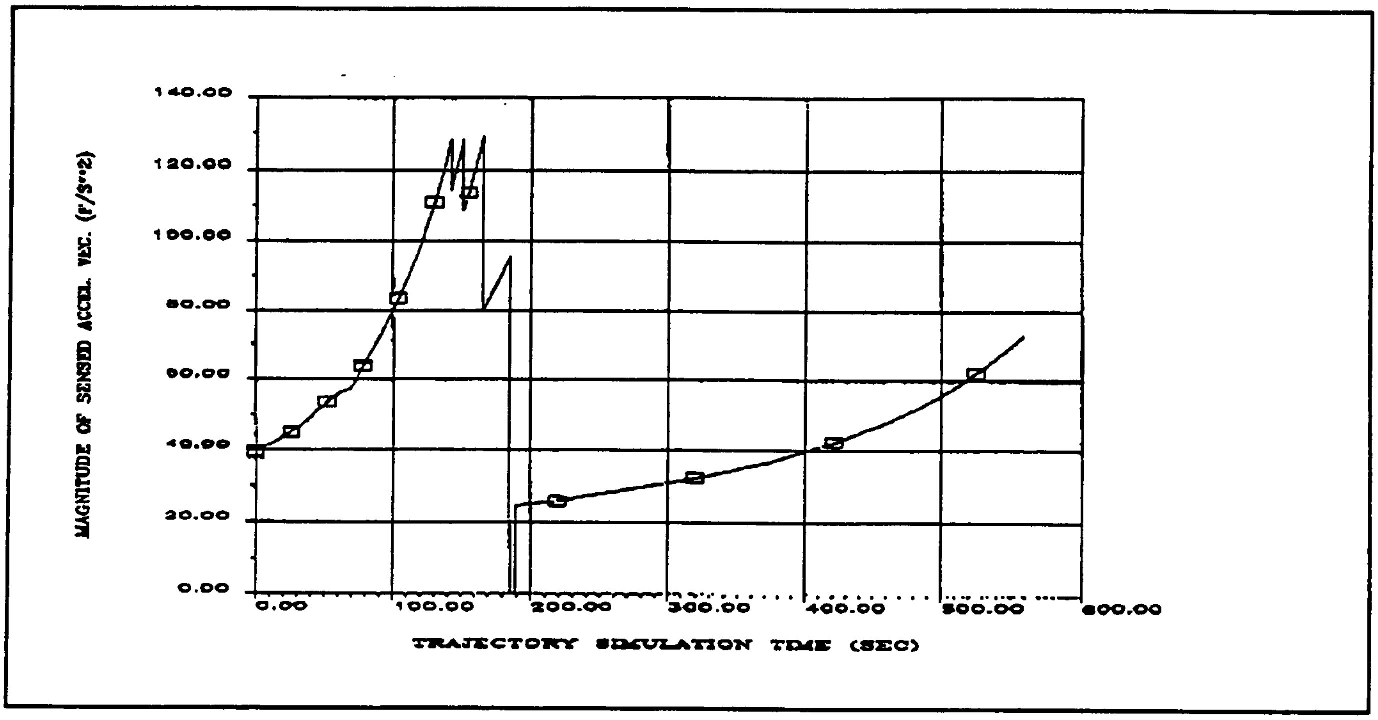


Figure 4.2.2.1-6 Reference Ascent Trajectory Acceleration Profile

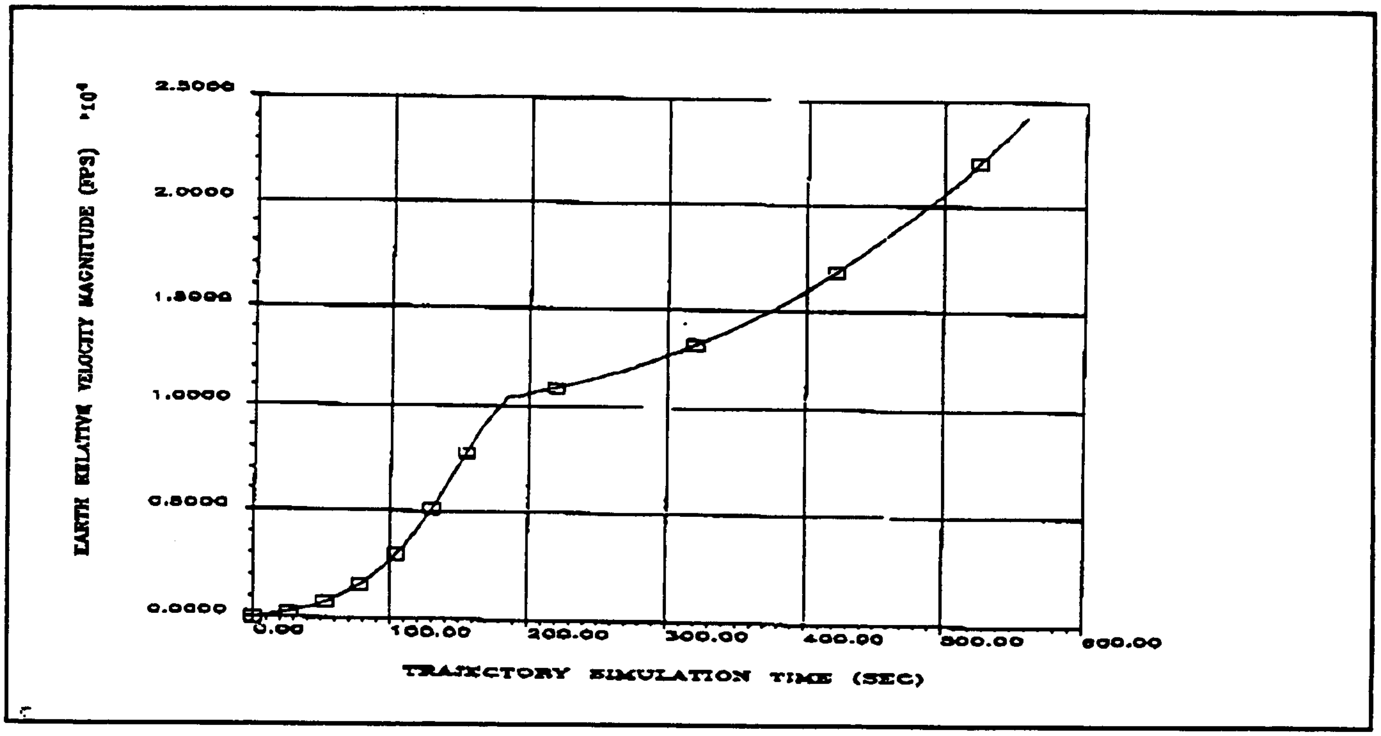


Figure 4.2.2.1-7 Reference Ascent Trajectory Relative Velocity Profile

4.2.2.2 Booster Element

The reference booster configuration and mass properties data are presented in Figure 4.2.2.2-1.

SATURN DERIVED HLLV
BOOSTER CHARACTERISTICS
(ALL MASSES IN LBM)

FORWARD STRUCTURES & NOSE CONE	6,402
ATTACH & SEPARATION	11,330
LOX TANK	17,764
INTERTANK	5,612
RP-1 TANK	11,931
AFT STRUCTURES	2,346
SECONDARY STRUCTURES & FAIRINGS	3,746
TANK INSULATION	730
THRUST STRUCTURE & HOLD-DOWN	9,580
ENGINE (2 F-1As)	38,000
ENGINE MOUNTS	404
BASE HEAT PROTECTION	1,912
LOX SYSTEM	8,091
RP-1 SYSTEM	5,664
TVC (GIMBAL SYSTEM)	5,219
ELECTRICAL, HYDRAULIC, & POWER SYSTEMS	3,219
AVIONICS	2,976
CONTINGENCY	13,193
DRY MASS	148,119
RESIDUALS	21,724
INERT MASS	169,843
TOTAL USABLE PROPELLANT	2,172,377
TOTAL STAGE MASS	2,342,220

STAGE MASS FRACTION	0.927
STAGE IGNITION T/W	1.21
STAGE BURN TIME	168 SECONDS

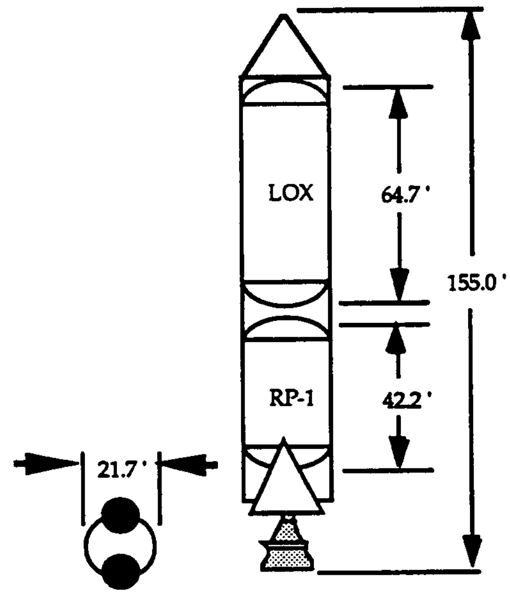


Figure 4.2.2.2-1 Saturn V-Derived Booster Mass Properties Summary

A 6.6 m (260 in.) booster diameter was selected on the basis of earlier studies and sizing considerations regarding the forward booster attach location. Results from vehicle sizing optimization trade studies performed during early phases of this study indicated that the optimum booster burn duration was relatively insensitive to variations in booster diameter, across the diameter range evaluated. For the propellant volume requirements corresponding to these durations, a 6.6 m (260 in.) diameter provided acceptable attach locations and was therefore selected as the baseline. Subsequent HLLV sizing analyses were performed using this diameter to arrive at the reference booster system definition, which has an overall length of 47.3 m (155 ft) for the maximized payload, length-constrained, lunar HLLV. With the constraints and assumptions imposed on the Saturn V-Derived HLLV configuration during this study phase, it was unnecessary to consider trade studies for the number of engines per booster since a minimum of two engines was required to achieve acceptable lift-off thrust-to-weight ratios and a greater quantity would have resulted in unacceptable vehicle acceleration levels for practical burn durations. Additionally, a minimum number of engines was desired in the interest of enhancing vehicle reliability and cost. From a numerical reliability perspective, overall vehicle or element reliability is decreased with any relative increase in the number of engines and the requisite propulsion feed subsystem components, presuming that the numerical reliability is known for each engine and feed subsystem component. It is true, however, that the ability of the vehicle to successfully achieve the desired orbital insertion conditions, after having sustained an engine-out condition, can be enhanced from a performance perspective with a relative increase in the number of engines. Since there was no explicit requirement to provide mission success capability after sustaining a booster engine out, a minimum number of engines was preferred. The booster hardware was assumed to not be reusable or recoverable. Consistent with the NLS-derived reference vehicle, standard 2219 aluminum was selected for all major primary and secondary structural elements.

The tabulated booster subsystem mass properties were estimated on a basis consistent with Saturn V design philosophy and materials. For purposes of this study, acceptable forward attach locations for the strap-on boosters were limited to the forward skirt and nose cone regions of the booster and to the forward skirt and interstage regions of the modified S-IC stage, in order to avoid structural attachment into barrel segments of an oxidizer tank. It was assumed, based on historical design experience, that greater design complexity would be incurred if the booster forward attach location was anywhere on the first stage liquid oxygen (LOX) tank, rather than at the interstage or forward skirt. On the reference vehicle, the forward attach location joined the booster nose cone to the forward skirt assembly of the modified S-IC stage and the aft attach location joined the stage elements via the thrust structure assemblies. The booster thrust loads are transmitted to the core vehicle through the booster's diagonal aft attach struts. Booster lateral loads are transmitted to the core vehicle at the booster's horizontal aft and forward attach struts. The attach strut hardware was placed at the booster thrust structure and forward skirt elements, which is more structurally efficient than being at the pressurized volume of the booster propellant tanks. Booster propulsion is supplied by two F-1A engines that are attached to the aft thrust structure assembly, which transmits thrust loads to the core vehicle elements. The baselined load path reacts the booster thrust loads directly into the thrust structure assembly of the modified S-IC core stage via the thrust struts at the aft booster attach location. The booster thrust structure assembly supports the booster while on the pad or at test facilities and serves as the primary attach structure for the base heat shield, engine fairings, engine actuators, and propellant lines. Mass properties for the aft structure assembly were estimated assuming design similarity to the S-IC stage thrust structure arrangement of ring frames, stiffeners, and thrust posts. The propellant container assembly consists of two separate, cylindrical tank configurations with the fuel (RP-1) tank located aft and the oxidizer (LOX) tank located forward. Both tanks were assumed to be of similar construction to that of the S-IC propellant tanks, and are characterized by ring baffle strengthened, integrally stiffened cylindrical skin segments joined to ellipsoidal ($\sqrt{2}$ ratio of semi-major to semi-minor axes) upper and lower bulkheads. The RP-1 tank was composed of a 8.2 m (26.9 ft) cylindrical segment and two bulkhead segments of 2.3 m (7.7 ft) in length each, providing 387 m^3 (13,670 ft^3) of container volume. The LOX tank cylindrical segment was 15 m (49.4 ft) in length and utilized bulkhead segments of the same geometry as the RP-1 tank, producing a container volume of 623 m^3 (21,985 ft^3). The intertank assembly was assumed to utilize a longitudinally stiffened skin structure, stabilized by internal ring frames similar in design to the S-IC intertank. The assembly provides structural continuity between the cylindrical tank segments and allows for a 0.9 m (3 ft) clearance between tank bulkheads. The forward skirt structure joins the oxidizer container to the nose cone structure and was assumed similar in concept to the stiffened cylindrical skin structure of the S-IC skirt. A basic ring frame, stiffened skin panel structure was assumed for the load bearing booster nose cone which connects to a light-weight non load-bearing nose fairing structure coated with ablative insulation. The nose cone geometry is a right cone with a 30 degree half angle. Attachment structure mass estimates for joining the booster elements to the modified S-IC reflect the aft attach thrust strut structure which reacted axial, lateral, and torque loads and the forward attach support strut structure, which reacted lateral loads.

Booster performance for the lunar HLLV provides a total of 32 million N (7.2 Mlbf) of sea-level thrust augmentation at lift-off. Each booster has a total fueled mass of 1,061 t (2.34 Mlbm) and consumes propellant at a rate of 6.1 t/sec (13,350 lbm/sec) during 100 percent RPL operation. Limitation of ascent dynamic pressure to a maximum of 43.1 K N/m^2 (900 psf) is achieved without throttling of the booster engines. During nominal ascent, the booster F-1A engines are permanently step-throttled to 75 percent RPL at approximately 146 seconds into flight for vehicle acceleration control. The total booster burn duration is 168 seconds, corresponding to shut-down of the two F-1A engines on each booster.

4.2.2.3 Core Stage I Element

The modified S-IC stage characteristics and mass properties for the reference HLLV are shown in Figure 4.2.2.3-1. The stage characteristics represent the results of integrated vehicle sizing analyses performed to define a maximized payload capability configuration for the 125 m (410 ft) length-constrained lunar HLLV. These analyses consider modified S-IC stage options with either five or six F-1A engines in conjunction with various engine combination options on the other core stages. Variations to the S-IC stage length were assessed simultaneously with length variations to the other core stages, subject to the fixed overall vehicle height, on the basis of the corresponding staging velocity performance impacts to post-TLI payload capability. Constraints imposed during the sizing process to screen out undesirable configurations resulted in practical limitations to acceptable S-IC length modifications. These constraints include boundaries on acceptable ignition and burnout thrust-to-weight ratios and the limitations imposed on the forward booster attach location (see booster description). For the down-selected, maximum payload HLLV configuration, the S-IC stage modifications were characterized by a 6.8 m (22.4 ft) stretch resulting in a 48.8 m (160 ft) overall stage length (excluding interstage). Total stage usable

RP-1/LOX propellant capacity was increased to approximately 2,730 t (6 Mlbf) and total stage mass was increased to 2,943 t (6.5 Mlbm). The Stage I hardware was assumed to not be reusable or recoverable, by design. Standard 2219 aluminum was selected for all major primary and secondary structural elements.

SATURN DERIVED HLLV

S-IC STAGE (MODIFIED) CHARACTERISTICS

(ALL MASSES LBM)

INTERSTAGE	9,774
FORWARD STRUCTURES	5,443
LOX TANK	49,212
INTERTANK	12,286
RP-1 TANK	33,053
AFT & INSTALL STRUCTURES	5,443
SECONDARY STRUCTURES & FAIRINGS	7,390
TANK INSULATION	1,367
THRUST STRUCTURE	91,696
TPS, BASE HEAT PROTECTION	4,109
ENGINE (5 F-1As)	95,000
ENGINE MOUNTS	1,011
RP-1 SYSTEM	14,159
LOX SYSTEM	20,227
TVC (GIMBAL SYSTEM)	13,046
ELECTRICAL, HYDRAULIC, & POWER SYSTEMS	5,209
SEPARATION SYSTEM	2,185
AVIONICS	3,396
CONTINGENCY	37,400
DRY MASS	411,406
RESIDUALS	60,181
INERT MASS	470,587
TOTAL USABLE PROPELLANT	6,018,144
TOTAL STAGE MASS	6,489,731

STAGE MASS FRACTION (INCLUDES INTERSTAGE)	0.927
STAGE IGNITION T/W	1.21
STAGE BURN TIME	189 SECONDS

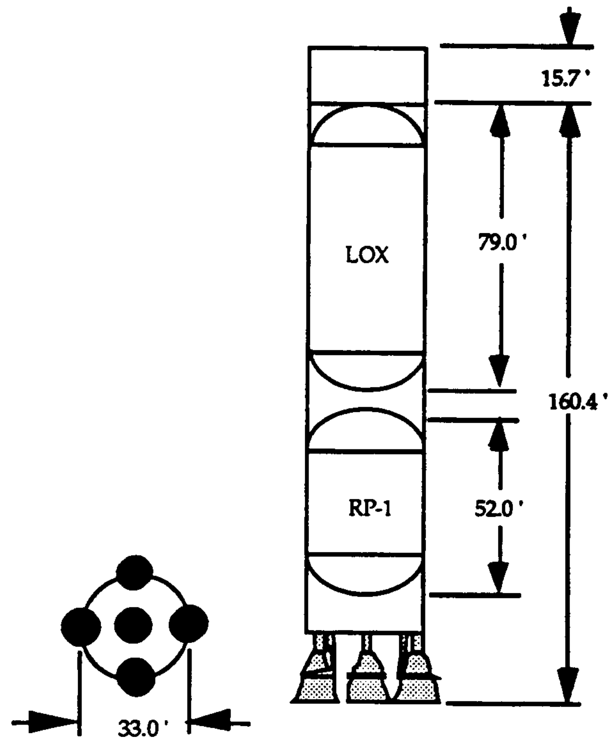


Figure 4.2.2.3-1 Saturn V-Derived Core Stage I Mass Properties Summary

The modified stage mass properties were estimated assuming no significant changes to design philosophy or materials used on the S-IC. Stage propulsion was provided by five F-1A engines supported by a modified thrust structure assembly. The skin stringer frame configuration of the S-IC thrust structure was strengthened to account for the increased thrust level of the F-1A engines as well as to react the thrust loads of the attached booster elements. The assembly provides support for the base heat shield, engine fairings, engine actuators, and propellant lines, and contains the hold-down structure for vehicle restraint during thrust build-up/check-out. Modification to the base heat shield was required for the increased radiant heating environment induced by the F-1A engine thrust levels and the proximity of the booster engines. The S-IC fin assemblies were removed in consideration of the attach locations for the booster elements. The integral propellant container configuration is similar to that of the S-IC and was modified to accommodate increased fuel and oxidizer capacities. The aft RP-1 tank was sized with a 8.7 m (28.7 ft) cylindrical segment connecting to two ellipsoidal ($\sqrt{2}$ ratio of semi-major to semi-minor axes) bulkhead segments each 3.6 m (11.7 ft) in length, for an overall stretch of 2.7 m (9 ft) relative to the S-IC fuel tank. Total fuel container volume was increased to 1,073 m^3 (37,860 ft^3). The LOX tank, located forward, was lengthened approximately 4.5 m (15 ft) relative to the S-IC oxidizer tank. It incorporated a 17 m (55.7 ft) cylindrical segment and two bulkhead segments similar in geometry to the fuel tank bulkheads, resulting in a total oxidizer container volume of 1,726 m^3 (60,900 ft^3). The intertank assembly structural joins the cylindrical tank segments and was assumed to be analogous in design to the S-IC corrugated skin, frame-stiffened intertank structure. A 0.9 m (3 ft) clearance was provided between tank bulkheads. The forward skirt structural assembly joins the cylindrical segment of the oxidizer tank to the interstage structure using the same structural configuration as the S-IC skirt. The forward skirt structure serves as the attach location for the booster forward attach struts and reacts the lateral booster loads.

The modified S-IC core stage delivers 40 million N (9 Mlbf) total sea-level thrust. At 100 percent RPL engine operation, stage propellant is consumed at a rate of 15.1t/sec (33,370 lbm/sec). Maximum ascent dynamic pressure is limited to 43.1 K N/m^2 (900 psf) without requiring throttling of the core stage engines. During nominal ascent, the five F-1A engines are permanently step throttled to a 75 percent RPL at approximately 155 seconds into flight in order to provide ascent acceleration control. The total stage burn duration was 189 seconds, and corresponds to shut down of the five F-1A engines.

4.2.2.4 Core Stage II Element

Characteristics and mass properties for the modified S-II stage of the reference HLLV configuration are provided in Figure 4.2.2.4-1. The stage definition was derived on the basis of the integrated HLLV sizing analyses performed for a maximized payload capability lunar vehicle, constrained to a 125 m (410 ft) overall height. The sizing analyses considered modified S-II stage options with either five or six J-2S engines in conjunction with various engine combination options on the other core stages. Length modifications to the S-II stage were evaluated simultaneously with length variations to the other core stages, subject to the fixed maximum HLLV height, on the basis of the corresponding staging velocity performance impacts to post-TLI payload capability. A constraint imposed during the sizing process to discriminate those configurations with unacceptably low ignition thrust-to-weight ratios for the modified S-II stage resulted in some limitation to the domain of length modifications. Evaluation of the numerous vehicles defined during the sizing analysis process against the criterion of maximizing payload led to selection of the reference HLLV incorporating a modified S-II stage characterized by a 6.6 m (21.5 ft) stretch, for a 31 m (102 ft) total stage length (excluding interstage), and a propulsion configuration of six J-2S engines. All vehicles sized with modified S-II stages utilizing five J-2S engines were found to have either less performance capability or characteristics which violated imposed HLLV constraints (e.g., unacceptable first stage acceleration levels) for the 125 m (410 ft) configuration. The modifications to the S-II stage increased the stage usable LH_2/LOX propellant capacity to approximately 634t (1.4 Mlbm) and increased the total stage mass to 700 t (1.5 Mlbm). The Stage II hardware was assumed to not be reusable or recoverable, by design. Standard 2219 aluminum was selected for all major primary and secondary structural elements.

SATURN DERIVED HLLV

S-II STAGE (MODIFIED) CHARACTERISTICS

(ALL MASSES IN LBM)

INTERSTAGE	9,774
FORWARD STRUCTURES	5,443
LH2 TANK (COMMON BULKHEAD)	30,010
LOX TANK (COMMON BULKHEAD)	13,999
AFT STRUCTURES	5,443
SECONDARY STRUCTURES	905
TANK INSULATION	3,461
TPS, BASE HEAT PROTECTION	584
THRUST STRUCTURE	6,360
ENGINES (6 J-2Ss)	22,800
ENGINE MOUNTS	159
LH2 SYSTEM	3,975
LOX SYSTEM	3,180
TVC (GIMBAL SYSTEM)	2,051
ELECTRICAL, HYDRAULIC, & POWER SYSTEMS	2,296
SEPARATION SYSTEM	623
AVIONICS	2,520
CONTINGENCY	11,358
DRY MASS	124,941
RESIDUALS	13,830
IN-FLIGHT LOSSES	5,947
INERT MASS	144,718
TOTAL USABLE PROPELLANT	1,398,705
TOTAL STAGE MASS	1,543,423

STAGE MASS FRACTION (INCLUDES INTERSTAGE)	0.906
STAGE IGNITION T/W	0.75
STAGE BURN TIME	383 SECONDS

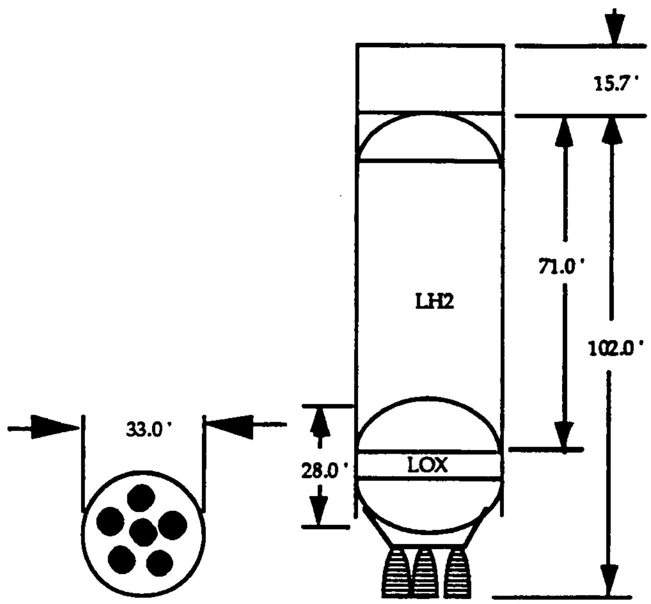


Figure 4.2.2.4-1 Saturn V-Derived Core Stage II Mass Properties Summary

Stage mass properties estimates were developed in the same manner as with the other vehicle elements, assuming basic S-II stage design and materials properties. The stage propulsion was supplied by the six J-2S engines attached to a modified thrust structure assembly. A configuration similar to the S-II design was assumed, consisting of a conical thrust structure arrangement, center support assembly, and engine mount frame, but is modified to accommodate six engines and the increased loads. The thrust structure assembly reacts the engine thrust loads into the aft skirt of the stage and provides support for the base heat shield and propellant feed lines. Boost loads from the lower stages are transmitted via the aft interstage to a structurally modified aft skirt assembly, which attaches to the cylindrical segment of the stage oxidizer tank. A common bulkhead propellant container arrangement similar to the S-II configuration was used for the modified S-II stage for overall length efficiency. The integral oxidizer tank was sized with ellipsoidal ($\sqrt{2}$ ratio of semi-major to semi-minor axes) bulkhead segments, each 3.6 m (11.7 ft) in length, connected by a 1.4 m (4.7 ft) cylindrical section to accommodate increased LOX capacity. The forward LOX tank bulkhead serves as the aft bulkhead of the fuel tank. The 8.5 m (28 ft) oxidizer tank provides a total volume of 490 m^3 (17,325 ft^3). The integral fuel container incorporates an ellipsoidal forward bulkhead, similar in geometry to the oxidizer bulkhead, and a 18.1 m (59.3 ft) cylindrical segment which extends to the base of the common bulkhead. The RP-1 container volume was increased to 1,437 m^3 (50,710 $ft.^3$). All ascent reserve propellants, totaling approximately 6.7 t (14,800 lbm), are carried within this stage. A modified forward skirt structural assembly is joined to the base of the fuel tank forward bulkhead using a structural configuration similar to the S-II.

The modified S-II stage delivers 7.1 million N (1.59 Mlbf) total vacuum thrust and has a design burn duration of 383 seconds. Stage propulsion is operated continuously at 100 percent RPL and consumes propellant at a rate of 1.65 t/sec (3,645 lbm/sec).

4.2.2.5 TLI Stage Element

The TLI stage characteristics and mass properties of the reference HLLV configuration are provided in Figure 4.2.2.5-1. The stage configuration selection is based on the results of integrated HLLV sizing analyses which were performed for a 125 m (410 ft), maximum payload capability vehicle. A 10 m (33 ft) diameter is baselined for the TLI stage to provide tooling commonality with the modified S-IC and S-II stages. TLI stage options using either one or two J-2S engines are considered in combination with various engine configurations on the other core stages to define the vehicle configuration with maximum payload performance. The spectrum of vehicles defined by the sizing analyses consider TLI stage options both with and without sub-orbital operation phases. Variations in TLI stage length are evaluated simultaneously with length modifications to the S-IC and S-II stages, for a fixed vehicle height of 125 m (410 ft), by assessing the corresponding staging velocity performance impacts to post-TLI payload capability. A constraint is applied during the sizing process to screen out vehicles incorporating TLI stages which would operate suborbitally with unacceptably low ignition thrust-to-weight ratios. The constraint results in practical limitation to the extent of TLI stage length variations under consideration. Evaluation of the matrix of vehicles, defined during the sizing process, using the study groundrules and the criterion of maximizing payload, leads to selection of the reference HLLV incorporating a TLI stage not designed for sub-orbital operation. Performance gains associated with suborbitally operated TLI stage options are small for the single J-2S configurations. For these configurations, the growth in TLI stage mass necessary to deliver the required lunar mission payload, for even short sub-orbital operation segments, degrades the overall ascent performance for the majority of vehicles evaluated, as a result of low ignition thrust-to-weight ratios. All vehicles which incorporate two-engine TLI stage options are capable of longer duration sub-orbital phases, however, overall payload performance for these vehicles is not found to exceed the reference HLLV capability, when the HLLV sizing was performed for fixed 125 m (410 ft) height configurations. Consequently, the attributes of only a single engine and only a single engine-start for the TLI stage are reflected in the reference configuration. The new TLI stage element is characterized by a usable propellant capacity of 135 t (298,500 lbm), a total stage mass of approximately 157 t (345 Klbm), and a stage length of 16.6 m (55 ft). The TLI stage hardware is assumed to not be reusable or recoverable, by design. Standard 2219 aluminum is selected for all major primary and secondary structural elements.

SATURN DERIVED HLLV

TLI STAGE CHARACTERISTICS (REF. MSFC)

(ALL MASSES IN LBM)

FORWARD TRANSITION STRUCTURE	(included in payload fairing mass))
FORWARD SKIRT	3,239
LH2 TANK	5,620
INTERTANK	8,930
LOX TANK	4,310
AFT STRUCTURE	2,715
TPS, BASE HEAT PROTECTION	2,628
ENGINE (1 J-2S)	3,700
THRUST STRUCTURE	1,472
PROPULSION SUBSYSTEMS	4,170
TVC (GIMBAL SYSTEM)	400
SEPARATION SYSTEM	118
RCS	897
AVIONICS	2,200
CONTINGENCY	4,000
DRY MASS	44,399
RESIDUALS + RCS PROPELLANT	2,638
INERT MASS	47,037
PROPELLANT	295,507
RESERVE PROPELLANT	2,985
TOTAL USABLE PROPELLANT	298,492
TOTAL STAGE MASS	345,529
STAGE MASS FRACTION	0.864
STAGE IGNITION T/W	0.48
STAGE BURN TIME	486 SECONDS

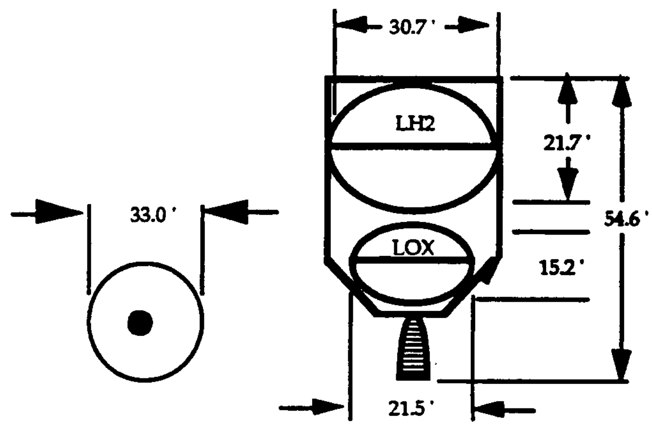


Figure 4.2.2.5-1 Saturn V-Derived TLI Stage Mass Properties Summary

The mass property estimates for the reference HLLV TLI stage are developed on the basis of similar assumptions as applied to the other stage elements. The single J-2S engine is attached to a conical thrust structure assembly which transmits the thrust loads into the intertank stage structure. A basic skin-stringer-frame configuration is assumed for the interstage structure. The propellant tank assembly consists of two separate, non-integral, ellipsoidal tank configurations with the oxidizer (LOX) tank located aft and the fuel (RP-1) tank located forward. The 6.6 m (21.5 ft) diameter oxidizer tank is sized with ellipsoidal ($\sqrt{2}$ ratio of semi-major to semi-minor axes) bulkheads each 2.3 m (7.6 ft) in length. The total oxidizer tank volume is approximately 104 m^3 (3,670 ft^3). The 9.4 m (30.7 ft) diameter ellipsoidal fuel tank utilizes bulkheads which were each 3.3 m (10.8 ft) in length and provides a total container volume of approximately 305 m^3 (10,760 ft^3). Mass property estimates for the non-integral propellant tanks include the mass of attach structure required to join the tank assembly to the intertank structure. A forward skirt structure, similar in concept to the S-II stage skirt, structurally joined the intertank assembly to the payload shroud transition structure and transmitted the loads from the lower stages.

The reference HLLV TLI stage delivers 120 t (265 Klbf) total vacuum thrust and has a design burn duration of 486 seconds. The single J-2S engine is operated continuously at 100 percent RPL throughout the burn duration and consumes propellant at a rate of 276 Kg/sec (608 lbm/sec).

4.2.3 Aerodynamics

HLLV aerodynamic forces during ascent are simulated using reference aerodynamic coefficient data developed by Boeing in 1966 for similar Saturn V-Derived vehicle configurations, equipped with up to four strap-on booster elements. During early phases of the study, comparative trajectory simulations were completed with a lunar HLLV, using both the referenced HLLV aerodynamic data and aerodynamic force data developed for simulations of the National Launch System (NLS) HLLV configuration. Results of the simulations demonstrate greater performance capability in the case which used the NLS aerodynamic data. Since the Saturn V-Derived vehicle aerodynamic data are considered more conservative, they were baselined for all subsequent performance analyses. Aerodynamic force sensitivities to the specific payload shroud configurations under consideration, have not been accounted for in the present analyses. Power-on base effects aerodynamics were assumed to not be major configuration design drivers, and thus were not modeled or assessed.

4.2.4 Stability and Control

It is assumed that both the boosters and core vehicle would require some form of TVC, since the boosters separate from the core vehicle prior to core first stage burn-out. The mass properties for the vehicle reflect estimates of the TVC hardware requirements. Stability and control analyses have not yet been performed to ascertain the degree of control authority between the boosters and core vehicle during first stage, nor the precise timing of any control authority hand-over from the boosters to the core vehicle.

4.2.5 Manufacturing Facilities and Tooling

It is assumed that the stage propellant tanks, interstages, and intertanks will be manufactured at MAF. Maximum utilization of existing MAF infrastructure is also assumed. In consideration of demonstrated manufacturing capability and the potential for reduced manufacturing equipment costs, the S-IC / S-II stage diameter of 10 m (33 feet) was baselined for the core stage elements of the Saturn V-derived configuration. Feasibility and sensitivity studies for increased core stage diameters have not been addressed.

4.2.6 Schedules

Figures 4.2.6-1, 4.2.6-2, 4.2.6-3, and 4.2.6-4 display preliminary schedules for the development and acquisition phases of a Saturn-derived S-IC stage, S-II stage, TLI stage, and boosters, respectively.

The major features of the schedules are a two-year in-house preliminary definition study, immediately followed by a five-year Phase C/D, beginning in early Fiscal Year (FY) 1995. Initiation of the preliminary definition studies in the last quarter of FY 1992 would be necessary to accommodate a launch in 1999. These schedules also show estimates for long lead item procurement and fabrication requirements for the major Saturn-derived HLLV subsystems.

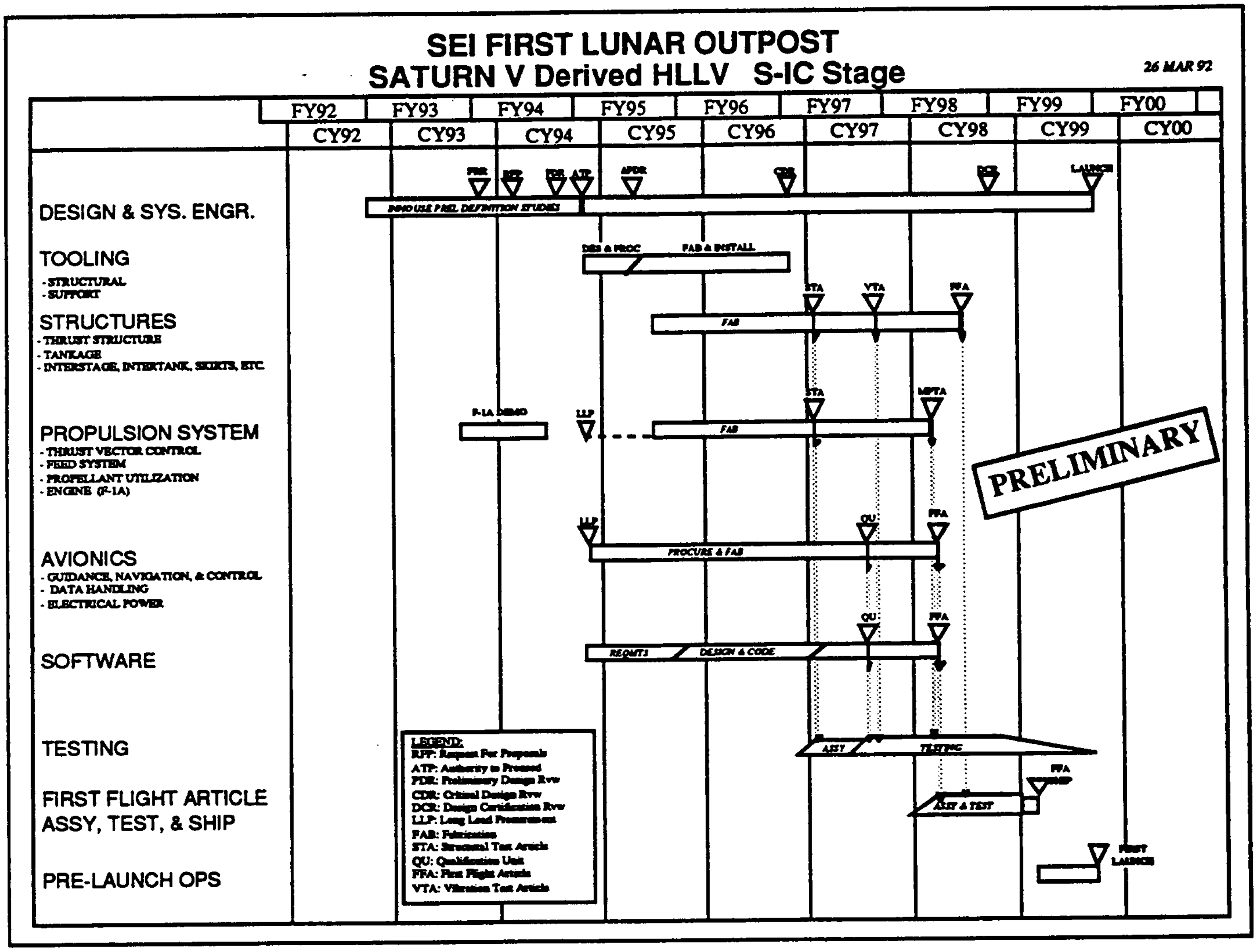


Figure 4.2.6-1 Saturn V-Derived Stage I Development & Acquisition Schedule

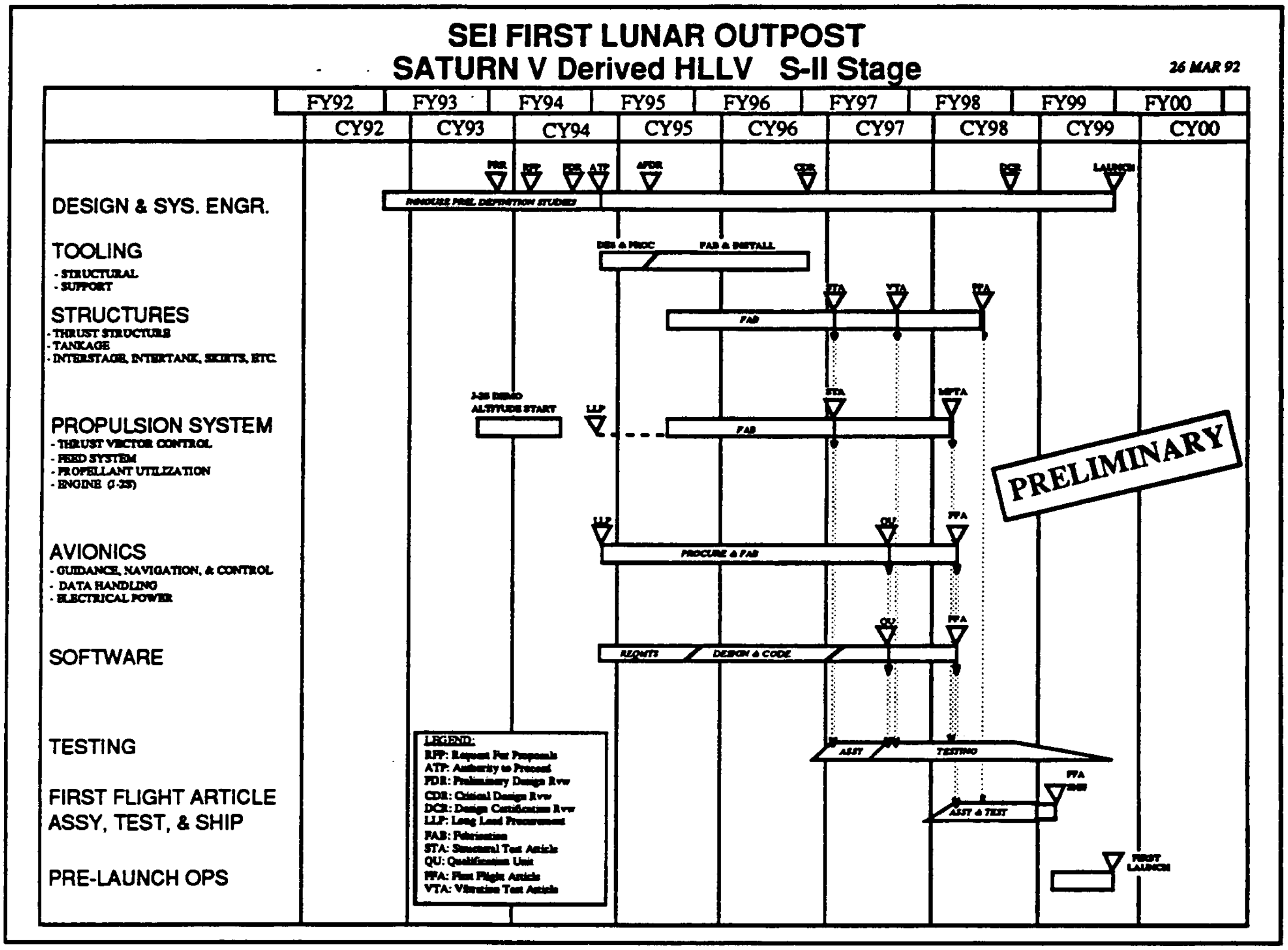


Figure 4.2.6-2 Saturn V-Derived Stage II Development & Acquisition Schedule

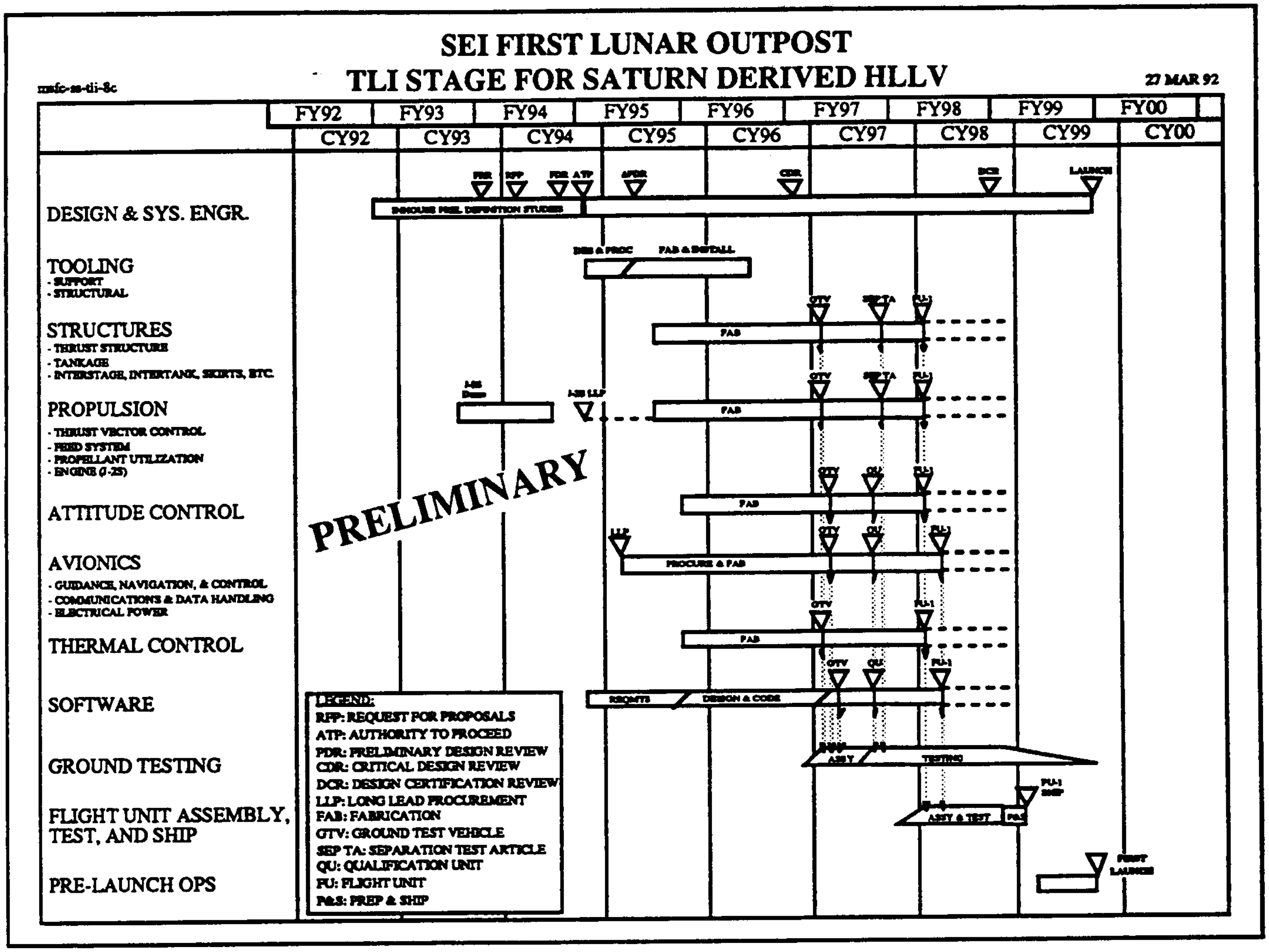


Figure 4.2.6-3 Saturn V-Derived TLI Stage Development & Acquisition Schedule

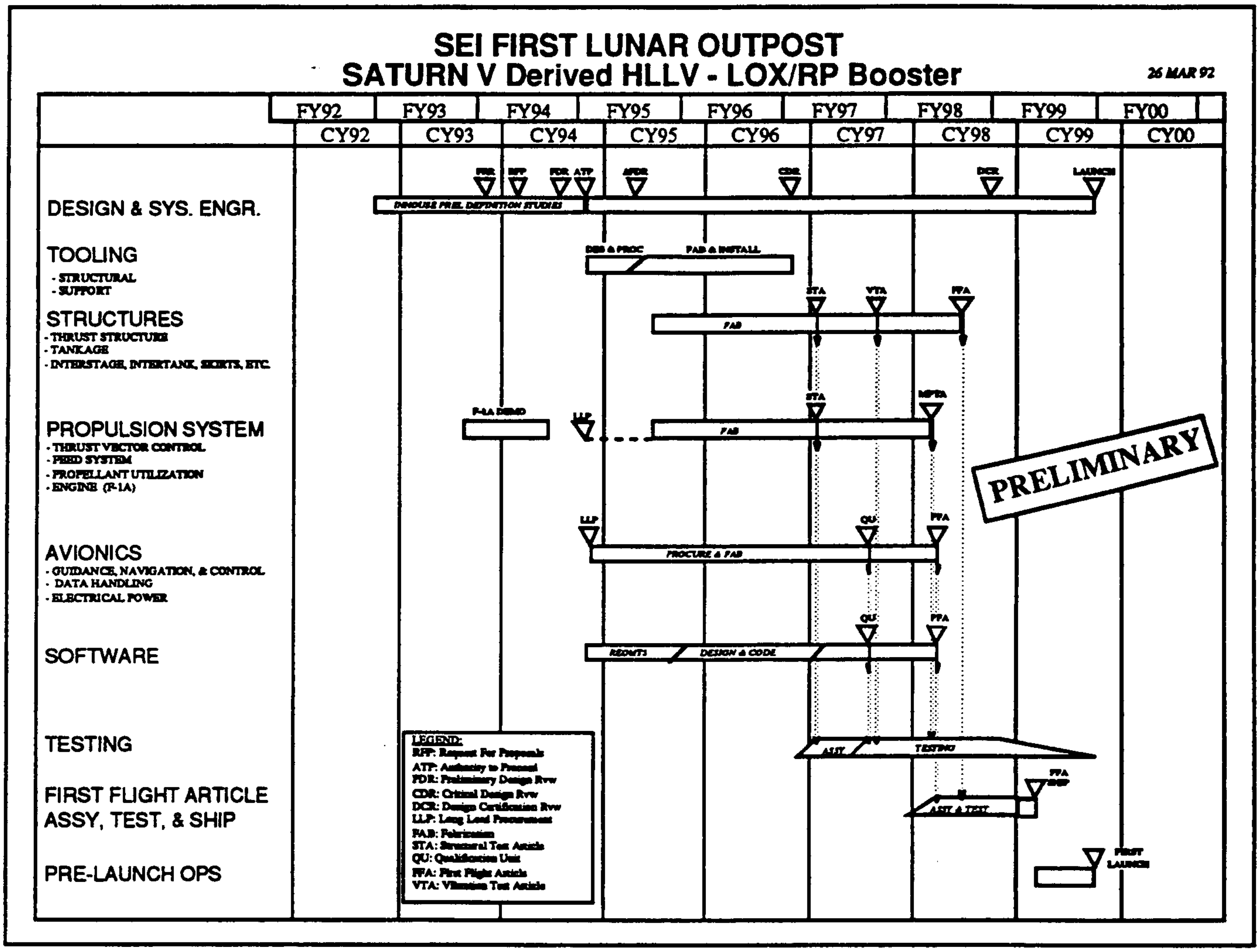


Figure 4.2.6-4 Saturn V-Derived Booster Development & Acquisition Schedule

5. Payload Shroud

5.1 Piloted and Cargo Versions

5.1.1 Shroud Specifications

Figure 5.1.1-1 illustrates the lunar payload shroud configurations. Three configurations are shown: cargo shroud with a biconic nosecone (15 deg/27.6 deg) and a piloted and cargo shroud using a common diameter and nosecone shape. The biconic has better aerodynamic characteristics, but because it cannot accommodate the piloted abort requirements using a common shroud, the latter two are selected as the reference.

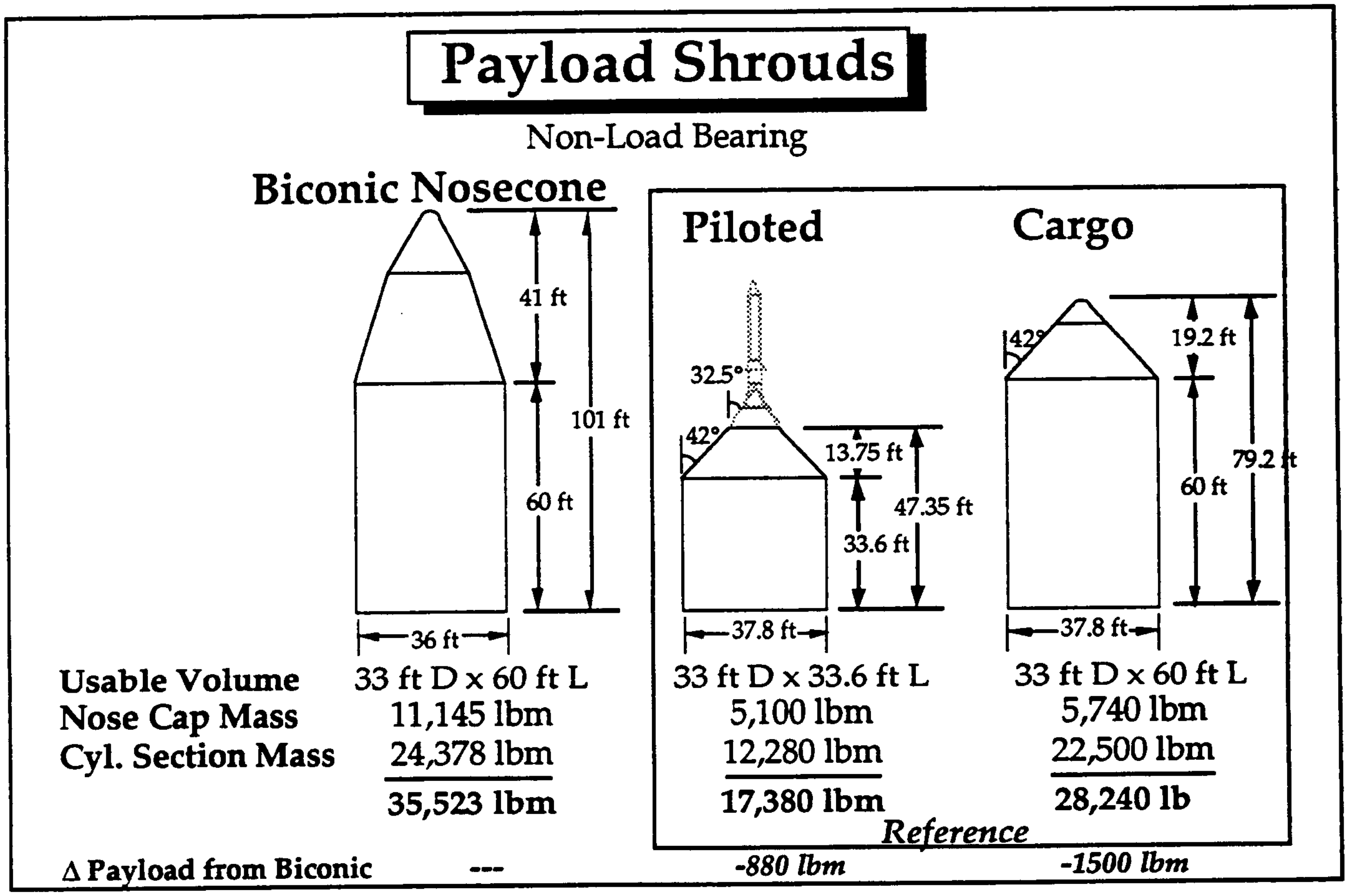


Figure 5.1.1-1 Payload Shroud Configurations

The payload configurations were unknown until the close of the study, therefore all aerodynamic and structural analysis assumed the biconic shroud shape. The shrouds are not designed to support the payload in the axial direction, but have some lateral support capability. The payloads are essentially supported at their base by the forward adapter of the core. The shroud usable diameter is 10 m (33 ft), which is a study requirement. Based on preliminary structural analyses, the shroud's outer diameter is approximately 38 feet. This is driven by the depth of the ring frame where payload lateral loads are removed. The shrouds are constructed using Aluminum 2219 isogrid skins and ringframes. The nosecone 42 degree half angle is driven by the piloted lunar lander, ascent stage, and cargo size. The cylindrical section length is driven by the lander, ascent stage (if required), and habitat (if required) length. The piloted crew module and launch escape system protrude through the top portion of the nosecone, for launch abort capability.

Table 5.1.1-1 compares the NLS-derived reference shroud specifications with those of the baseline NLS payload shroud.

Table 5.1.1-1 Mass Properties Comparison Between NLS and NLS-Derived Shrouds

SHROUD 9 RESULTS COMPARISON

Shroud Components	NLS	NLS-Derived
Nose Cone	1,361 lbm	10,771 lbm
Shroud Cylinder	10,364 lbm	22,486 lbm
Separation	766 lbm	1,145 lbm
Shroud Total	12,491 lbm	34,402 lbm
Shroud Adapter	9,355 lbm	9,362 lbm
Skirt	1,093 lbm	1,280 lbm
Adapter Total	10,448 lbm	10,642 lbm

5.1.2 Vehicle Aerodynamics and Performance

See the aerodynamics and performance trade study results in Section 7.3.1.

6. Test Program and Facilities

6.1 Propulsion System Test Program and Facilities

6.1.1 Baseline

A groundrule was established that individual engine testing would be performed both during the developmental phase and the operational phase, the latter equating to a flight readiness qualification firing. The complete engine/main propulsion feed subsystem integrated test will be performed only during the developmental phase, at which time the system design would be qualified. Functional component testing and flight readiness certification of the propellant feed subsystem will be performed at the manufacturing facility during the operational phase.

It is assumed that the Stennis Space Center (SSC) will be used as the primary facility for full-scale propulsion article testing. There are four engine/stage class test stands at SSC and they are identified as A-1 and 2 and B-1 and 2. The B stands are designed for higher thrust levels than the A stands. These stands were built during the Apollo program for ground testing of S-IC and S-II development and flight stages. All of the S-II stages and all, but the initial S-IC flight stages, were tested at SSC and shipped to KSC. Subsequent to the Apollo program, all of the stands, except B-2, were modified for development, qualification and flight testing of the SSMEs and this testing continues today. Test stand B-2 was modified and used to test the Space Shuttle Main Propulsion Test Article (MPTA). This stand is currently inactive and the facility remains in the MPTA configuration.

In support of NLS program planning, SSC has committed to convert B-1 to a dual position STME test stand and B-2 to an engine cluster/main propulsion feed system test stand. The Shuttle program has agreed to scale back to two SSME test stands (A-1 and A-2), beginning in FY95. For the purposes of this study, SSC has assumed that it must meet the requirements of all three programs (NLS, SSME and FLO). Based on the two baseline vehicle options and the groundrule that SSC would be the location of engine and propulsion system testing, SSC's recommendations are discussed in the following two sections.

6.1.2 NLS-Derived Vehicle

Table 6.1.2-1 summarizes the SSC test stand resources that would be available if the SEI program was supported concurrently with the NLS program. To meet SSME program requirements, SSME testing should continue on test stand A-2, which has a diffuser to simulate altitude conditions. Historically, SSME test schedules have been constrained by the lack of engine hardware, not be test facility capability. Therefore, it is feasible to limit SSME testing to one stand. In addition, SSME altitude start and restart testing can be performed through limited modification of the A-2 stand and both programs can be supported.

Table 6.1.2-1 Stennis Space Center Test Stand Resource Summary

SSC Resources And Test Plans For SEI - NLS Option

TEST STANDS / RESOURCE USE	A-1	A-2	B-1	B-2	NEW	NEW	NEW	NEW
ORIGINAL STAND PURPOSE	SATURN S-II	SATURN S-II	SATURN S-IC	SATURN S-IC	N/A	N/A	N/A	N/A
SSC IN 1991	SSME	SSME	SSME	STS MPTA INACT.	N/A	N/A	N/A	N/A
SSC WITH NLS (PLANNED)	SSME	SSME	STME	NLS VEH	N/A	N/A	N/A	N/A
SEI NLS OPTION (PROPOSED)	STME*/**	SSME	BOOSTER**	NLS VEH	F-1A	TLI STAGE	N/A	N/A

* Requires SSME Program Office approval
** Requires NLS/STME Program Office approval

Multiple engine/propellant feed system testing will be required for the core and booster. Again, because of the thrust levels involved and because construction of single engine stands is cheaper than construction of stage stands, the B stands are best suited for this testing.

It is proposed that the "displaced" STME testing be moved to test stand A-1, because of the existing LOX/LH_2 facilities on this stand. A new stand will be required for LOX/RP-1 F-1A testing, again based on the commitment of the other stands to other test elements and the cost advantage noted above.

TLI stage testing will be best suited to a new stand, since vacuum capabilities do not currently exist at SSC.

6.1.3 Saturn V-Derived Vehicle

To meet SSME program requirements, it is recommended that SSME testing continue on test stand A-2, which has a diffuser to simulate altitude conditions. Historically, SSME test schedules have been constrained by the lack of engine hardware, not by test facility capability. Given this fact and anticipated test rates, it is feasible to eventually limit SSME testing to one stand. In addition, if the SSME is considered for the Saturn-derived vehicle and altitude start and restart testing are pursued, the test program can be accomplished through limited modification of the A-2 stand and both programs can be supported.

The structural support hardware that was previously used for Saturn S-II integrated stage propulsion testing is still intact and available for use on test stand A-1 at SSC. The stand is adequate for the increased second stage thrust levels and is recommended for this stage test.

The B-1 test stand, previously constructed for Saturn V S-IC testing, can be used for booster and core vehicle testing, where large thrust levels up to 53.4 MN (12 Mlbf) are involved. The B-1 RP-1 systems can also be reactivated. STME testing will move to a new stand, because construction of a new single engine stand is significantly less expensive than construction of a new vehicle stand.

As noted above, the B-2 stand is targeted by the NLS program for core vehicle testing. If the NLS core vehicle does come on-line during the time that a FLO HLLV requires testing, it will be best to allocate the B-2 resources to the NLS vehicle because of the thrust level, and construct new engine stands as required. In this case a new F-1 test stand and a new J-2 stand will have to be built for single engine testing. TLI stage testing will be best suited to a new stand, since vacuum capabilities do not currently exist at SSC.

6.1.4 Unresolved Issues

Major items resulting from the analysis of engine and propulsion system testing included the following:

- The quantity of different propulsion systems and engines impacts the cost of testing and the cost of facilities

- Present day environmental regulations dictate that work to address environmental issues be started immediately, if the October 1999 launch date is to be met

- Without any more detailed information about the modified SSME, F-1A, or J-2S engines and the proposed vehicle feed systems, it is not possible to take a hard look at cost cutting and time saving possibilities. Proposed test schedules show less test time than has been historically required. Use of modified existing or robust engine designs should enhance the ability to meet reduced schedules, but it is not possible to assess this issue without more information and discussion.

- Initial reviews would indicate that testing engines in a horizontal, rather than a vertical, position could result in a cost savings. This issue needs to be pursued in further detail.

- The required schedules are significant factors relative to the application of test facility resources (i.e., the ability to sequence testing is reduced)

- Although the rate at which engines are tested is probably a minor factor in determining annual procurement and production rates, use of test stands below the test rate capacity does increase test costs. Actions could be taken to determine optimum production and test rates and plans, once the program reaches maturity, as one of the new ways of doing business.

6.2 Complete Test Program and Facilities

Analysis of other aspects of the test program, such as structural tests, material qualification testing, etc., has not yet been addressed.

7. Key Trade Studies

Several trade studies were performed to help identify and assess the reference vehicle configurations, as well as to help identify and assess alternative configurations.

7.1 NLS Derived

7.1.1 Reference Vehicle Concepts

Figure 7.1.1-1 shows selected NLS derived options that lead to the selection of the reference configuration, shown on the far right. Gross lift-off weight (GLOW), post-TLI payload, and the mass of the system in low-earth orbit (i.e., prior to TLI burn) are indicated. All options ignite the booster and core engines at lift-off and hold the core propellant load to be the same as the current NLS reference core. The options are essentially in chronological order from left to right. The payload requirement until early March was 76 t post-TLI, therefore, the three options shown on the left were sized for this requirement. All payload capabilities shown are the result of optimizing the booster and upper stage propellant loads, with the exception of the reference configuration's TLI stage. All options except the three-stage vehicle suborbitally ignite the TLI stage, which has been shown in previous analyses to significantly increase post-TLI payload performance.

The first option uses NLS 1.5 stage derived boosters (six engine boattail) for maximum NLS commonality (1.5 stage already a stand-alone vehicle). This option utilizes a single J-2S engine and delivers 60 t to TLI. Two two-engine F-1A boosters are added to the core with a five-RL-10A4 upper stage, for maximum commonality with currently proposed concepts for Lunar Transfer Systems. This option only delivers 54 t to TLI. Addition of a third engine and an SSME upper stage delivers 83 t to TLI. This launch vehicle is the reference upon which most of the structural and stability and control analysis is performed. However, when the payload requirement increases to 93 t, this option (even with a separate upper stage/TLI stage) cannot meet the requirement with two boosters. Four two-engine boosters are strapped to the core to meet the new requirement. Four two-engine boosters are used rather than two four-engine boosters partly because the 4 G acceleration constraint requires booster engine shut-down in flight. Having four boosters provides more throttling and shut-down flexibility in order to control dynamic pressure and acceleration constraints. This configuration is the current NLS-derived reference.

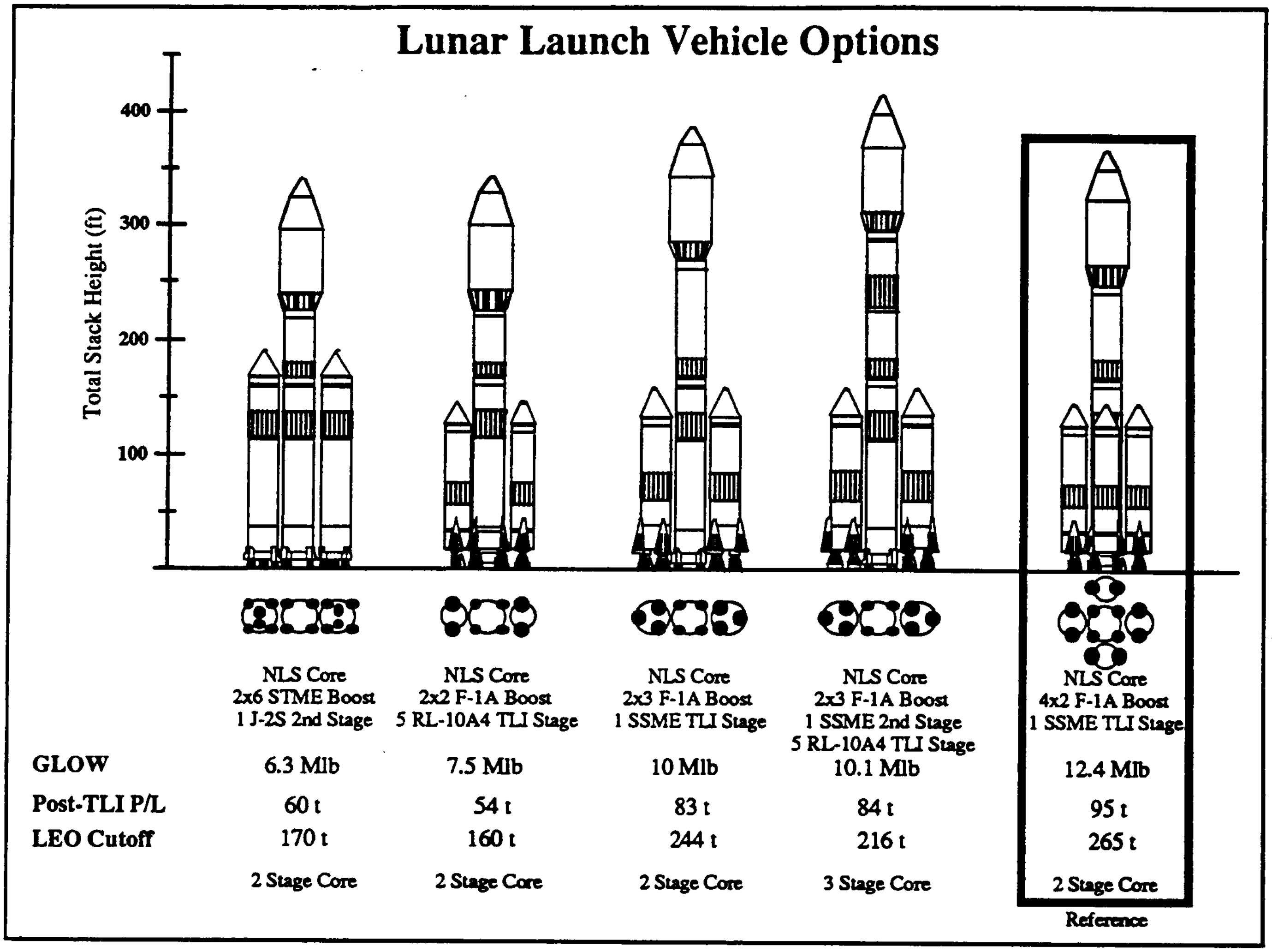


Figure 7.1.1-1 Reference NLS-Derived Configuration Evolution

Figure 7.1.1-2 shows the sensitivity of post-TLI payload to booster configuration and upper stage engine type and propellant load. All options utilize the NLS derived core (1.69 Mlbm propellant load and 4 STMEs). All two-booster options have three F-1A engines per element. The four-booster option has two F-1A engines on each strap-on booster. The addition of a third stage, that is dedicated to the TLI burn only, does not significantly increase payload over a suborbitally ignited combination second stage and TLI stage. The result is due to the fact that the addition of the extra mass that the second stage has to inject into LEO is increased over the single stage concept containing two separate tanks, thrust structure, and an extra interstage, without increasing the thrust of the second stage. It had also been shown in previous analysis that using an STME with a 45:1 expansion ratio on the upper stage would produce a curve of the same slope as an SSME, but approximately 10 t (22 Klbm) lower in delivered payload mass and a 91 t (200 Klbm) greater propellant requirement due to the STME's lower Isp. An STME with a vacuum skirt expansion ratio of 65:1 will split the difference in half. It has also been shown that the performance characteristics of two J-2S engines is similar to those of one STME with a 45:1 nozzle expansion ratio. RL-10A4 engines having a vacuum thrust of 91.2 KN (20.5 Klbf) each do not have sufficient thrust for the payload required.

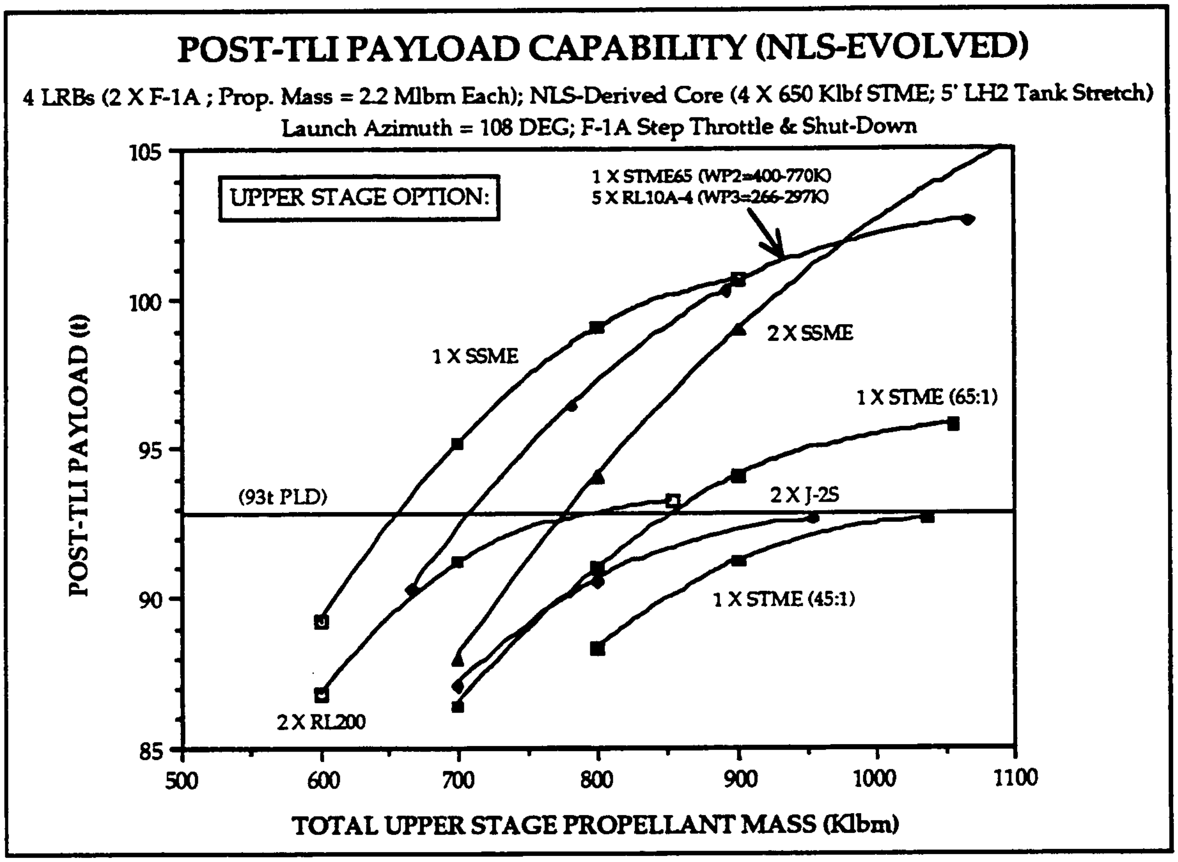


Figure 7.1.1-2 Payload Sensitivity to NLS-Derived TLI Stage Sizing

7.1.2 Common NLS Derived Core/Booster Diameter Approach

In order to maximize vehicle element commonality, minimize vehicle dry weight, minimize structural and main propulsion subsystem design complexity, and to allow for performance growth options; all while adhering to facility constraints at the Kennedy Space Center (KSC), NLS-derived configurations have been designed and assessed that utilize a common diameter dimension for the boosters, core, and TLI stage. The ground processing facility physical limitations end up imposing a fundamental limitation on the design of the vehicle by limiting the vehicle's length in order to utilize the current Vehicle Assembly Building (VAB). A variety of design options are assessed for each of the NLS-derived configuration elements, as well as for manufacturing methods and vehicle performance assessments. A three-stage core vehicle concept also is assessed, which contains the basic NLS core vehicle, a new LOX/LH_2 second stage, and a LOX/LH_2 TLI stage, in addition to two LOX/RP-1 boosters. This configuration helps to identify any performance payoffs for replacing two parallel-burn boosters with one series-burn upper stage.

7.1.2.1 Groundrules and Assumptions

The two most overriding groundrules are the use of common core vehicle and booster tank diameters, and a length limit on the core vehicle LH_2 tank to no more than 4.6 m (15 ft) greater than that for the SSP ET. The intent of those groundrules is to limit design, development, test, and evaluation (DDT&E) costs. The 4.6 m (15 ft) tank extension limit is based upon minimizing the redesign cost impact to an LH_2 tank static load cell at MAF. It has been identified that 3 m (10 ft) extensions to the ET LH_2 tank can be accommodated without any design impacts to the load cell. A 4.6-5.5 m (15-18 ft) extension can be accommodated with minimum to moderate cell modifications. Extensions greater than 5.5 m will result in substantial cell modifications, and equate to the cost of a new check-out cell. By freezing the core and booster tank diameters to that of the ET and by limiting the length of the core LH_2 tank, the booster attach point locations become defined a priori. Limiting the attach points therefore limits the booster propellant loading.

7.1.2.2 Design Options

<u>Shroud</u>

The biconic shroud is utilized, as it was the reference at the start of the analysis activity, with the associated mass properties. NLS forebody aerodynamics representing a Titan IV biconic shroud are also used.

<u>TLI Stage</u>

Three engine types are assessed: J-2S, STME, and SSME. Two-engine combinations are sized for use of STMEs or SSMEs, while use of six or eight J-2Ss is sized. Due to anticipated main propulsion feed system and thrust structure complexities associated with engine clusters of greater than four, the J-2S was dropped as a viable candidate. The mass properties are determined through the use of a mass fraction derived from the S-IVB stage, that was a function of the TLI stage propellant load. It is recognized that the S-IVB used common bulkheads for the propellant tanks, while the NLS-derived TLI stage does not. Thus the TLI stage mass fraction is slightly optimistic, although a 10 percent inert mass margin is also accounted for. A bottoms-up mass properties assessment based upon those used by NLS is to be performed at a later date. Two propellant loads are sized: one for a minimum amount based upon a dome-to-dome LOX tank (ET diameter), which gives 267.6 t (590 Klbm) of propellant, and a performance-optimal propellant loading, which gives 345 t (760 Klbm) of propellant when adhering to the VAB high bay vertical clearance limit.

Second Stage

Two engine types are assessed: SSMEs and STMEs. Two-engine combinations are sized. The mass properties are determined through the use of a mass fraction derived from the S-IVB stage, that is a function of the TLI stage propellant load. Three propellant loads are sized, corresponding to 1.5, 3.0, and 4.6 m (5, 10, and 15 ft) extensions to the LH_2 tank.

NLS Core

The NLS reference HLLV core vehicle is used, which contains four STMEs. NLS Cycle 0 mass properties are used and three propellant loads are sized, corresponding to 1.5, 3.0, and 4.6 m extensions to the LH_2 tank.

Boosters

The F-1A engine is used at two maximum thrust levels: 8 MN (1.8 Mlbf) and 8.9 MN (2.0 Mlbf) sea level thrust. Three and four-engine combinations are sized. The propellant loading is sized based upon the location of the core vehicle's attach struts, which was a function of the LH_2 tank stretch quantity. The mass properties are determined through the use of a mass fraction derived from the S-IC stage, that is a function of the TLI stage propellant load. Four different booster engine layouts are assessed for controllability, structural, and plume heating issues.

Manufacturing Methods

In order to minimize manufacturing and tooling costs, each of the vehicle elements utilize common stage diameters, common tank domes, common intertanks, common interstages (where applicable), and separate propellant tank bulkheads. The relative size benefit (and thus performance benefit) of utilizing common propellant tank bulkheads is assessed for each of the stage elements.

Vehicle Performance Assessments

A three-degree-of-freedom simulation and optimization tool is used to assess the nominal ascent performance of the candidate vehicle configurations and to help in refining vehicle sizing. The ascent trajectories are optimized subject to dynamic pressure and thrust acceleration constraints. The dynamic pressure constraint is adhered to during ascent via two methods: trajectory lofting and stage engine throttling. The acceleration constraint is adhered to via two methods: stage engine throttling and engine shut-down. NLS Cycle 0 aerodynamics (forebody and base effects) are also used.

7.1.2.3 Assessment Results

Stack Lift-off Thrust-to-Weight Ratio

The number of engines to have on each booster is one of the first key design parameters needing to be assessed. An assessment of lift-off thrust-to-weight ratios is performed as a function on number of boosters, number of booster engines, booster engine thrust level, booster propellant load, and core stage propellant load. The candidate vehicle configurations consist of a core stage, a TLI stage and either two or four boosters strapped onto the side of the core stage. Two propellant loads are used on the core and booster stages. Two, three or four F-1As are used on each booster. The F-1A engines are run at two sea level thrust values. A lift-off thrust-to-weight ratio of 1.25 is considered to be the minimum acceptable value. Engine-out capability at lift-off is groundruled to not be a requirement.

The conclusion reached is that vehicle configurations with two F-1As per booster do not have sufficient thrust to be viable designs. Therefore, vehicle configurations with three and four F-1A per booster are used in the analysis. Figure 7.1.2.3-1 summarizes the results of the thrust-to-weight assessment.

STACK LIFT-OFF THRUST-TO-WEIGHT RATIO

	Number of F-1As on a Booster		
F-1As @ 1.8E06 lbf (sea level)	2 F-1As	3 F-1As	4 F-1As
Nominal Core and 2 Boosters	1.051	1.444	1.833
Stretched Core and 2 Boosters	0.951	1.415	1.784
Nominal Core and 4 Boosters	1.099	1.564	2.023
Stretched Core and 4 Boosters	0.989	1.408	1.822

	Number of F-1As on a Booster		
F-1As @ 2.0E06 lbf (sea level)	2 F-1As	3 F-1As	4 F-1As
Nominal Core and 2 Boosters	1.140	1.558	2.010
Stretched Core and 2 Boosters	1.032	1.547	1.956
Nominal Core and 4 Boosters	1.205	1.722	2.231
Stretched Core and 4 Boosters	1.084	1.550	2.010

Rule of Thumb: Minimum nominal thrust-to-weight @ lift-off >/= 1.25

Conclusion: Vehicle configurations with 2 F-1As per booster do not have sufficient thrust to be viable designs.

Figure 7.1.2.3-1 Lift-Off Thrust-to-Weight Ratio Assessment Summary

Stage Mass Fraction Derivations

Propellant mass fractions are used as a means for determining vehicle stage element dry mass and usable propellant values. In order to enhance the fidelity of the mass properties calculations, the applicable mass fractions are computed as a function of the desired stage usable propellant load for each of the candidate engine options.

Engine Layout

A common engine orientation is utilized for either 2-booster or 4-booster NLS-derived configurations. Figure 7.1.2.3-2 shows the reference engine layout that result from a qualitative analysis of the following primary design drivers: engine gimbal clearance (avoiding bell-to-bell hard-over collisions), control authority, thrust structure complexity, and attach strut length penalties. Convective plume heating is acknowledged to be a secondary design consideration at the time of the analysis, and will require further assessment at a later time.

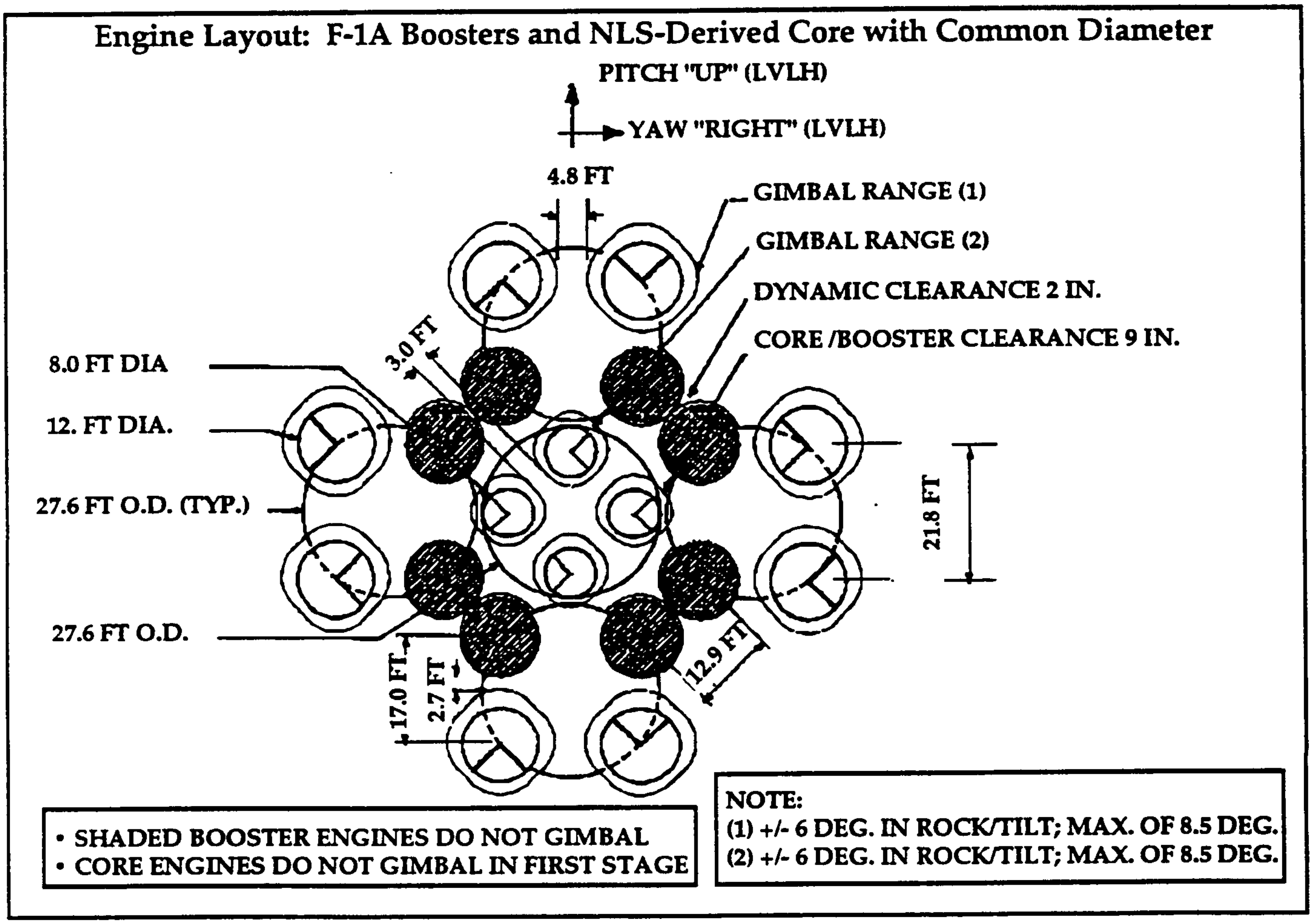


Figure 7.1.2.3-2 Layout of Booster and Core Vehicle Engines

Placing the booster engines on the propellant tank perimeter simplifies the thrust structure required to shear the thrust loads into the aft propellant tank (RP-1 tank), but restricts how close the boosters can be clustered about the core vehicle. While being highly desirable to accept the STME positioning as currently baselined by the NLS program for the NLS-1 vehicle, it is found that the best compromise between booster and core engine locations and attach strut length is to locate the STMEs slightly inboard of the core tank perimeter. A minimum dynamic clearance of two inches is maintained between either booster or core engines, to allow for thrust vector control actuator dither. The desired locations of the engines on the boosters and core are closely coupled with the design of the thrust vector control subsystem, and become part of an iterative solution when considering attach strut lengths and plume heating. If convective plume heating between F-1A pairs (upper or lower engine pairs, in a local vertical/local horizontal sense) requires the F-1As to be spaced farther apart, the boosters will not be able to be placed as close to the core vehicle, necessitating longer attach struts and more core vehicle skin stiffening at the attach struts.

The booster engines are spaced relative to each other to allow for all four F-1As to be gimballed if the booster is to be used as a stand-alone launch vehicle. The resulting gimbal traces of either the are then designed to be within two inches of each other. It is also assumed that aerodynamic fairings will be required for each booster engine, since their locations on the booster perimeter will place the engine bells into the freestream flow during first stage ascent. Since the core engines are not required to gimbal for thrust vector control during first stage, their inboard location on the core vehicle will shield them from the first stage freestream flow, therefore not requiring the use of aerodynamic fairings. Removal of core engine fairings allows the boosters to be placed closer to the core.

<u>Vehicle Description and Performance Summary</u>

Figure 7.1.2.3-3 summarizes the four candidate lunar HLLV concepts that resulted from the sizing and assessment of over thirty different vehicle combinations. Two of the lunar vehicles used four 4-engine boosters strapped onto an NLS-derived core that used the NLS reference five foot extension to the LH_2 tank from the Space Shuttle External Tank (ET) dimensions. The two vehicles utilized different TLI stage propellant loadings: one representing the minimum propellant load when utilizing two LOX tank domes together, 263 t(580 Klbm) propellant load; and one representing the maximum TLI stage propellant load that still kept the total core vehicle length to a value below the VAB highbay door vertical clearance limitation of 119-122 m (390-400 ft), which was a 345 t (760 Klbm) propellant load. One of the lunar configurations utilized an extra 3 m (10 ft) of length to the NLS reference LH_2 tank, or a 4.6 m (15 ft) extension over the current ET's LH_2 tank length. That particular configuration utilized the minimum TLI stage propellant loading, in order to assess the effect to payload mass of increasing the core vehicle's propellant load instead of the TLI stage's propellant load, for a fixed number of engines on each stage. The results showed that more payload could be gained by increasing the core propellant load instead of the TLI stage's propellant load. A fourth lunar vehicle configuration used only

two boosters, but added a second stage on top of an NLS core stage with a 3 m LH_2 tank stretch. The intent is to assess the payload gain for placing the delta velocity capability into an upper stage instead of in two extra boosters, for a fixed TLI stage propellant load. The results show that from a performance standpoint, it is more effective to use two extra boosters instead of a second stage. The three-stage core with two boosters meets the payload goal but causes the VAB high bay door clearance limit to be exceeded. Figures 7.1.2.3-4, 7.1.2.3-5, 7.1.2.3-6, and 7.1.2.3-7 summarize the dimensions, performance, and element mass properties of the candidate configurations.

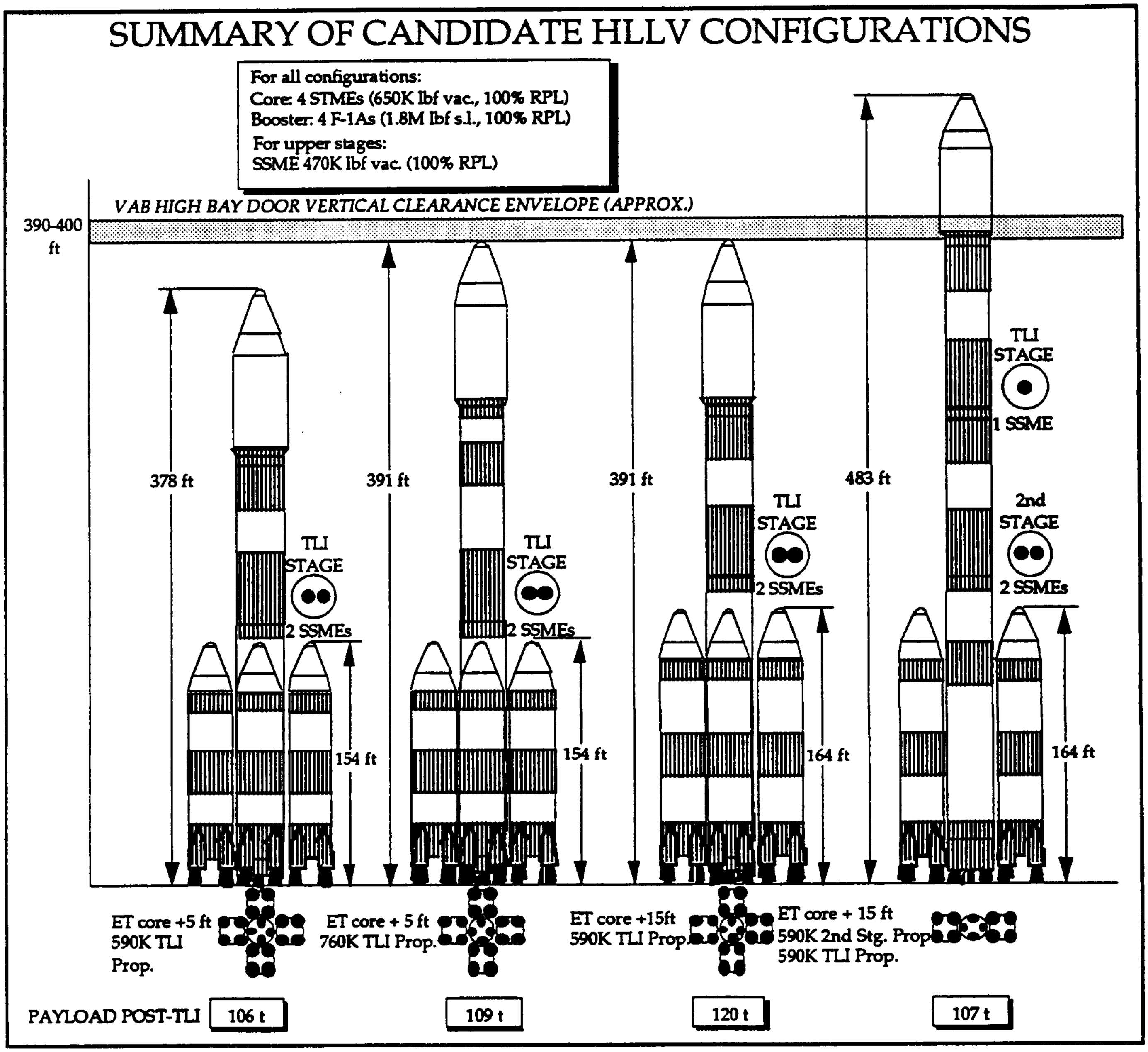
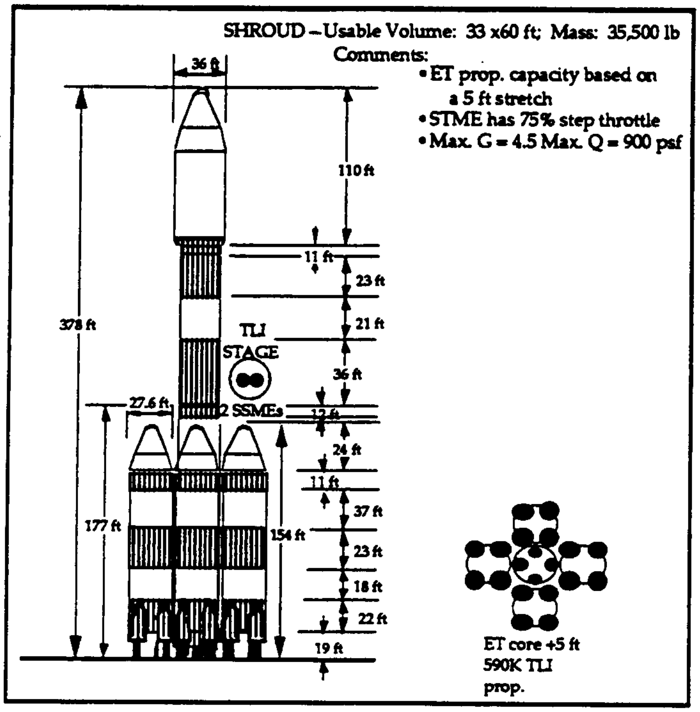


Figure 7.1.2.3-3 NLS-Derived Configuration Summary

Lunar Vehicle: Standard NLS Core w/ LOX/RP1 Boosters and TLI stage (590 k lbm propellant)



Payload: 235 K lbm (106 t)
Final Position: TLI

GLOW: 15,359,000 lbm

CORE:
Inert Mass: 187.8 K lbm
Propellant Mass: 1.69 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 173 ft
Diameter: 27.6 ft
Reusability: N.A.

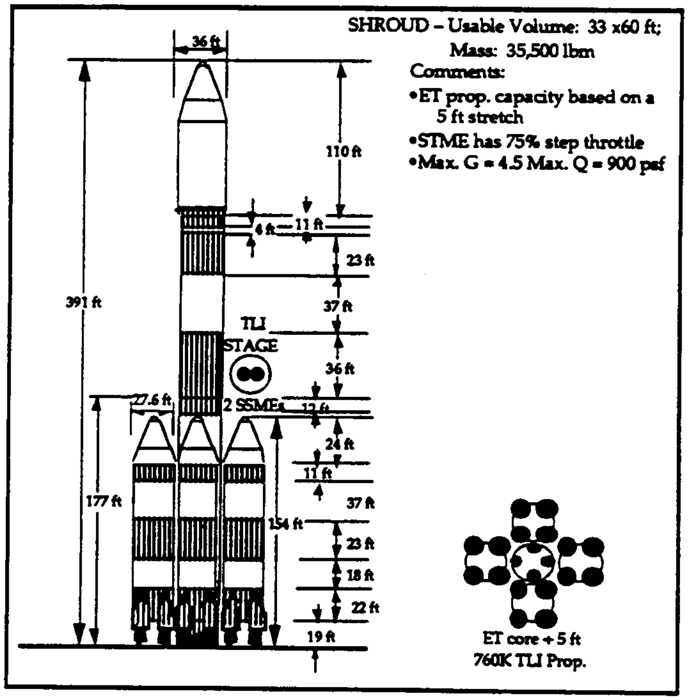
2nd Stage:
Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A,
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusablility: N.A.

BOOSTER:
Number/Type: 4/ET+ 5 ft
Inert Mass: 235 K lbm
Propellant Mass: 2.90 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 154 ft
Diameter: 27.6 ft
Reusablility: N.A.

TLI Stage:
Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusablility: N.A.

Figure 7.1.2.3-4 Two-Stage Configuration: 106 Ton Payload Class

Lunar Vehicle: Standard NLS Core w/ LOX/RP1 Boosters and TLI stage (760 k lbm propellant)



Payload: 240 K lbm (109 t)
Final Position: TLI

GLOW: 15,551,000 lbm

CORE:
Inert Mass: 187.8 K lbm
Propellant Mass: 1.69 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 173 ft
Diameter: 27.6 ft
Reusability: N.A.

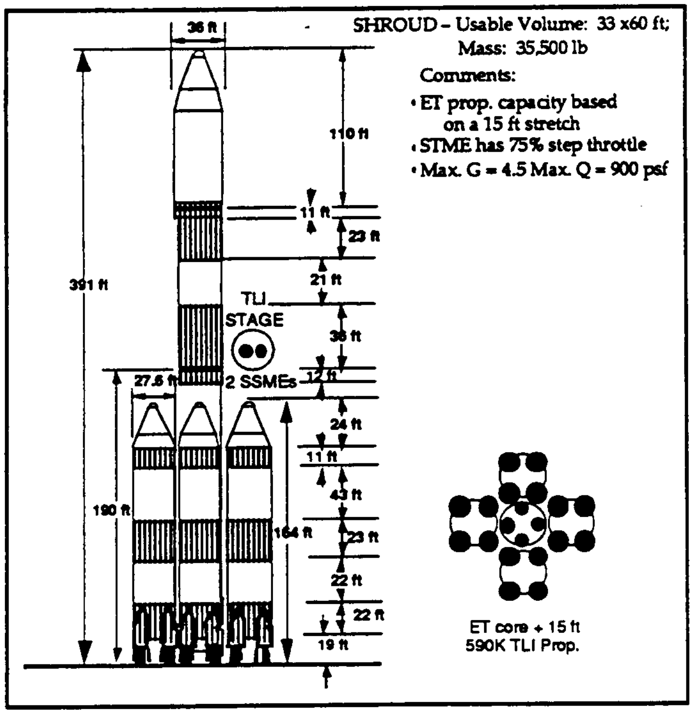
2nd Stage:
Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusablility: N.A.

BOOSTER:
Number/Type: 4/ET+ 5 ft
Inert Mass: 235 K lbm
Propellant Mass: 2.90 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 154 ft
Diameter: 27.6 ft
Reusablility: N.A.

TLI Stage:
Inert Mass: 95.9 K lbm
Propellant Mass: 0.76 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5sec
Engine Exit Dia.: 96 in.
Length: 101 ft
Diameter: 27.6 ft
Reusablility: N.A.

Figure 7.1.2.3-5 Two-Stage Configuration: 109 Ton Payload Class

Lunar Vehicle: Stretched NLS Core w/ LOX/RP1 Boosters and TLI stage (590 k lbm propellant)



Payload:	265 K lbm (120 t)
Final Position:	TLI

GLOW: 17,077,000 lbm

CORE:
Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

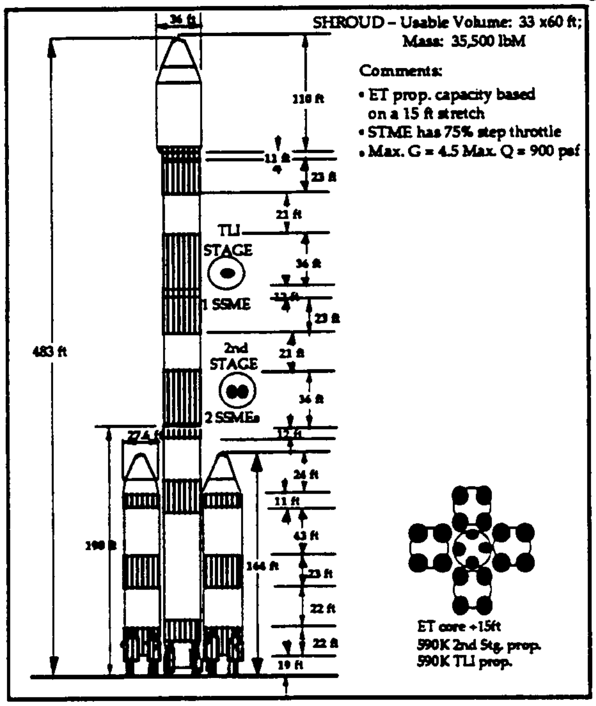
2nd Stage:
Inert Mass: N.A.
Propellant Mass: N.A.
Propellant Type: N.A.
Engine Type/No.: N.A.
Vac Thrust (ea.): N.A.
Vac ISP: N.A.
Engine Exit Dia.: N.A.
Length: N.A.
Diameter: N.A.
Reusablility: N.A.

BOOSTER:
Number/Type: 4/ET+ 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP: 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusablility: N.A.

TLI Stage:
Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusablility: N.A.

Figure 7.1.2.3-6 Two-Stage Configuration: 120 Ton Payload Class

Lunar Vehicle: Stretched NLS Core w/ LOX/RP1 Boosters, 2nd Stage, (590 k lbm propellant) and TLI stage (590 klbm propellant)



Payload:	236 K lbm (107 t)
Final Position:	TLI

GLOW: 10,686,000 lbm

CORE:
Inert Mass: 206.4 K lbm
Propellant Mass: 1.86 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: STME/4
Vac. Thrust (ea.): 650 K lbf
Vac ISP: 428.5 sec
Engine Exit Dia.: 96 in.
Length: 186 ft
Diameter: 27.6 ft
Reusability: N.A.

2nd Stage:
Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/2
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusablility: N.A.

BOOSTER:
Number/Type: 2/ET+ 15 ft
Inert Mass: 251 K lbm
Propellant Mass: 3.26 M lbm
Propellant Type: LOX/RP1
Engine Type/No.: F-1A/4
Vac Thrust (ea.): 2020 K lbf
Vac ISP 304.2 sec
Engine Exit Dia.: 143.5 in.
Length: 164 ft
Diameter: 27.6 ft
Reusablility: N.A.

TLI Stage:
Inert Mass: 78.9 K lbm
Propellant Mass: 0.59 M lbm
Propellant Type: LOX/LH2
Engine Type/No.: SSME/1
Vac Thrust (ea.): 470 K lbf
Vac ISP: 452.5 sec
Engine Exit Dia.: 96 in.
Length: 88 ft
Diameter: 27.6 ft
Reusablility: N.A.

Figure 7.1.2.3-7 Three-Stage Configuration: 107 Ton Payload Class

Flight Mechanics

Dynamic pressure limiting is required for the candidate configurations for structural and thermal considerations. Trajectory lofting is a more direct way of controlling dynamic pressure but it incurs significantly higher performance losses, as reflected in gravity and thrust vector velocity losses. Throttling is a less direct controller, but results in much less gravity and thrust vector losses. Thrust acceleration limiting (G limiting) is required for structural considerations, and can be accomplished via engine throttling and engine shut-down. Throttling is effective when minimum rated power levels of 65-75 percent are available. Engine shut-down is a much less precise controller, and requires multiple shut-downs for moment ballance and thrust vector loss minimization. Only small exceedances over 4 Gs are observed in the candidate configurations.

Manufacturability

Use of common propellant tank, intertank, aft skirt, and forward skirt/interstage piece-parts allows reasonable vehicle configurations to be designed, without incurring inordinate performance losses. The associated manufacturing cost savings would easily justify the performance non-optimality of the candidate designs. If the NLS-derived core has its propulsion module integrated at MAF, then the following tooling/manufacturing cost impacts are predicted:

- 10 foot LH_2 tank stretch: approximately $65 million in non-recurring costs
- 15 foot LH_2 tank stretch: approximately $120 million in non-recurring costs

Tooling and manufacturing certification requirements and LOX compatibility issues remain to be answered for the use of aluminum-lithium, although 8090 aluminum-lithium is currently being used to manufacture the Titan IV payload shroud conical adapter. Common bulkhead manufacturability and cost issues also remain for use of common or nested propellant tank bulkheads.

Launch Operations

If a new VAB is to be built for SEI applications, then the core vehicle should be stretched to its maximum length, as constrained by recurring cost, and SSMEs should be used on a second stage and the TLI stage for payload maximization. The combination of the VAB highbay door vertical clearance constraint and the final mission payload requirements may require the use of common propellant tank bulkheads. Widening of the VAB high bay door opening beyond the current 71 feet will be more cost-effective than rotating the orientation of a four-booster stack on the mobile launch tower (MLT), in order to roll into the VAB. A rotation will work for VAB clearance, but will incur significantly higher design and cost impacts at the launch pad. A 45 degree rotation of the vehicle on the MLT will be feasible from a T-0 umbilical and ascent performance standpoint. Utilizing one type of engine on the core vehicle would be preferred to minimize ground processing costs, and pre-launch thermal conditioning of any air-startable engines can be accomplished but complicates ground processing.

Cost

Cost benefit trade-offs remain to be performed on marginal cost of design, development, test, and evaluation cost reduction versus recurring cost reduction, for the candidate common-diameter configurations. The cost of providing throttle-down capability on the boosters would most likely be offset by the resulting increase in payload capability, as costed in terms of equivalent numbers of flights. The VAB high bay doors can be economically modified, up to a point, to accommodate booster height, but not economically for core vehicle height.

7.1.3 Propellant Thermal Control Trades

To provide a historical perspective, the thermal control system (TCS) for the Saturn SIV-B stage and the Space Shuttle External Tank (ET) has been summarized in Figure 7.1.3-1.

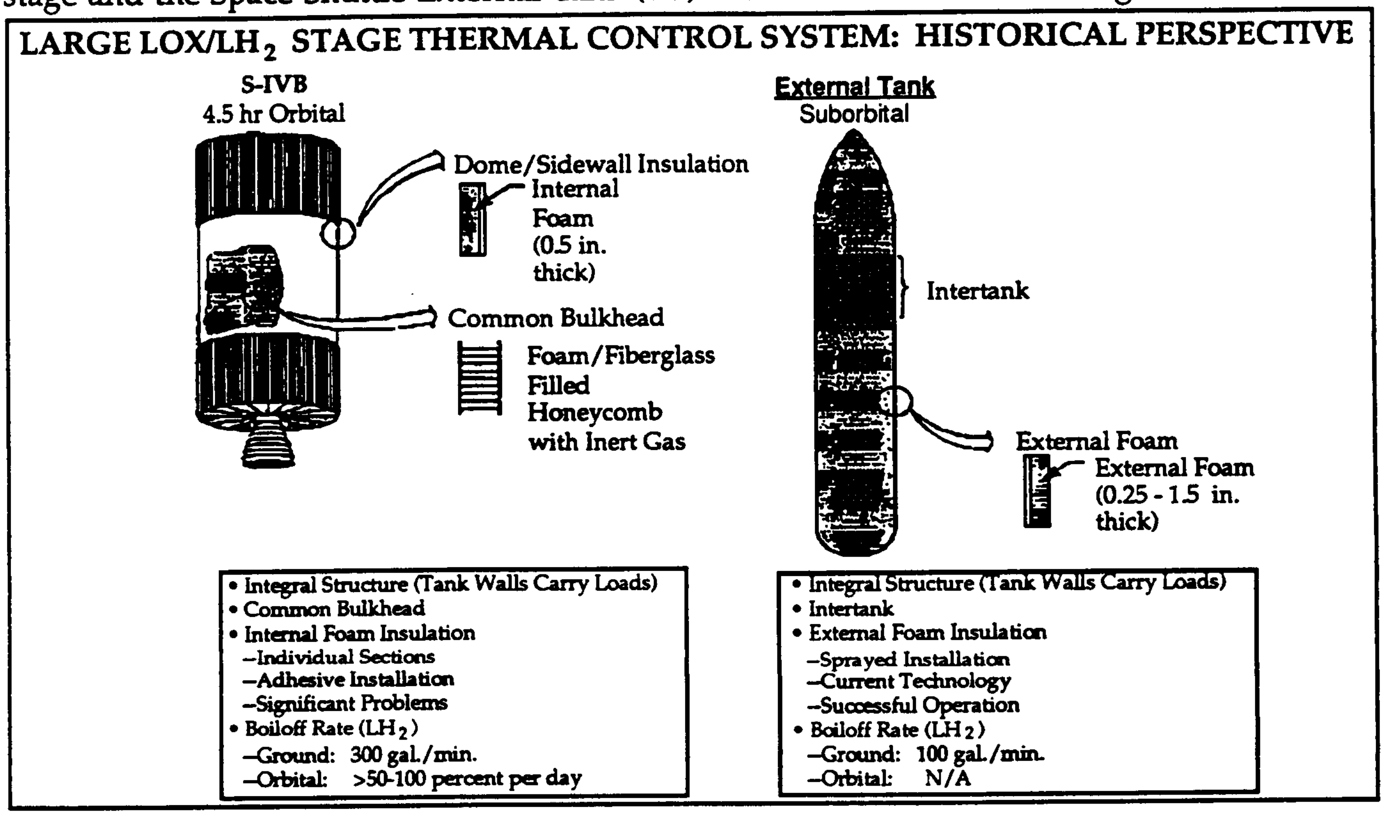
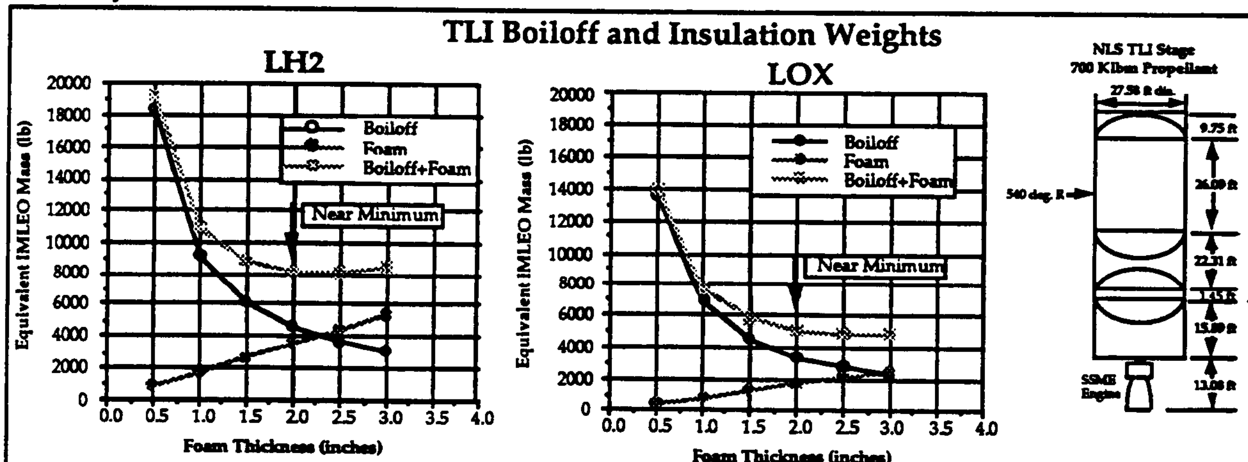


Figure 7.1.3-1 Thermal Protection Historical Perspective

Tank walls on both stages are an integral part of the load carrying structure. The S-IVB tanks has a common bulkhead, while the ET tanks are separated by an intertank. No significant thermal preference between separate tanks and a common bulkhead is anticipated for a 4.5 hour mission. The S-IVB has internal foam insulation and the ET tanks have external sprayed on foam insulation (SOFI). The SIV-B experiences much higher boiloff rates than expected due to cracks in the internal foam which form when exposed to liquid hydrogen temperatures. The ET insulation has been much more reliable, although it does not go all the way to orbit. Not shown is the Centaur, a contemporary to both these stages. Although a much smaller stage than the two pictured here, the Centaur utilizes both foam and a few layers of multilayer insulation, has a common bulkhead and the insulation is a low density glass felt, with a radiative shield.

The top level design trades are generally based on previous analytical experience and historical flight data. Because of the short storage time, the thermal design of the TLI stage for the Single /Direct launch mission can be an updated version of the S-IVB Stage. Propellant tanks can be an integral part of the load carrying structure and a common bulkhead would be thermally acceptable. ET-type foam insulation will be applied to the tank exterior, instead of to the inside as on the S-IVB. A meteoroid shield is not anticipated. A liberal boiloff rate of 25 percent per day is acceptable as a design target. Figure 7.1.3-2 illustrates the results of a simplified analysis to determine the optimum insulation thickness for the 317.5 t (700 Klbm) TLI propellant load. The effect of the launch environment on the SOFI external optical properties is uncertain, so a relatively warm external surface temperature of 540 R is used for calculations. A more detailed TRASYS/SINDA model is under construction to more accurately define the orbital environment and thermal response of the vehicle. Boiloff and insulation masses for the 4.5 hour mission are multiplied by mass exchange factors to obtain the equivalent initial mass in low earth orbit (IMLEO). The total foam insulation plus the 4.5 hour boiloff losses are then plotted to determine the insulation thickness that produces the minimum total mass. The optimum foam thickness for both the LH_2 and LOX tanks is approximately 5 cm (2.0 inches). There are no manufacturability issues associated with applying a 5 cm thick SOFI layer.



Results - 4.5 Hr. Mission

2.0 inches of Sprayed on foam insulation (SOFI) is near the minimum foam+boiloff weight and results in the following:

	Boiloff	Foam mass (lbm)
LH2:	1.0 %/hr	1682
LOX:	0.13 %/hr	789

ASSUMPTIONS

SOFI Properties :
- K = 0.014 Btu/hr ft R
- r = 2.4 lb/ft^3
- External Temp. = 540 deg. R
- Full tanks
- Mass Exchange Factors:
  Boiloff: 1.11
  Foam: 2.13

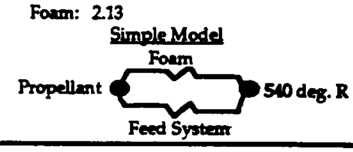
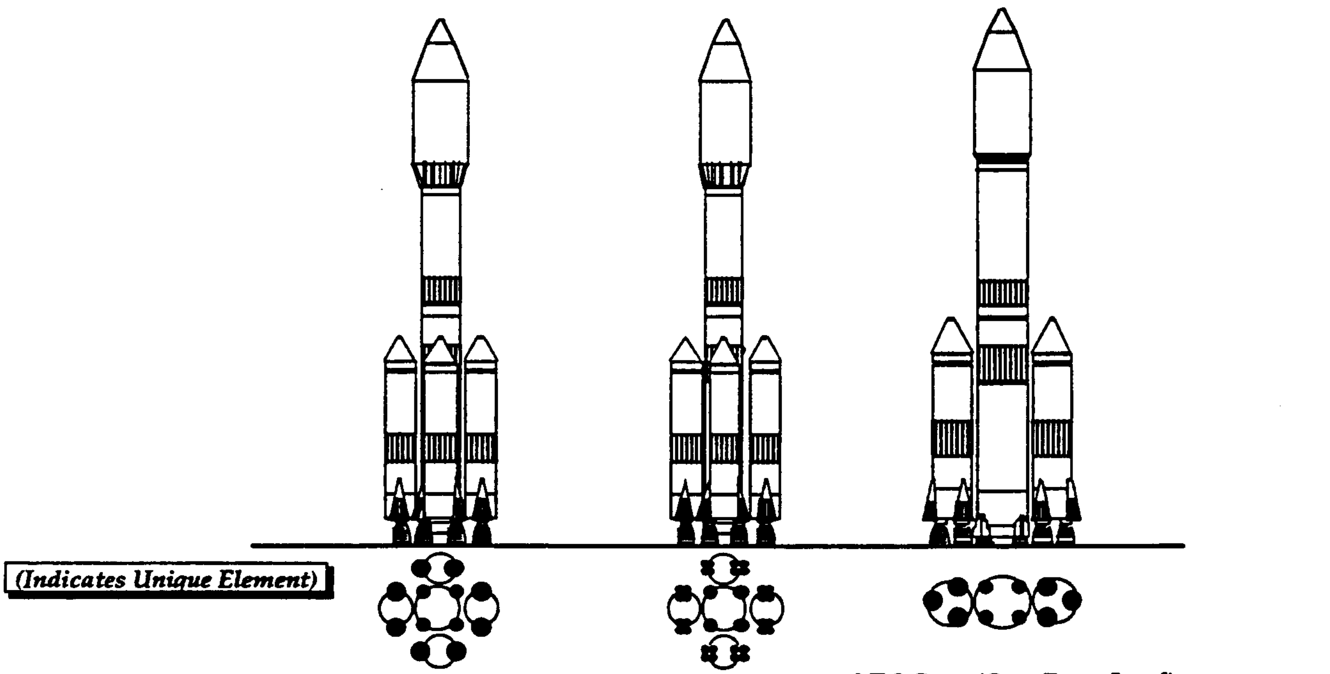


Figure 7.1.3-2 NLS-Derived TLI Stage Thermal Protection Assessment

7.1.4 Alternate Launch Configurations

Figure 7.1.4-1 shows several alternate configuration options. All options ignite the booster and core engines at lift-off. The first uses the reference NLS derived option as a base, but replaces the four STMEs with four expendable SSMEs. Resizing the boosters and TLI stage results in a 5 t payload increase. The second option replaces the F-1A engines on the boosters of the reference NLS derived option with Energia (Confederation of Independent States) RD-170 engines. The boosters and TLI stage are resized for this option, which also results in a 5 t performance increase. The final vehicle is an optimized propellant load core (4 STMEs) with two three-F-1A engine boosters. The core diameter is constrained to 10 m (33 ft), which is the Michoud Assembly Facility physical limit without requiring a completely new building. This allows the two booster option to meet the 93 t payload requirement.

Alternate Launch Configurations



	NLS Core *(4 Expend. SSME)* 4x2 F-1A Boost 1 SSME TLI Stage	NLS Core *4x2 RD-170 Boost* 1 SSME TLI Stage	NLS Core *(Opt. Prop. Load)* 2x3 F-1A Boost 1 SSME TLI Stage
GLOW	12.4 Mlb	10.9 Mlb	10.8 Mlb
Post-TLI P/L	100 t	100 t	95 t
LEO Cutoff	277 t	282 t	275 t
	2 Stage Core	2 Stage Core	2 Stage Core

Figure 7.1.4-1 Other Alternative NLS-Derived Configurations

7.2 Saturn V-Derived

7.2.1 Reference Configuration Trades

Vehicle definition analyses for the Saturn V derived HLLV configuration were performed for two distinct classes of performance capability over the course of this study phase: a 76 t post-TLI payload capability class and a maximum payload capability class (vehicle height constrained). The 76 t HLLV payload delivery requirement was specified on the basis of lunar surface payload requirements and preliminary mass estimates for the cargo lander. In order to establish an upper bound on HLLV performance potential, a maximum capability vehicle also is considered. With this case, the vehicle configuration is constrained to an overall height of 125 m (410 ft) in order to enable use of the existing VAB at KSC. The 76 t capability requirement, initially imposed on the HLLV, was subject to periodic revision throughout the course of study as updated estimates of cargo and lander masses became available. Subsequent to completion of the 76 t class vehicle definition analyses, the 76 t requirement was replaced with a 93 t requirement, derived on the basis of piloted mission lander mass estimates. Since the 93 t requirement closely coincides with the performance capability of the maximum payload class vehicle under study, no additional definition analyses are required. The final reference Saturn V derived configuration is selected on the basis of results of the maximum payload class vehicle definition analyses.

Approach

The overall process utilized for HLLV definition is illustrated in Figure 7.2.1-1.

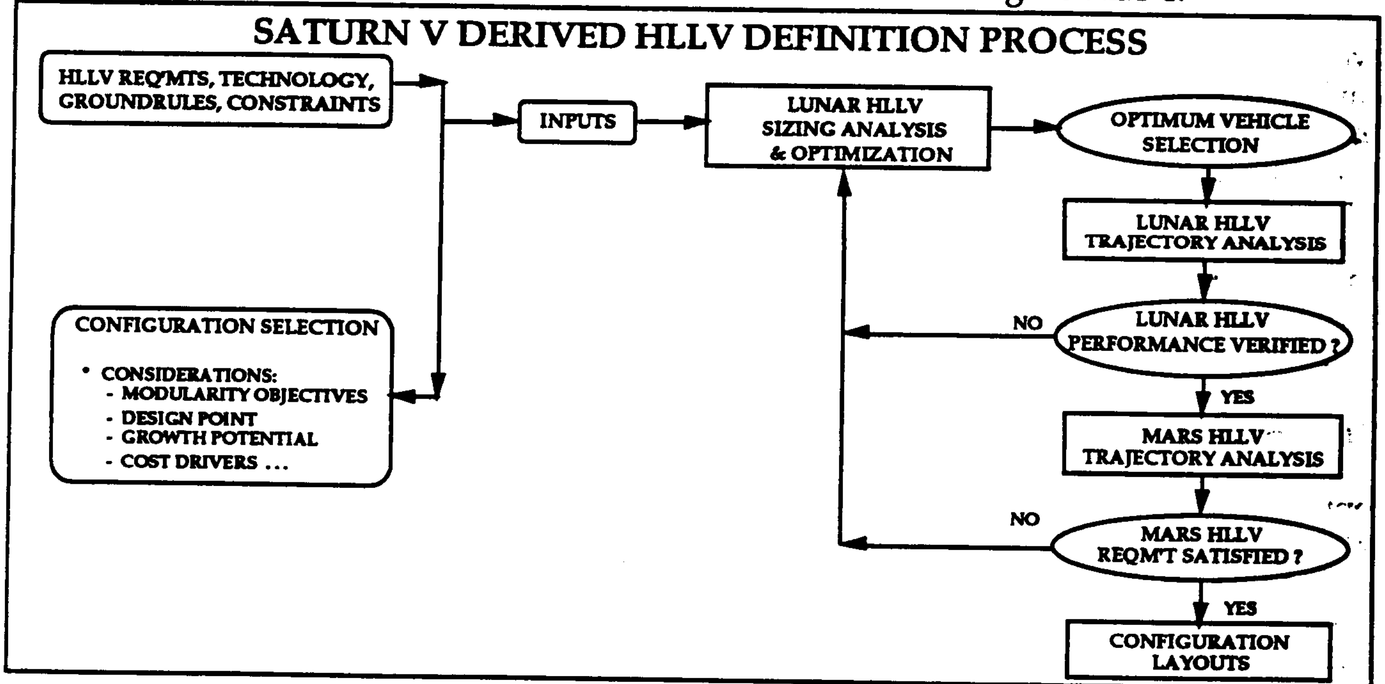


Figure 7.2.1-1 Saturn V-Derived Trade Study Approach

Independent sets of vehicle definition studies were completed in order to produce a matrix of candidate Saturn V derived vehicles for both the 76 t and maximum capability vehicle classes. Integrated HLLV sizing analyses are performed for each vehicle class using the baseline lunar HLLV configuration, consisting of three core stage elements and two booster elements, and the reference F-1A and J-2S propulsion system options. Subsystem level mass properties are predicted for each stage element within the sizing software using empirical weight estimating relationships derived from Saturn V mass properties data. All vehicle definition analyses are completed using the lunar vehicle as the design point, leaving the Mars HLLV performance dependent upon the resulting lunar HLLV stage characteristics. The lunar configurations, however, are scarred with core thrust structure for the four-booster Mars HLLV. Vehicle mass properties, geometry, and first order performance estimates are obtained from sizing software, while final performance capability is estimated from trajectory analysis.

Among the primary vehicle characteristics to be defined from the sizing activity are the number of engines per stage, stage performance, and booster diameter, for an optimized configuration. Vehicle candidates are defined for each capability class through parametric variation of staging velocity conditions, resulting in variation to stage element lengths, for various options of engine numbers per stage. In addition, a booster diameter sensitivity study was conducted during initial sizing analyses for the 76 t class vehicle. During the sizing processes, resulting vehicles are screened to eliminate configurations which violated thrust-to-weight limitations or booster element attach constraints. Thrust-to-weight limitations are considered for both ignition and burnout conditions. A minimum lift-off thrust-to-weight constraint is imposed based upon a study groundrule. Separate constraints are imposed on the upper stages, if operated during ascent, in order to eliminate configurations with unacceptably low ignition thrust-to-weight ratios which would significantly degrade vehicle trajectory performance. Additionally, a burnout thrust-to-weight constraint is imposed on the vehicle's first stage elements in order to ensure that vehicle acceleration levels could be controlled with the use of engine throttling during trajectory simulations. An attach constraint is also imposed for this phase of study to limit the forward booster to core attach location to the forward skirt and nose cone regions of the booster and to the forward skirt and interstage regions of the modified S-IC stage. This constraint serves to screen out vehicle configurations requiring structural attachment to barrel segments of a booster or core stage oxidizer tank, in consideration of the anticipated complexity and cost of this approach.

The resulting vehicle candidates, which satisfy all sizing constraints, are then evaluated on the basis of selected objective variables which served as the criteria for defining the most optimum vehicle for each of the two capability classes. For the vehicles sized to the fixed 76 t payload capability, the selected criterion of minimum vehicle dry weight is used to define the optimum vehicle. For the vehicles sized to a fixed height of 125 m (410 ft), the optimization criterion was maximum payload performance (post-TLI). Overall vehicle reliability was an additional criterion applied to both vehicle capability classes, and was evaluated in terms of the total number of engines required and the number of engine starts required per stage.

Results

The matrix of design cases used to define vehicle options for the two capability classes is presented in Figure 7.2.1-2.

SATURN V-DERIVED LUNAR HLLV DESIGN CASES

PAYLOAD OBJECTIVE (Post - TLI)	BOOSTER DIAMETER	NO. OF ENGINES (Booster, Core Stg I,II,III)	STAGING VELOCITIES
• 76 mt PAYLOAD	260 in., 300 in., 331 in. 260 in. 260 in. 260 in. 260 in.	$\boxed{2,5,5,1}$ 2,5,6,1 2,5,6,2 2,6,6,1 2,6,6,2	150 Combinations • • • •
• MAXIMUM PAYLOAD - 410 ft. Fixed Height HLLV	260 in. 260 in. 260 in. 260 in. 260 in. 260 in.	2,5,5,1 2,5,5,2 $\boxed{2,5,6,1}$ 2,5,6,2 2,6,6,1 2,6,6,2	150 Combinations • • • • •

☐ Reference Vehicle Engine Combination

Figure 7.2.1-2 Saturn V-Derived Design Case Matrix

For each engine configuration considered, vehicle sizing analyses are performed using a domain of staging velocity combinations to define a matrix of candidate vehicles. Variation to the staging velocities provide a convenient method for evaluating acceptable stage length variations for fixed stage diameters. A down-select process is then applied to the vehicles satisfying the design constraints, using the selected optimization criteria. At the study outset, a booster diameter sensitivity analysis was performed for the 76 t vehicle class using an HLLV engine configuration consisting of two F-1A engines per booster, five F-1A engines on the modified S-IC stage, five J-2S engines on the modified S-II stage, and one J-2S engine on the TLI stage. This engine configuration will be denoted 2,5,5,1 in a format to be used throughout the discussion. Three booster diameters are evaluated with this configuration: 6.6 m (260 in), 7.6 m (300 in), and 8.4 m (331 in); the latter corresponding to the Shuttle ET diameter. Results of the analysis show that the 6.6 m (260 in) booster offers the most configuration solutions satisfying the booster attach location constraint. Additionally, the vehicles defined with the larger diameter boosters all result in higher vehicle dry and gross weights than the optimum 6.6 m (260 in) booster vehicle. It is also observed for vehicles using the 8.4 m (331 in) diameter boosters that the required booster burn duration approaches the burn duration of the modified S-IC stage, in order to satisfy the attach constraint. Consequently, for these vehicles, the boosters and modified S-IC stage act effectively as a single series stage element resulting in reduced staging benefits. In consideration of the attach constraint limitations and lower vehicle weights, the 6.6 m (260 in) diameter booster is baselined for the remaining definition studies to be discussed.

Sizing analyses for the 76 t class HLLV indicated that performance objectives and constraints could be met with a minimum dry weight vehicle by using a 2,5,5,1 engine combination on the stage elements. Although this result was anticipated, alternate engine quantities on the core stages were evaluated for comparison and for assessment of vehicle performance and sizing sensitivities. With few exceptions, all vehicles defined for this class of capability resulted in geometries less than 125 m (410 ft) in overall height. The matrix of design cases included vehicles with and without a sub-orbital operation phase of the TLI stage. Vehicles sized with suborbitally operated TLI stages incorporating only a single J-2S engine were observed to have adversely low ignition thrust-to-weight ratios, even with cases where the sub-orbital burn durations were short. As a result, candidate vehicles designed for sub-orbital TLI stage operation generally required two J-2S engines.

Evaluation of the vehicle candidates against the chosen optimization criteria led to selection of a reference HLLV incorporating a TLI stage which is not operated suborbitally. Since these sizing analyses were performed for a fixed payload vehicle, performance benefits associated with TLI stage sub-orbital burn equated to a reduction in total vehicle weights. The dry weight benefits observed with the vehicles incorporating suborbitally operated TLI stages were not considered substantial enough to outweigh reliability considerations associated with an additional engine start and, in most cases, a required additional engine. After evaluation of all vehicles defined, a reference HLLV selected from the candidates using a 2,5,5,1 engine combination provided the best vehicle solution with fewest number of engines. The overall vehicle height was well within the 125 m (410 ft) constraint, since only minor stretches were required for the S-IC and S-II stages. The estimated vehicle mass properties resulted in a lift-off thrust-to-weight ratio of approximately 1.3. Prior to completing refinements to the reference vehicle definition, the HLLV payload requirement was increased to 93 t (205 Klbm) post-TLI and all subsequent activities are focused on the maximum payload capability HLLV.

Evaluation of the matrix of vehicles sized to a fixed height of 125 m (410 ft) lead to selection of a reference HLLV incorporating a 2,5,6,1 engine combination on the stage elements and a TLI stage designed for trans-lunar insertion only. Reference vehicle selection are based upon the criteria of maximum payload performance, in order to establish an upper capability limit with the height constrained HLLV, and vehicle reliability. Vehicle candidates with TLI stages sized for sub-orbital operation are found to have low TLI stage ignition thrust-to-weights for cases using only a single J-2S engine, which reduce the benefits of the additional stage burn. The vehicles evaluated with two-engine TLI stage options all provide sufficient thrust-to-weight ratios, even with the larger TLI stages sized for longer duration sub-orbital burns; however, performance capabilities of these vehicles are not found to exceed the reference HLLV when sized to the fixed 125 m (410 ft) height. Selection of a configuration incorporating six J-2S engines on the modified S-II stage is made after reviewing the payload capability of all candidate vehicles. Although many of the vehicles defined with only five J-2S engines produce reasonable performance capability, none provide the level of payload capability demonstrated with the reference HLLV. From the reference vehicle characteristics, it is evident that performance optimum burn durations are achieved with length modifications of approximately 6.7 m

(22 ft) to both the S-IC and S-II stages. The additional engine on the modified S-II stage, relative to Saturn V, supplies the needed thrust augmentation for adequate thrust-to-weight levels with the increased stage weight. Five F-1A engines on the modified S-IC are found to provide sufficient first stage thrust levels in conjunction with the booster elements. Vehicle candidates using six F-1A engines on the modified S-IC are typically characterized by excessive lift-off accelerations and shortened duration burn times for the booster and modified S-IC stages, in order to maintain acceleration limits. Consequently, no performance benefits are observed with these options.

7.3 Payload Shroud

7.3.1 Aerodynamics and Performance

A trade study was conducted to determine the effect on aerodynamics and performance using the common shroud configuration. The percent increase in axial force over a biconic nose shape versus Mach number is shown in Figure 7.3.1-1.

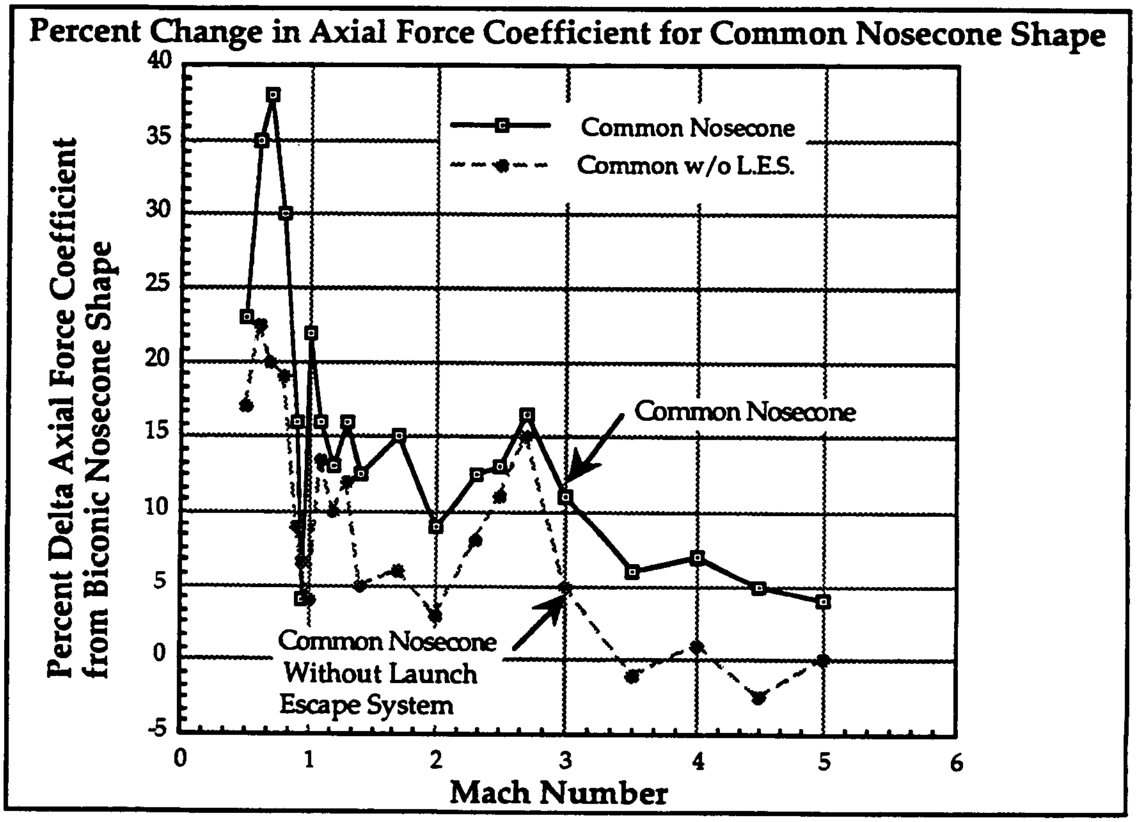


Figure 7.3.1-1 Common Shroud Performance Versus Biconic Design

Data for the common nosecone with and without the launch escape system is also given. The effect of increased axial force on performance for the common nosecone is evident. This effect is relatively minor when compared with overall TLI payload requirements (93 t). No normal force data exists for the common nosecone shape, however, several observations can be made based on historical evidence (e.g., Atlas booster experience). The resulting bending moment on the vehicle is a function of nosecone shape and will be increased for this nosecone configuration due to its large half-angle (42 deg). Turbulence due to detached shock waves also resulting from this large half-angle will likely cause vibration and dynamic stability problems as well as a significant excursion from linear load behavior. Normal force data must be obtained from wind tunnel testing to make an accurate determination of its effects on launch vehicle design.

7.4 TLI Disposal Options

If the TLI stage is left orbiting in the Earth-Moon space after burn-out and separation from the lunar transfer vehicle (LTV), it may become a collision hazard to other spacecraft. To minimize the potential orbital debris problems posed by the spent TLI stage it will be necessary to dispose of the TLI stage. Three disposal options for tank set removal after the TLI burn have been considered, as shown in Figure 7.4-1: the TLI stage could be targeted for Earth reentry and burn-up on the first orbit with a delta velocity cost of 20 m/sec or less; targeting for a lunar impact would incur a delta velocity of about 5 m/sec; and (3) the gravity-assist from a posigrade swing-by of the Moon, at a delta velocity cost of about 30 m/sec, effectively removes the tanks from Earth - Moon space. Each of these options could be implemented in two ways: an avionics/retro package on the TLI stage could perform the targeting maneuver after separation from the LTV; or the maneuver could be made by the LTV system which would then have to be retargeted onto its planned course to the Moon.

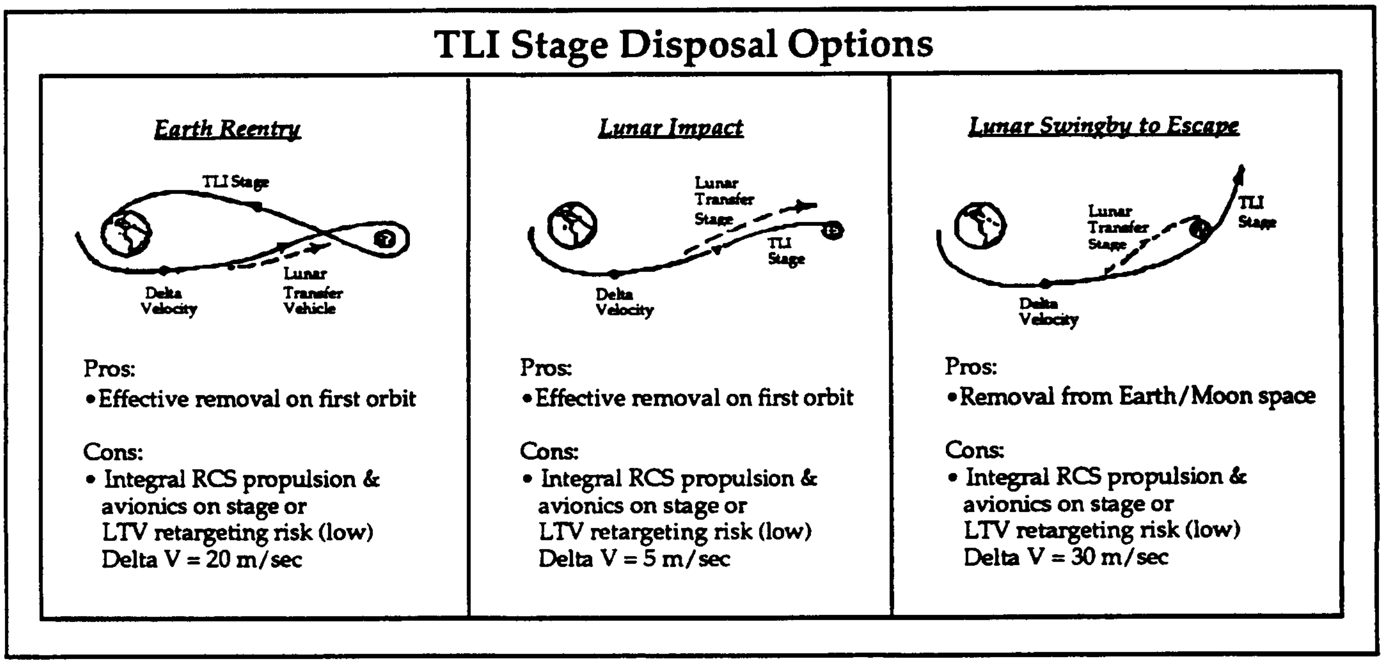


Figure 7.4-1 Candidate TLI Stage Disposal Options

Clearly, the mass penalty for disposal is less if the delta velocity is made by the separated tank which weighs only a fraction of the total vehicle stack after the TLI burn. This would be true even if the tank thrusters used hydrazine (or a solid retro) instead of the high performance LOX/LH_2 propellant. On the other hand, the LTV is designed for accurate propulsion burns, having the full capability of its supporting subsystems, whereas the tank is not. Targeting accuracy may be an important factor in selecting the best disposal option since it would be desirable to make only one maneuver (i.e., a tank is not a spacecraft). A counter-argument against the LTV taking on the disposal/retargeting burden is that it would take the LTV off-course, especially from the nominal free-return trajectory designed for mission abort contingencies, and this would incur some degree of risk should a maneuvering subsystem failure occur. Much more detailed analysis of the mass penalty, accuracy, and risk tradeoffs is required for before a definitive selection of the best disposal option for the TLI stage can be made.

8. Conclusions

Based on the design and assessment results presented herein, two point-design HLLV concepts have been identified that meet the minimum requirement of providing 93 t of payload after performing a nominal ascent trajectory and a TLI burn. The two designs represent a "snap shot in time," and serve merely as a first point of departure for the identification of HLLV requirements for the Single Launch lunar mission profile. Two different programmatic approaches have been shown to produce viable HLLV configurations: one using vehicle elements that are evolved from the current reference NLS family of vehicles, and one using a design approach that is evolved from the original Saturn V concept. The analysis performed to date indicates that there are no significant technological hurdles that must be overcome in order to enable the Single Launch requirements. The two most important design assumptions for the NLS-derived concept are the timely development and certification of the STME and the F-1A, for use on the core vehicle and boosters. A closely related design assumption is the ability to develop and qualify an air-start capability for the SSME, for use on the TLI stage. The two most important design assumptions for the Saturn-derived concept are the timely development and certification of the F-1A and J-2S, for use on the boosters, core first stage, core second stage, and TLI stage. It is acknowledged that the reference payload shroud design for both the cargo and piloted missions is projected to produce excessive unsteady aerodynamic loads on the core vehicle, and further payload packaging and shroud design analyses are required to eliminate the loads issues.

A straw-man development and acquisition process has been identified that meets the intended first-launch date of 1999. The analysis activities documented herein have concentrated on the conceptual design of candidate HLLV configurations due to the current uncertainty in projecting the precise funding, mixed fleet launch vehicle production, and operations environments that will exist in the years immediately preceding and following 1999. More analysis is required, however, to identify programmatic and legislative actions that could be taken that would significantly reduce the risk in meeting the 1999 launch date.

9. Recommendations

Given the preliminary nature of the design and assessment results described above, it is recommended that the following trade studies and analyses be performed in order to further develop Single Launch HLLV requirements that could be used to initiate Phase A design activities:

- Number of engines per booster
- Number of boosters
- Booster attachment concepts
- Common versus separate propellant tank bulkheads
- Type of core and booster engines (including foreign engines)
- TLI engine type
- Number of TLI engines
- Payload shroud configuration and LTV accommodations
- Test plan methodology
- Manufacturing, test, and launch facilities implications
- Alternative payload packaging and shroud design concepts
- Clean-sheet and monolithic (common stage diameters) concepts
- Alternative structural materials
- Vehicle-specific forebody and base effects aerodynamics
- Vehicle-specific distributed loads
- Alternative launch site design implications
- Use of foreign launch vehicle elements
- Cargo and piloted abort scenario development and performance assessments
- Engine-out protection design implications

First Lunar Outpost

Heavy Lift Launch Vehicle

VTOL

Side Entry

- Cylinder
- Flattened Conic
- Bent Conic
- Biconic
- ***Symmetrical Conic***
  - Propellant Tank Design
    - ***Nested Bulkhead***
    - Common Bulkhead
    - ***Separate Bulkhead***
    - Toroidal Tank
    - Cylindrical Tank
    - ***Conical Tank***
- TPS
  - ***TABI/FRSI/FRCI/Carbon-Carbon***
- Structure
  - Structure Type
    - Monocoque
    - ***Skin/Stringer (Tanks)***
    - ***Honeycomb (Unpressurized Structure)***
  - Structural Materials
    - Aluminum Alloys
      - ***Al-Li***
    - Composites
      - ***Graphite-Epoxy***
- Tank Layout
  - ***LH_2/RP-1/LOX (Aft-to-Fwd)***
  - RP-1/LOX/LH_2 (Aft-to-Fwd)
  - ***LOX/LH_2 (Aft-to-Fwd)***
  - LH_2/LOX (Aft-to-Fwd)
- No. of Engines
  - ***7***
- Engine Type/Cycle
  - ***Bell***
  - ***Modular***
- Propellant Combination
  - Bipropellant
    - ***LOX/Hydrogen***
    - ***LOX/Methane***
    - ***LOX/Propane***
  - Tripropellant
    - ***LOX/Hydrogen/Kerosene***
- TVC
  - ***Differential Throttling***
  - Gimbaling
- Pressurization
- Aerosurfaces
  - ***Body Flap***

Base Entry

Note:
Bold & Italics indicates path taken on ATSS TA-2 contract

J. B. McCurry
205-722-4509

TA-2 SSTO Definition & Assessment

VTHL

Lifting Body

- Propellant Tank Design
  - Nested Bulkhead
  - Common Bulkhead
  - ***Separate Bulkhead***
  - ***Cylinder***
  - ***Cone***
  - Toroidal
- Propellant Tank Layout
  - ***Fuel Tanks Outboard of LOX***
    - Cylindrical Tank
    - Conical Tank
    - Bent Conical Tank
    - ***Bent Biconic Tank***
    - Biconic Tank
  - LOX Tanks Outboard of Fuel Tank
  - LOX Tank Forward of Payload Bay
  - ***LOX Tank Aft of Payload Bay***
    - ***Cylindrical Tank***
    - ***Conical Tank***
- TPS
  - ***TABI/FRSI/FRCI/Carbon-Carbon***
- Structure
  - Structure Type
    - Monocoque
    - ***Skin/Stringer (Tanks)***
    - ***Honeycomb (unpressurized structure)***
  - Structure Materials
    - Aluminum Alloys
      - ***Al-Li***
    - Composites
      - ***Graphite/Epoxy***
- No. of Engines/Elements
  - ***5***
- Engine Type/Cycle
  - ***Bell***
  - ***Modular***
- Propellant Combination
  - Bipropellant
    - ***LOX-Hydrogen***
    - ***LOX-Methane***
    - ***LOX-Propane***
  - Tripropellant
    - ***LOX/Hydrogen/Kerosene***
- TVC
  - ***Differential Throttling***
  - ***Gimbaling***
- Pressurization
- Aerosurfaces
  - ***Body Flap***
  - ***Tip Fin/Rudder***
  - ***Elevon***

Wings

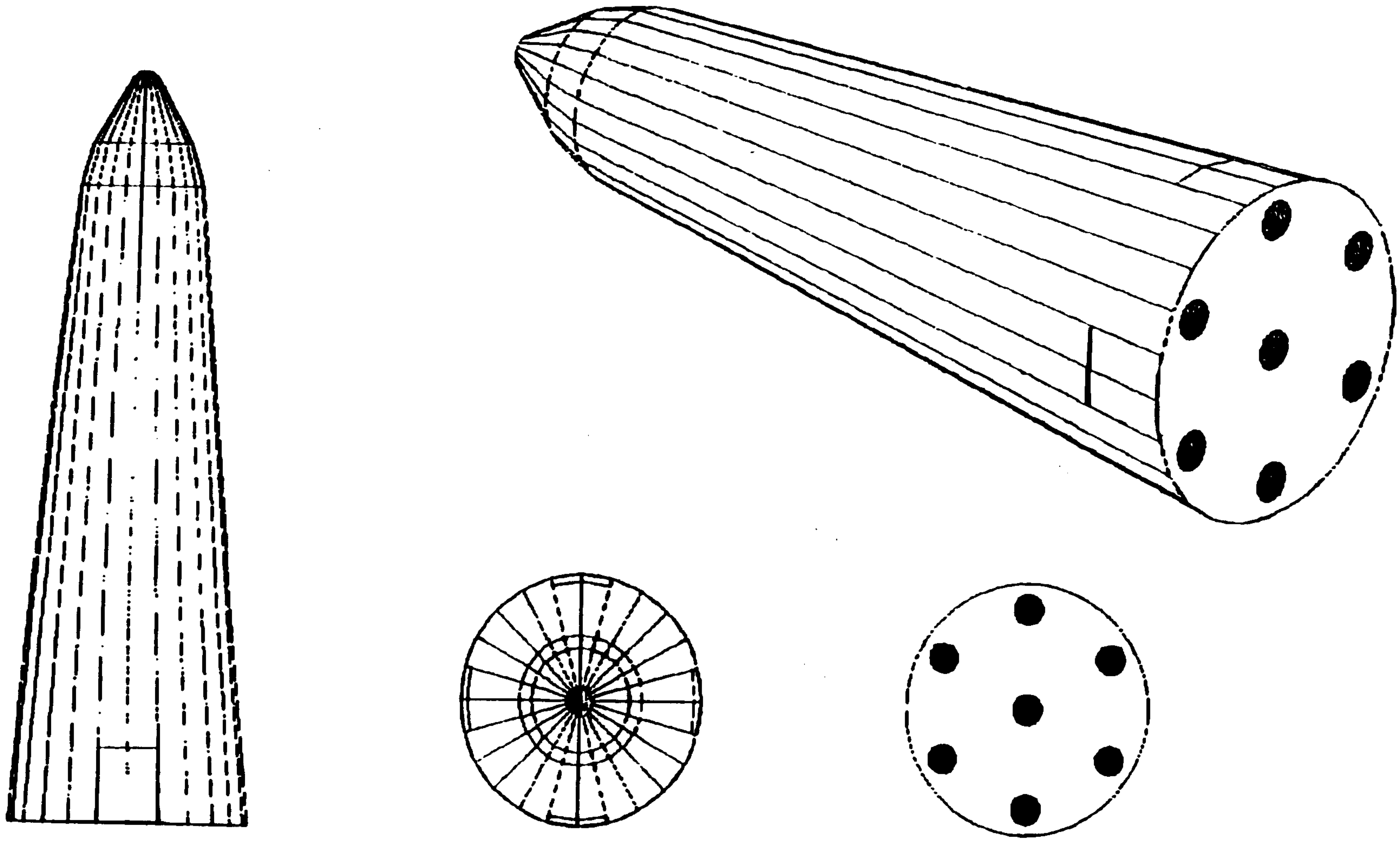
- **Option 3 Configuration**

Note:
Bold & italics indicates path taken on ATSS TA-2 contract

J. B. McCurry
205-722-4509

Configuration Overview

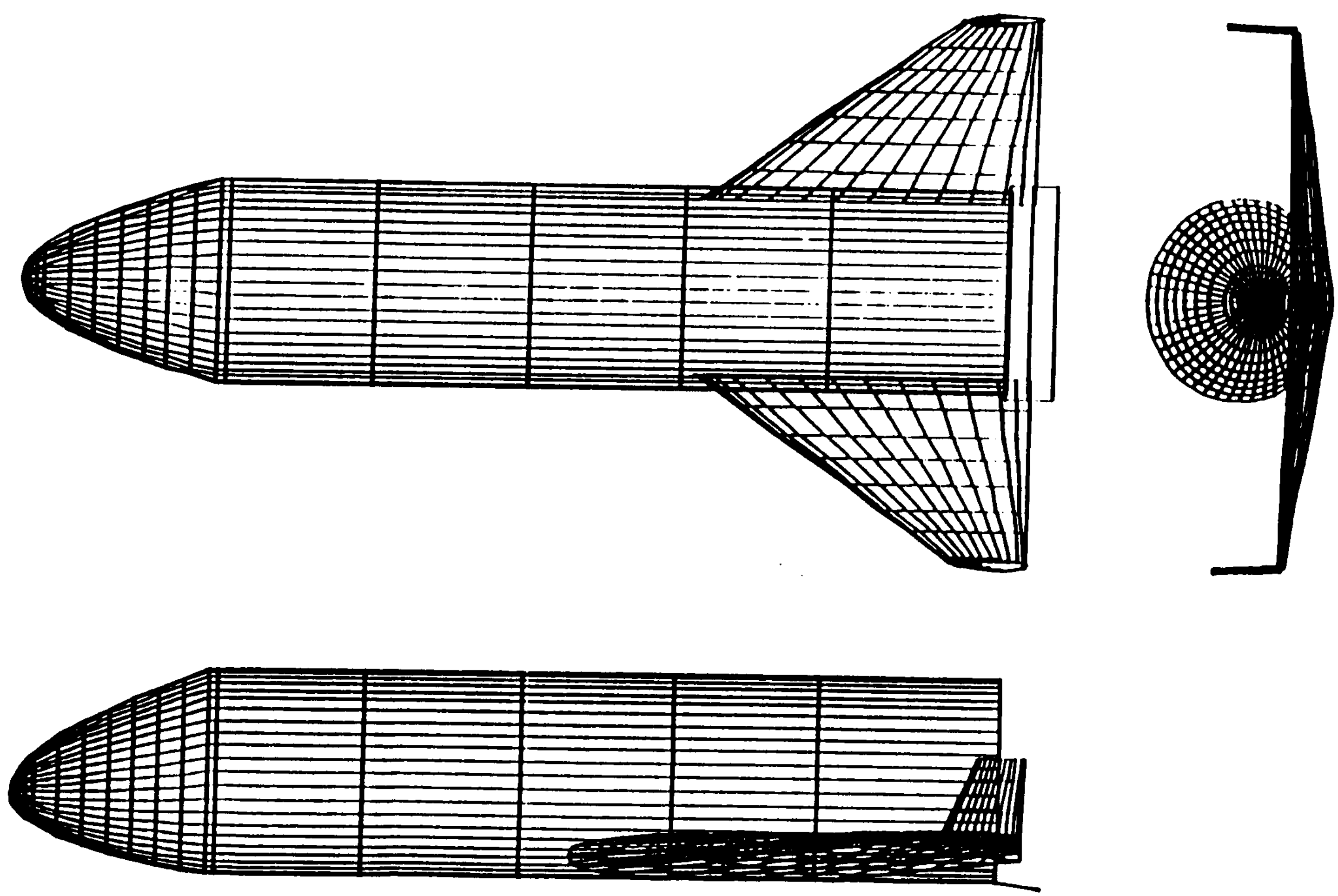
Vertical-Takeoff/Vertical-Landing Configuration



J. B. McCurry
205-722-4509

Configuration Overview

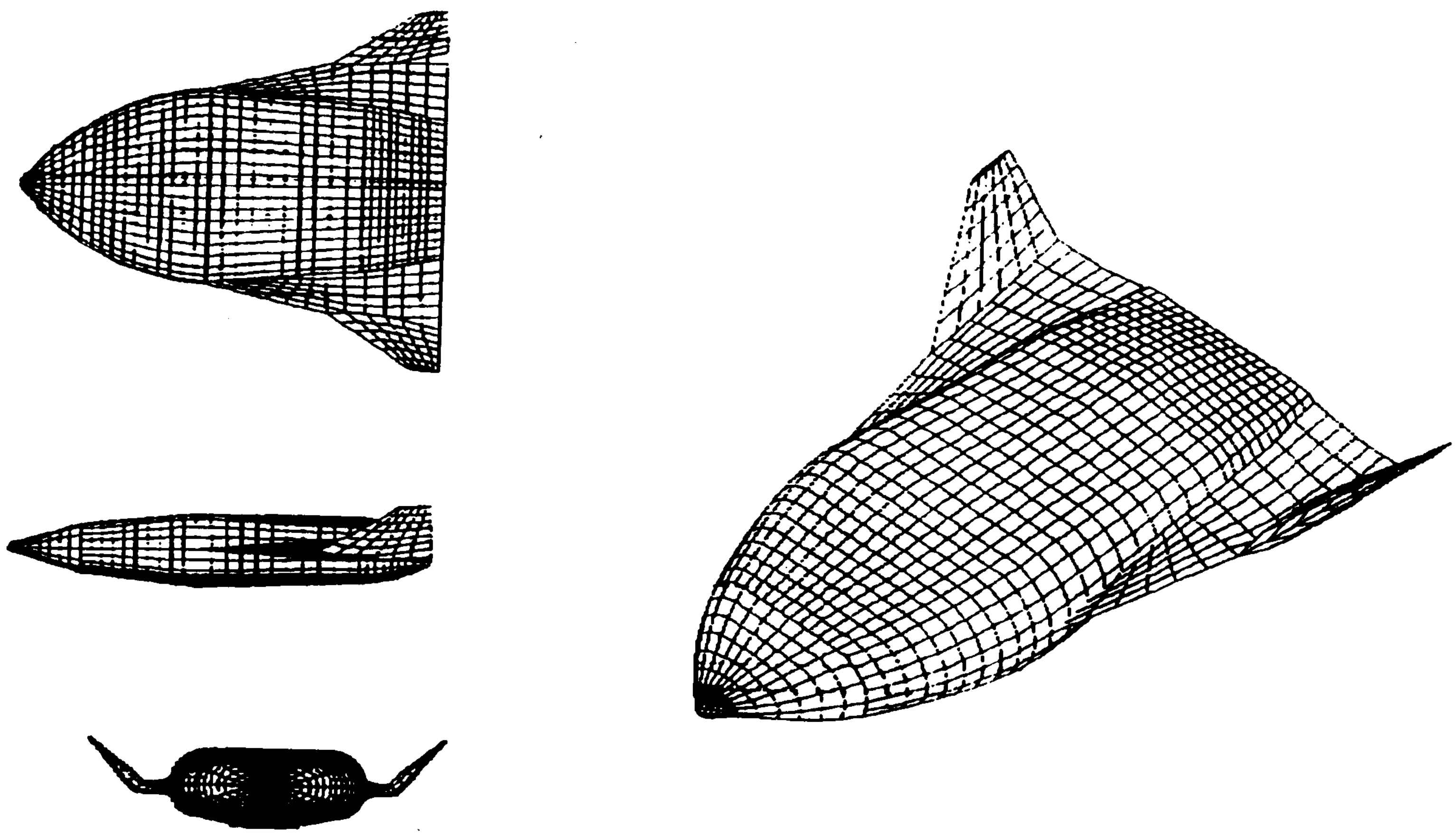
Wing-Body Vertical-Takeoff/Horizontal-Landing Configuration



J. B. McCurry
205-722-4509

Configuration Overview

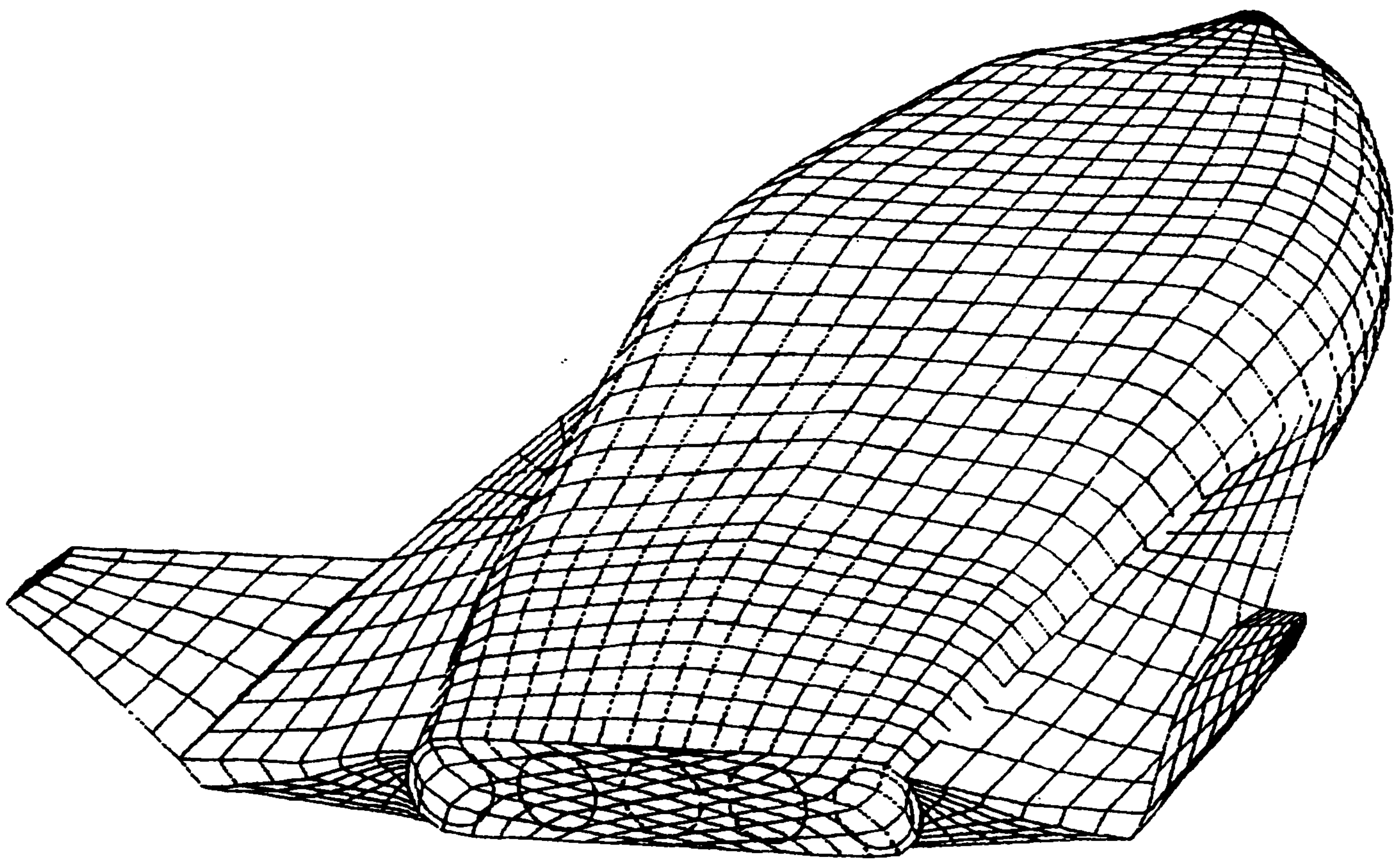
Lifting Body Vertical-Takeoff/Horizontal-Landing Configuration



J. B. McCurry
205-722-4509

Configuration Overview

Lifting Body Vertical-Takeoff/Horizontal-Landing Configuration



J. B. McCurry
205-722-4509

SSTO Design Groundrules

J. B. McCurry
205-722-4509

SSTO Design Groundrules

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

J. B. McCurry
205-722-4509

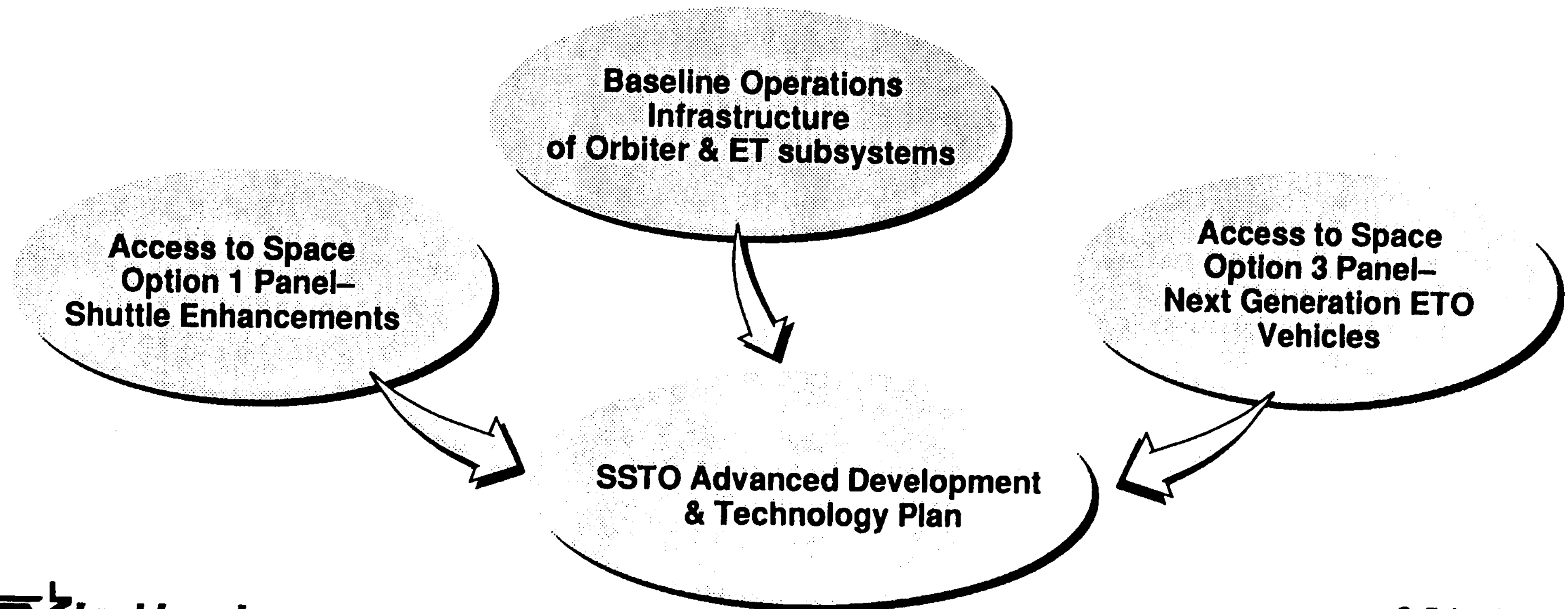
Operations Issues & Lessons Learned

- Lessons Learned
- Requirements Flow-Down
- Tripropellant Versus Bipropellant

G. F. Letchworth
407-383-2200 x2426

Operations Issues — Lessons Learned

- NASA has attempted only <u>one</u> 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/cost estimate)



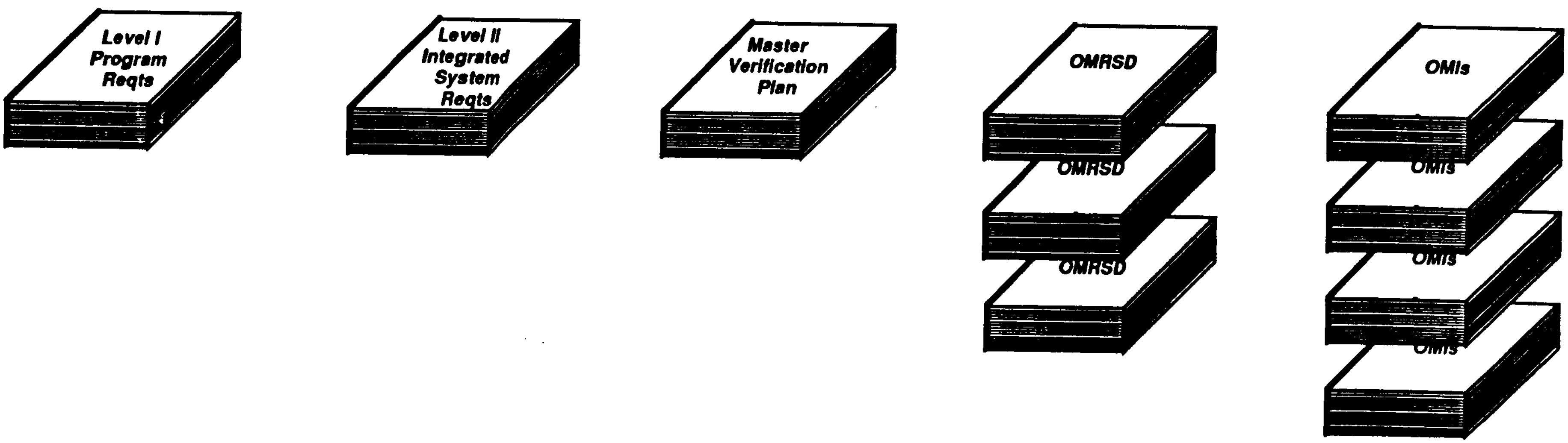
G. F. Letchworth
407-383-2200 x2426

Operations Issues — Requirements Flowdown

- **Ability to rapidly turnaround a reusable launch vehicle with a streamlined ground and mission operations infrastructure is strongly influenced by *program requirements* that dictate how to operate the vehicle**

Level I Program Requirements → **Level II Integrated System Requirements** → **Master Verification Plan** → **Operations & Maintenance Requirements & Specifications** → **Operations & Maintenance Instructions**

Way of Doing Business



G. F. Letchworth
407-383-2200 x2426

SSTO Operability Pros/Cons of Tripropellant vs. Bipropellant

- Lockheed, LSOC, and Aerojet brainstormed operability issues associated with tripropellant (LOX/Hydrogen/Hydrocarbon) main propulsion versus LOX/Hydrogen bipropellant
  - "Does tripropellant hurt/help recurring operations costs?"
- Generic main propulsion concepts were assumed (engine/cycle independent)
- Operational issues associated with cryogenic third propellant versus non-cryogenic were considered

G. F. Letchworth
407-383-2200 x2426

SSTO Operability Pros/Cons of Tripropellant vs. Bipropellant (Continued)

Major Assumptions

- Recurring operations costs are the dominant driver for SSTO viability
  - Outweigh relative differences in vehicle unit costs for different design concepts due to small fleet size (regardless of historical trend by U.S. Govt. to consider production cost a "sunk cost")
- All SSTO vehicle designs will meet the mission payload requirement
- For typical range of SSTO vehicle sizes within a class (VTVL or VTHL) the development of a "new" main engine will allow the same number of engines to be used, regardless of vehicle size (within a particular class)
  - Same number of engines for biprop. or triprop. solutions presuming rubber engines (thrust level & Pc sized as needed)
  - True for aerospike or bell nozzle concepts

G. F. Letchworth
407-383-2200 x2426

SSTO Operability Pros/Cons of Tripropellant vs. Bipropellant (Continued)

Assessment Question

- Does tripropellant concept help to reduce recurring operations costs as compared to a bipropellant concept?
  - If not, tripropellant engines should not be invested in; use technology development funds to obtain more reliable bipropellant engine

The following charts summarize the operability pros and cons of using tripropellant propulsion on an SSTO vehicle, as compared to using bipropellant propulsion.

A "pro" is a more desirable attribute associated with tripropellant propulsion (engines, feed subsystems, etc.) with respect to bipropellant options. A "con" is a less desirable tripropellant attribute, and a "neutral" is a tripropellant attribute that has no leveraging one way or the other.

G. F. Letchworth
407-383-2200 x2426

SSTO Operability Pros/Cons of Tripropellant vs. Bipropellant (Continued)

Pros	Cons	Neutral
• Less stand-off structure (for non-integral propellant tank designs) allows less structural maintenance • Less TPS allows less body TPS refurb. & repair • Use of noncryogenic third propellant facilitates prop. loading timeline vs. cryogenic third propellant • 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% • Increased parts count increases likelihood of unscheduled maintenance • Increased unscheduled maintenance increases logistics burden (spares) • "New" nature of triprop. propulsion increases likelihood of infant mortality failures in propulsion components • Increased complexity increases processing learning curve • Decreased vehicle size and increased parts count increases maintenance accessibility difficulty (if not considered in the design)	• Smaller vehicle but not a driver for SSTO class – Same number of engines to process as biprop. for new "rubber engine" – Processing not affected by vehicle size (up to a point)

G. F. Letchworth
407-383-2200 x2426

SSTO Operability Pros/Cons of Tripropellant vs. Bipropellant (Continued)

Pros	Cons	Neutral
	• Increased propulsion complexity will require more ground checkout and launch software • Increased ground checkout and launch software will increase sustaining software maintenance • 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 • No capability to verify 90% engine health onpad in both modes prior to liftoff • Higher flight performance reserve for 3 propellants • Use of cryogenic third prop. complicates prop. loading timeline • Fuel mode optimization complicates nominal/abort flight design	

G. F. Letchworth
407-383-2200 x2426

SSTO Operability Pros/Cons of Tripropellant vs. Bipropellant (Continued)

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

G. F. Letchworth
407-383-2200 x2426

SSTO Operability Pros/Cons of Tripropellant vs. Bipropellant (Concluded)

Conclusions

- Tripropellant propulsion inherently more complicated
- Tripropellant main propulsion has greater cost impact on recurring ground and mission operations than bipropellant propulsion
- Automated main propulsion feed subsystem test and check-out will help to mitigate hands-on processing but inherently higher parts count will result in higher unscheduled maintenance than bipropellant
- A significant reduction in number of required engines would be needed for tripropellant option to achieve parts count reduction over bipropellant option
- Recurring operations cost approximately 50% higher for tripropellant
- DDT&E and vehicle per unit cost also likely higher for tripropellant

Recommendation

- Apply limited DDT&E funds to pursue simpler, robust, operable bipropellant engine concept for SSTO

G. F. Letchworth
407-383-2200 x2426

VTOL/VTHL Pros & Cons

G. F. Letchworth
407-383-2200 x2426

Pros/Cons Assessment Process

- A concurrent engineering QFD session was held to identify qualitative strengths and weaknesses of generic vertical-takeoff/vertical-landing and vertical-takeoff/horizontal-landing SSTO configurations

- Technical disciplines of structures, loads, aerodynamics, stability, propulsion, flight mechanics, performance, sizing, operations, and cost analysis were represented

- Pros and cons were brainstormed and a common set of first order design, development, and operations categories were evolved between VTOL and VTHL; no relative ranking of the design categories made
  - Engine design & development
  - Structural efficiency
  - Vehicle processing and operations
  - Flight control risk
  - Landing opportunities
  - Landing system design

- Some non-common categories also resulted (payload integration, etc.)

G. F. Letchworth
407-383-2200 x2426

Pros/Cons Results Summary

	VTOL	VTHL
Pros	• More control authority (powered) over energy state during TAEM • Simpler load path and primary structure design • Larger static stability margin possible • Single ground/launch processing orientation	• No requirement for main engine conditioning & restart post-MECO • Less hazardous post-flight deservicing • Conventional flight mechanics & dynamics during all phases • Existing landing infrastructure
Cons	• More complicated main propulsion & feed subsystems • Higher risk entry/TAEM flight mechanics & dynamics • Higher risk post-flight deservicing • Vertical ground processing required	• Less volumetrically efficient outer moldline • Higher structural dry mass • Dual ground/launch processing orientations

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Vertical Landing

CATEGORY	PROS	CONS
Engine Design and Development	• Use of altitude compensating nozzle allows reentry on engine - Heat loads on engine nozzle higher when engine firing than during reentry • Plug nozzles can utilize engine for entry heat shield • Less yaw moment from engine-out • Easy to incorporate modular engine concept - Thrust cells • Easy to incorporate TVC by throttling engine sectors • Modular engine reduces development	• Requires deep (10:1) throttling engines • Engine design and development more complex - Deep throttling required - 2 position nozzle required - Restart required - 2 cycles/mission = 1/2 life • Requires restart conditioning • Base area large. Limits propulsion options • Requires engine restart for safe landing • Must have a roll control system (if use differential throttling) • No satisfactory existing plug nozzle engine - New engine development • Engine development test facilities more complex • Require on-board purge (engine) for restart (especially for an abort return)
Flight Control Risk		• Vehicle flight dynamic during landing • Slosh damping required for powered pitcharound in addition to ascent/trade of which momentum term sizes the baffles • Flight dynamics of powered pitcharound for landing is complex and risky, including plume blow-back issues • Unfamiliar control requirements • High gimbaling rate requirement • Larger gimbal angle requirement • Requires large body flaps for aerodynamic control • Failure modes associated with landing higher than horizontal landing • Array of intact abort options is more complicated to design autonomously than their benefit

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Vertical Landing (Continued)

CATEGORY	PROS	CONS
Flight Control Risk (continued)		• Vertical landing vehicles have inherent higher accident rates than horizontal landing vehicles
Landing Opportunities	• Propellant dissipation maneuver for abort (atmospheric) is hover mode • Return "anywhere" – More options • Minimum take-off landing facility • Small landing area • Does not require a runway for landing • More potential launch sites	• Landing dispersions • Low vehicle L/D translates to low cross range capability
Landing System Design		• Landing gear requires extra beef-up for drift protection as well as vertical loads • Requires robust landing gear
Miscellaneous	• Can probably evolve to a "Lunar" lander	• Landing acoustics
Payload Integration	• Larger C.G. envelope • Payload can be on top of vehicle – Easily accommodate variable length payload – Vehicle less sensitive to payload c.g. • Simpler orientation for payload integration & launch	

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Vertical Landing (Continued)

CATEGORY	PROS	CONS
Structural Efficiency	• High mass fraction structure • Should have best vehicle mass fraction – Smaller, lighter, cheaper vehicle • Body shape inherently stiff • Circular cross section tanks possible • Squat shape reduces vehicle height and loads • Shape should lend itself to "unitized" construction of major structural elements • Body shape has good volumetric efficiency • Simple structure • Simple load path • Lighter landing gear • Shape allows in-line propellant tank configuration • Less propellant tanks due to geometry • Less high temperature TPS area • Minimizes thermal protection surface • Allows for non-lifting body design which increases accessibility by not being as volumetrically limited • Simple aerodynamics (easy to predict) • Simple body shape for tooling and manufacturing	• Propellant for landing is payload • Must add either an extra subsystem or size extra propellant tanks, structure for landing ΔV • On-orbit storage of LO_2/LH_2 for 3-14 days (boiloff and propellant management) • Larger mission velocity requirement – More drag on ascent – Landing maneuver – Questions on how much hover capability required • Higher on-orbit and deorbit mass • Size propellant tanks to carry landing propellant which is payload hit (includes tanks, insulation, structures) • Fuel bias extra hit if can't handle fuel depletion cut • Base area larger, requiring more engineering to minimize base drag • RCS propellant required for landing maneuvers (weight penalty) • Perception problem, does not land on a runway • Added prop. sys. weight & complexity for safe abort capability during landing sequence • FPR sizing rqmts. for landing; hit to payload • Limited pilot visibility during final descent
Existing Test Vehicle	• Demonstrated approach "DC-X"	

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Vertical Landing (Concluded)

CATEGORY	PROS	CONS
Vehicle Processing and Operations	• Single orientation (vertical) for payload operations and integration	• Landing area blast debris • Vertical cargo integration • Vertical processing • Vertical checkout required • Requires vertical vehicle processing and payload integration facilities & attendant O&M • Range safety issue of landing with propellant • Post landing servicing of vehicle with propellant residuals • Vertical ground transportation of vehicle • Inefficient payload layout vs. vehicle tankage

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Horizontal Landing

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

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Horizontal Landing (Continued)

CATEGORY	PROS	CONS
Flight Control Risk	• Simpler flight software (fewer guidance modes) • Perception, lands on a runway • More robust landing method • Able to handle higher crosswinds during landing (terminal descent); body should have weathercock stability • Entry and terminal area energy management maneuvers are less dynamic and more predictable (no PPA, no slosh issues during entry) • Well understood landing process (Shuttle)	• More yaw moment from engine-out • Vehicle sensitive to cg • Use of bell engines makes vehicle cg more critical
Structural Efficiency	• Easier to fly a lifting trajectory • Moderate gimbal angle requirement • Moderate gimbal rate requirement • Reasonable volumetric efficiency possible • Lower inert weight than VTOL • No return propellant requirements • G_{loss} can be less if fly lifting ascent • Lower total mission velocity required • Possible to have simple load path • Structurally stiff • No inert weight penalty for wings if lifting body • Will not require ablative or actively cooled heat load • Lifting reentry reduces peak heat flux temperature	• Body shape less volumetrically efficient • Moderate gimbal rate requirement • Less efficient volume • More propellant tanks due to geometry • Possibly complex body shapes, increasing complexity of tooling, fabrication, production • Greater demand for TPS materials • More high temperature TPS area • Larger cross range is at expense of worse vehicle mass fraction • Load path vertical for ascent and horizontal for re-entry and landing • More restrictive payload bay, larger payload bay increases vehicle size and weight • Wings or body lift required for landing • Tankage not necessarily of circular cross section • Hard to achieve high mass fracture structure

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Horizontal Landing (Continued)

CATEGORY	PROS	CONS
Landing Opportunities	• Existing landing infrastructure • Can have large cross range • Improved landing opportunities	• Terminal landing speed higher (H-dot, Vx) may require heavier landing gear • Limited places to land or abort due to runway requirement • Requires large landing facility • Requires prepared landing surfaces
Landing System Design	• Consequences of landing/deceleration subsystem during terminal area energy management maneuvers/landing are more survivable (crew/payload) than VTOL	• High landing speed configurations require extensive landing gear tire development

G. F. Letchworth
407-383-2200 x2426

Vertical Take-off/Horizontal Landing (Concluded)

CATEGORY	PROS	CONS
Vehicle Processing and Operations	• Enables horizontal processing/check-out pre- and post-mission with better accessibility • Large experience base (aircraft, Shuttle) • Easy transport to processing facility • Can be towed from place to place on its landing gear by aircraft tow cart • Horizontal P/L integration, engine; • Horizontal crew module ground processing • Possible use of Shuttle OPF • Payload volume • Easy to integrate • Rollover ground transport • All horizontal payload integration (if baselined) • Option of vertical or horizontal checkout • No post-flight hazardous propellant to deservice	• Must have residual propellant disposal prior to landing • Requires horizontal to vertical repositioning • GSE • Rotation to vertical • Horizontal ground processing and vertical launch processing

G. F. Letchworth
407-383-2200 x2426

Design Results

- Major Design Considerations
- Outer Moldline Trends
- Alternative Internal Layouts
- Vehicle Sizing Process
- Technology Assumptions
- Sizing Groundrules
- Sizing Tool Description
- VTOL Concept Results
- VTHL Winged Body Concept Results
- VTHL Lifting Body Concept Results

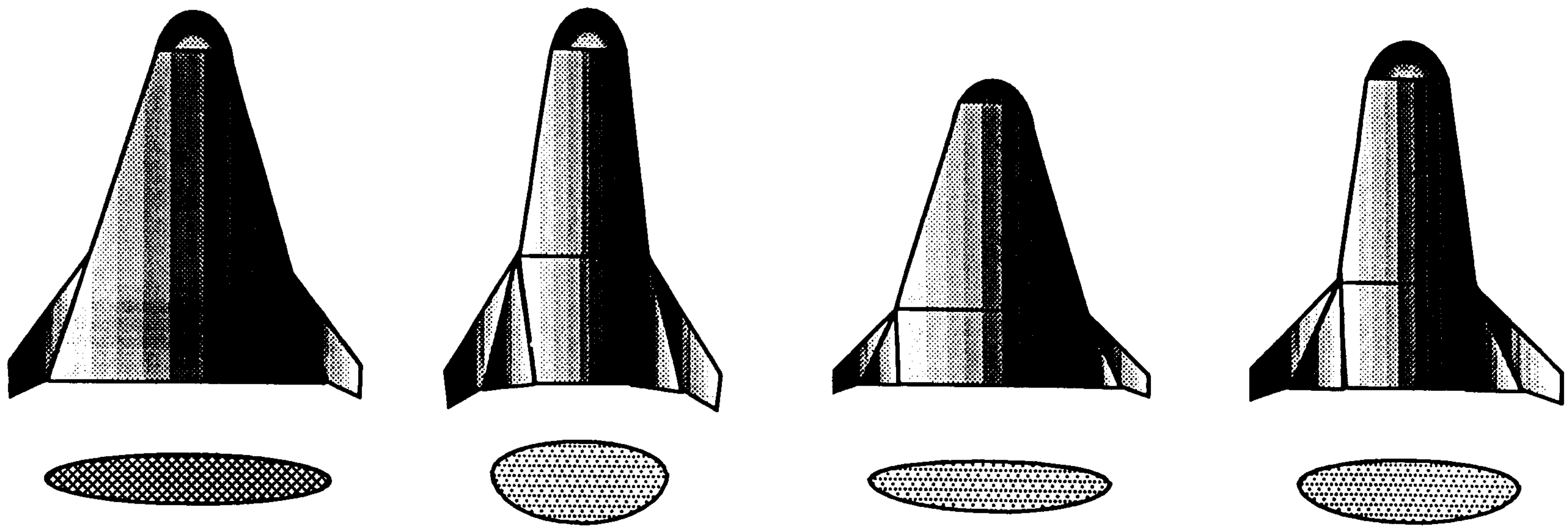
Keith Holden
205-722-4531

Major Design Considerations

- Outer Moldline
  - Entry heating (heat rate, heat load)
  - Aero (stability, drag, L/D)
- Major Structural Element Layout (tanks, thrust structure, payload, crew, subsystem, etc.)
  - Vehicle controllability in all regimes (c.g. versus c.p. location)
  - Load path during ascent and entry
  - Volumetric efficiency
- Propellant Combination
  - Bipropellant
  - Tripropellant
- Main Propulsion Choice

Keith Holden
205-722-4531

VTHL
Outer Moldline Design Trend

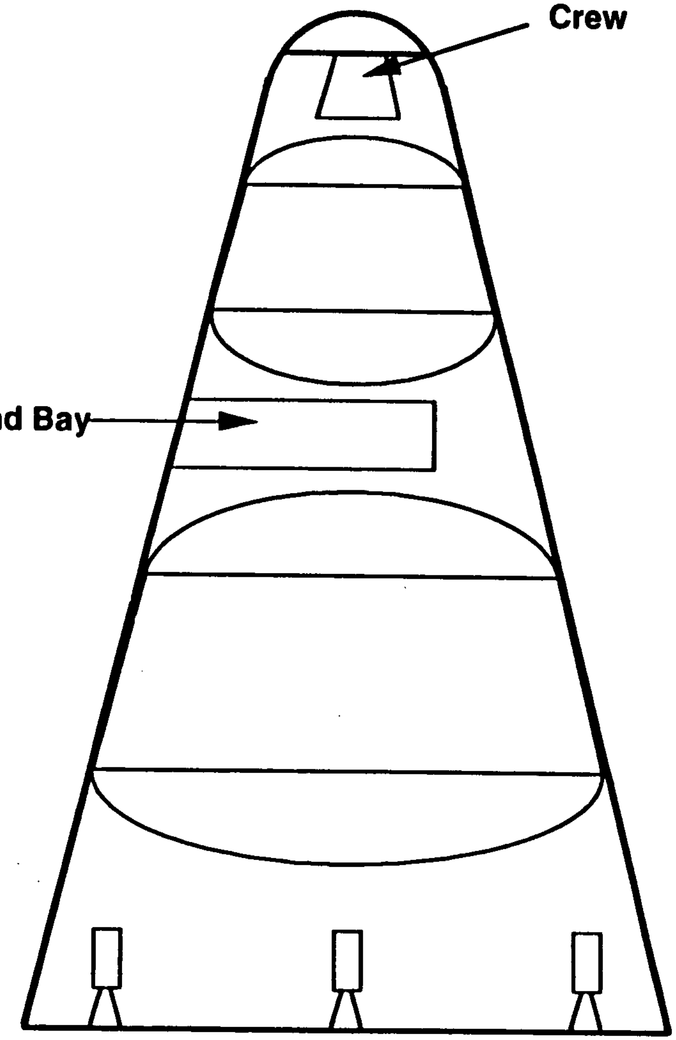


Dry Mass Class	~300 Klbm	~200 Klbm	~130 Klbm	~160 Klbm
Design Metric	• Maximum Mass • Maximum Aero	• Maximum Mass • Minimum Aero	• Minimum Mass • Moderate Aero	• Moderate Mass • Moderate Aero

Keith Holden
205-722-4531

Alternative Internal Layouts

Conical VTOL Configuration

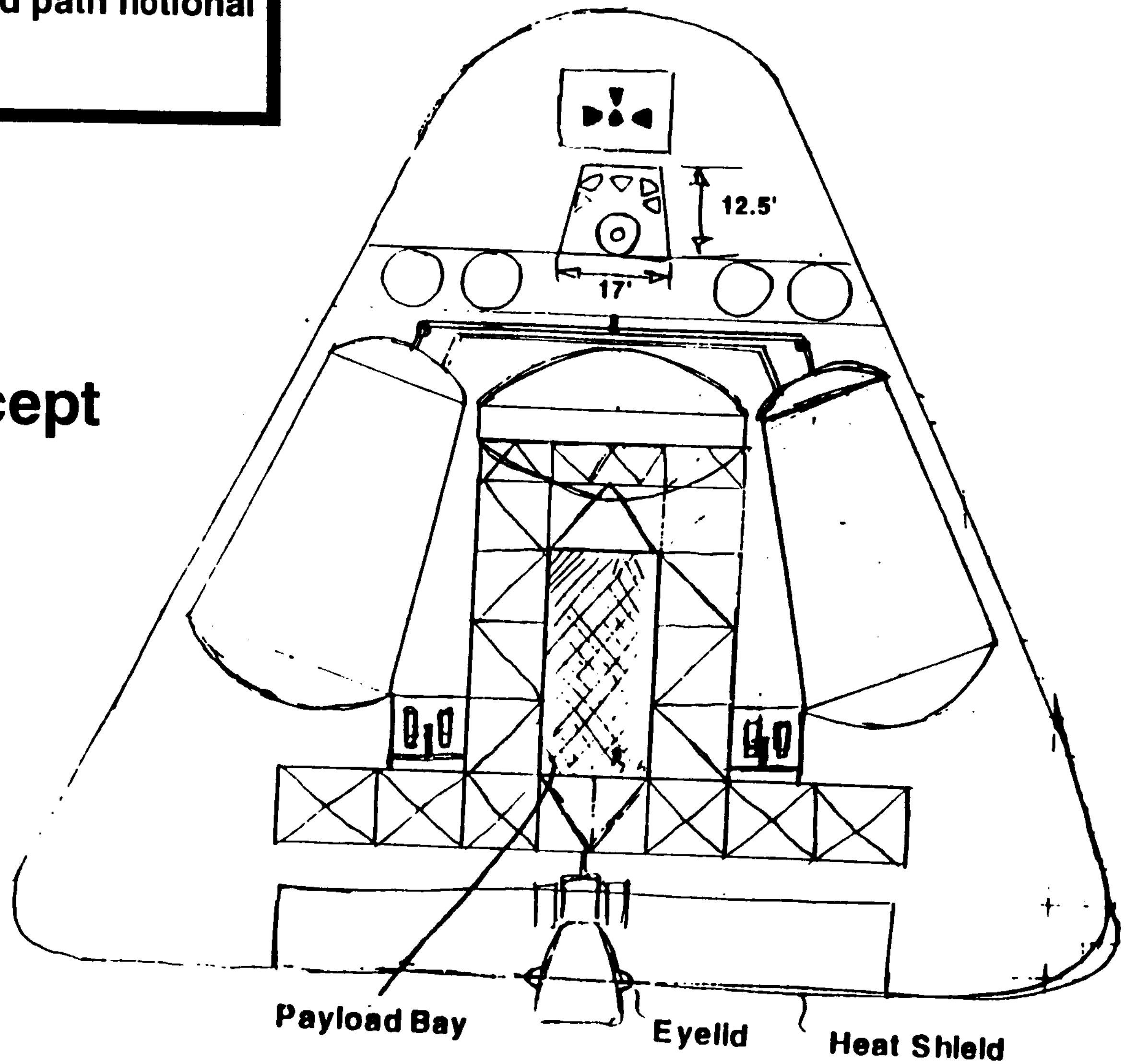


Keith Holden
205-722-4531

Alternative Internal Layouts

- — Body aspect ratio not to scale
- — Tank layout and load path notional
- — Tip fins not shown
- — CG forward of CP

Notional Concept

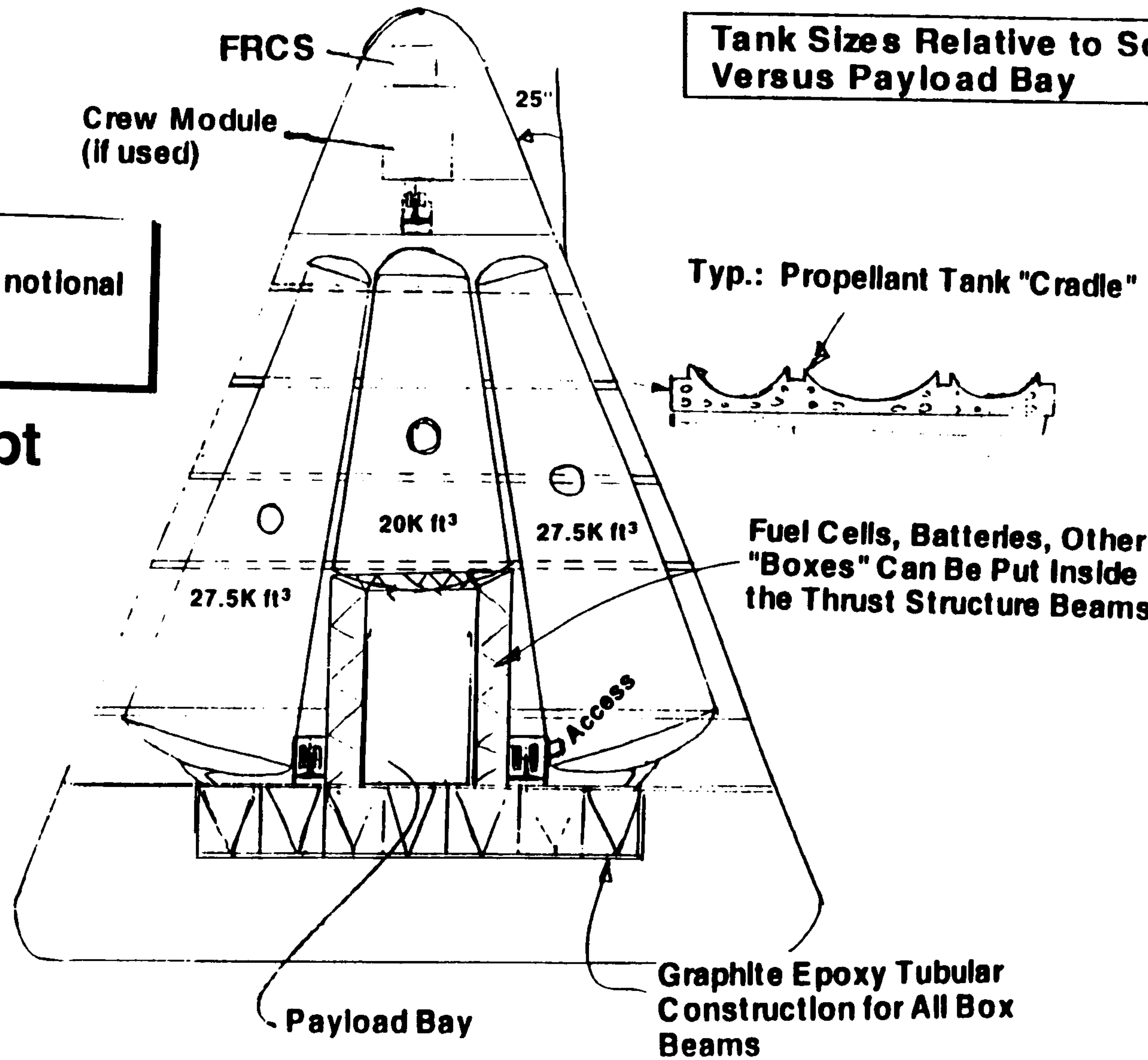


Keith Holden
205-722-4531

Alternative Internal Layouts

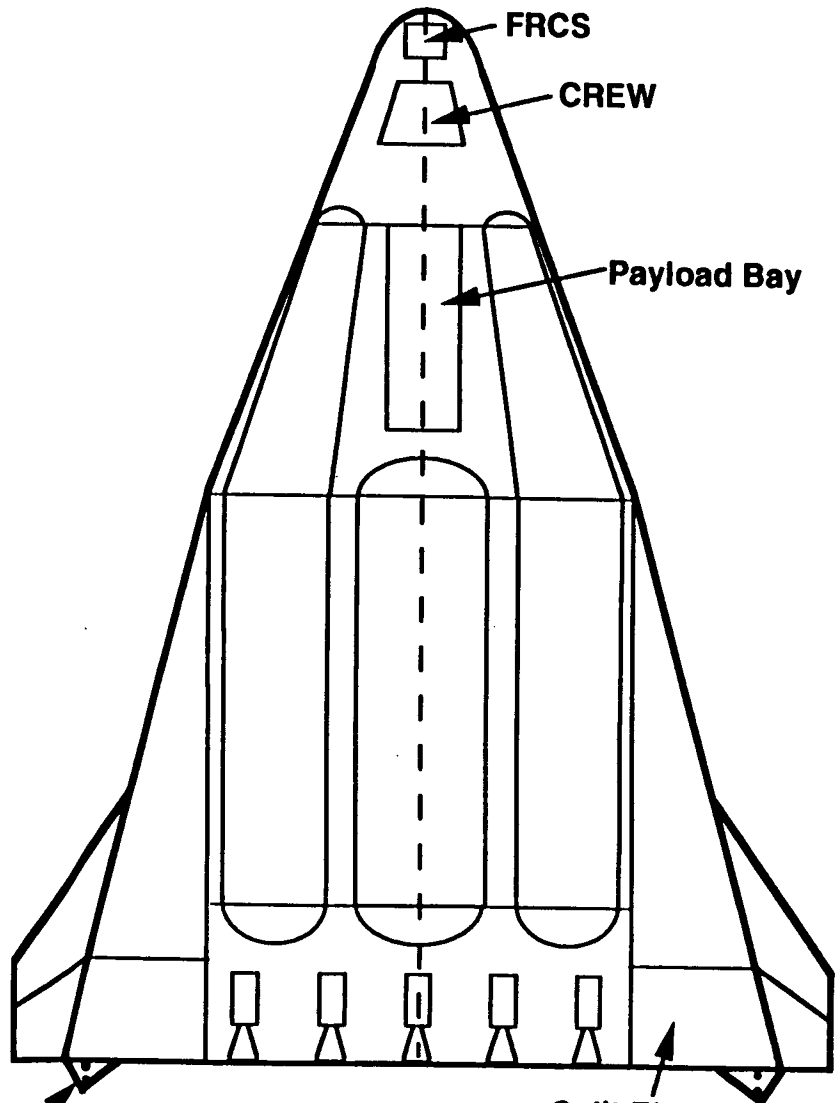
- Aspect ratio not to scale
- Tank layout and load path notional
- Tip fins not shown
- CG forward of CP

Notional Concept



Keith Holden
205-722-4531

Alternative Internal Layouts
Lifting Body VTHL Configuration
Top View



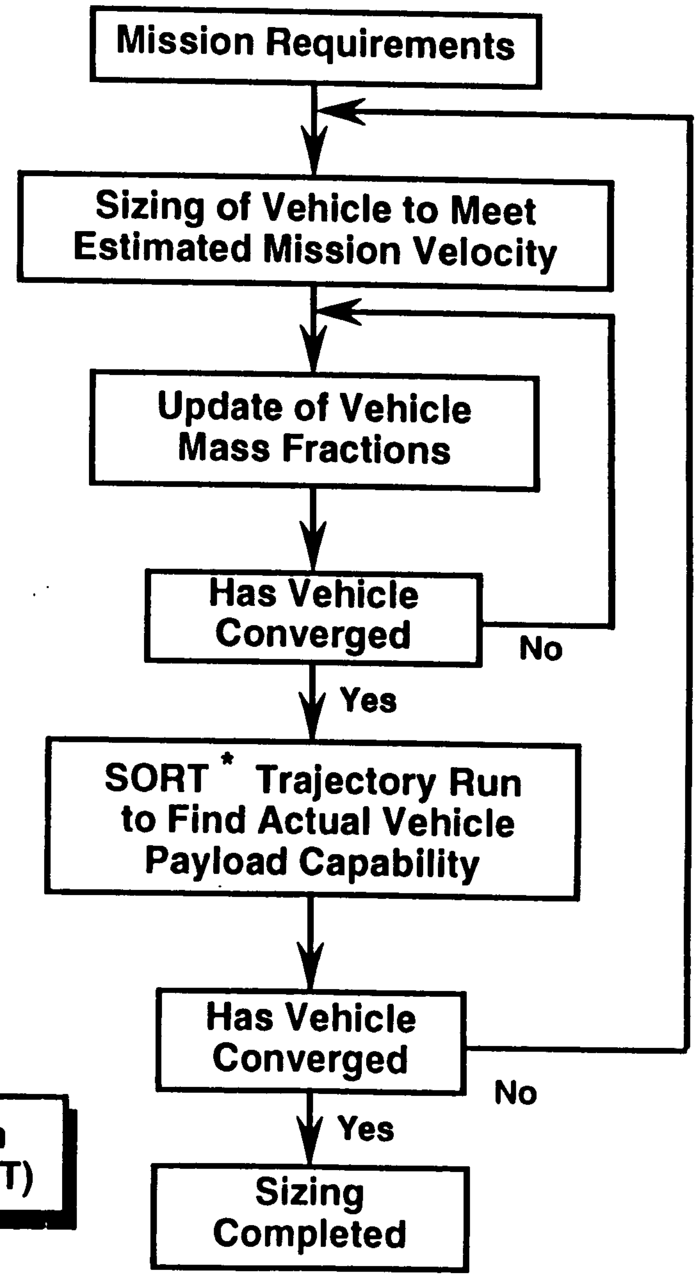
Keith Holden
205-722-4531

Vehicle Sizing Process

- Mission requirements are input into the vehicle sizing model
  - Vehicle payload
  - Orbit inclination, perigee, apogee
  - Ascent trajectory constraints (Q-bar, acceleration, Q-alpha)
- Exoatmospheric (second) burn delta velocity capability varied to find the endoatmospheric (first) and exoatmospheric burn propellant loads that result in the total vehicle minimum structural weight
- Simulation and Optimization of Rocket Trajectories (SORT) trajectory program used to calculate total vehicle payload capability subject to meeting ascent trajectory constraints
- If desired vehicle payload capability has not converged, the sizing model estimated mission velocity requirement was updated

Keith Holden
205-722-4531

Vehicle Sizing Process (Concluded)



* Simulation and Optimization of Rocket Trajectories (SORT)

Lockheed

Keith Holden
205-722-4531

Technology Assumptions

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
1 O2/H2
2 O2/H2/Propane
3 O2/H2/RP-1

Configuration Key
1 Side Entry Cone VTOL
2 Wing/Body VTHL
3 Lifting Body VTHL

Keith Holden
205-722-4531

Technology Assumptions (continued)

- 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
- *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

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

Keith Holden
205-722-4531

Technology Assumptions (Continued)

- *• 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**

Keith Holden
205-722-4531

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*

Keith Holden
205-722-4531

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%
  – Includes volume for residuals (4.25% used by Option 3; accounts for start-up)

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

Keith Holden
205-722-4531

Sizing Groundrules (Continued)

- *• 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

- *• Vehicle materials
  - Graphite epoxy LH2 propellant tank
  - Graphite epoxy unpressurized structures
  - Aluminum-lithium kerosene and LO2 tanks

- *• 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**

Keith Holden
205-722-4531

Sizing Groundrules (Continued)

* • 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 then pressurized to just over one atmosphere for entry
  - Main propellant flight performance reserves and residuals are vented

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

Keith Holden
205-722-4531

Sizing Groundrules (Concluded)

- 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

Keith Holden
205-722-4531

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

Keith Holden
205-722-4531

Sizing Tool Description (Continued)

Parentage of the Sizing Model (continued)

- 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

Keith Holden
205-722-4531

Sizing Tool Description (Continued)

- The side entry cone VTOL, the winged VTHL, and the lifting body VTHL sizing tools were developed from a generic single stage to orbit (SSTO) sizing tool
  - These launch vehicle configurations were too different to be covered by a general purpose detailed sizing tool
- Configuration specific aspects on the sizing tools will be discussed in each configuration section
- All configurations have a performance and a weights spread sheet
- An iteration loop keeps the information flowing between the performance and weights spreadsheets until a vehicle configuration is converged upon

Keith Holden
205-722-4531

Common Sizing Tool Features

Performance Spread Sheet

Input Data
- Mission
- Engine Definition
- Subsystem definition
- Etc.

Mission Performance
- Burn Velocity Split
- Burn Propellant Load
- Engine Thrust Requirements
- Etc.

Mission Velocity Requirements

Weights Spread Sheet

Propellant Tank Geometry
- Oxidizer
- Fuel 1
- Fuel 2

Vehicle Geometry

Subsystem Mass
- Engines
- Propellant tanks
- Thrust structure
- Etc.

Keith Holden
205-722-4531

Sizing Tool Description (Continued)

Performance Spread Sheet Sections

- 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 weights supplied by the weights spread sheet 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 one burn and the mode two burn initial thrust-to-weight ratios

Weights Spread Sheet Sections

- The required vehicle propellant load information is used to calculate the propellant tank geometry
- The vehicle geometry is calculated from the propellant tank geometry
- The final section calculates the vehicle subsystem weights

Keith Holden
205-722-4531

Sizing Tool Description (Continued)

Factors Used in Calculating the Propellant Tank Masses

- Tank pressures are calculated for the forward and aft endcaps and the barrel section as a function of the tank geometry, the tank ullage pressure and the vehicle liftoff thrust-to-weight ratio
- The tank section thickness is a function of the tank geometry, the section pressure, the material allowable stress and the safety factor
- The mass of the tank as a pressure vessel is a function of the tank geometry, the thickness of the sections and the material density
- The non optimum tank mass is a function of the tank geometry and the material density
- The total tank mass is the sum of the mass as a pressure vessel and the non optimum tank mass

Keith Holden
205-722-4531

Sizing Tool Description (Continued)

Factors Used in Calculating Other Vehicle Subsystem Masses

- The crew cabin mass is a function of the number of crew
- The body flap mass is a function of the body flap planform area
- The TPS mass is a function of the body wetted area
- The body insulation mass is a function of the body wetted area (used only if TPS does not provide adequate insulation to the underlying body)
- The cryogenic propellant tank insulation mass is a function of the tank surface area
- The landing gear and auxiliary systems mass is a function of the vehicle landed mass
- The main engine mass is a function of the number of engines or a function of the required thrust

Keith Holden
205-722-4531

Sizing Tool Description (Continued)

Factors Used in Calculating Other Vehicle Subsystem Masses (Continued)

- The main propellant system feedline and pressurization mass is a function of the propellant mass flow rate and propellant density
- The gimbal actuator mass is a function of the vacuum thrust
- The thrust structure mass is a function of the vacuum thrust and the number of engines
- The RCS mass is a function of the vehicle length and entry mass
- The OMS mass is a function of the initial vehicle on-orbit thrust-to-weight ratio and the mass of the OMS and RCS propellants
- The prime power system mass is a function of the total control surface area, the vacuum thrust, the avionics mass, the average on-orbit power requirements and the number of days spent on-orbit
- The power conversion and distribution system mass is a function of the total control surface area, the vacuum thrust and the vehicle landed mass

Keith Holden
205-722-4531

Sizing Tool Description (Concluded)

Factors Used in Calculating Other Vehicle Subsystem Masses (Concluded)

- The surface control actuator mass is a function of the surface control area
- The avionics system mass is a function of the vehicle dry mass
- The environmental control and life support system mass is a function of the crew cabin volume, the number of crew, the time spent on-orbit, the avionics system mass and the average on-orbit heat rejection power requirement
- The personal provisions mass is a function of the number of crew
- The personnel mass is a function of the number of crew
- The main propellant residual mass is a function of the propellant tank volume, the vacuum thrust and the propellant used

Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results

Configuration-Specific Technology Assumptions

- Main engines used for ascent and landing phases
  - Dependable engine ignition for landing maneuver
  - Sufficient throttle range at touchdown
  - The large number of thrust chamber assemblies in the modular engine configurations (dual throat and plug nozzle) make it easier to reach the low thrust levels required for the landing maneuver
  - The low area ratio in a plug nozzle thrust chamber assembly makes this engine configuration less sensitive to deep throttling at sea level
  - The engine restart propellant and the vehicle landing propellant is stored for the mission time (7 days) without appreciable boiloff

Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

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)
- 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 (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

Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

Configuration Specific Sizing Groundrules (continued)

- 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

Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

Configuration Specific Subsystem Mass Relationships

- The mass of a piece of unpressurized structure is a function of the mass of everything above it or of the aero loads on it during entry and the vehicle factor of safety
- The engine restart propellant mass is a function of the propellant used and the maximum vacuum thrust of the engines cooled down for restart during the landing maneuver
- The landing propellant mass is a function of the vehicle terminal velocity and the hover time selected

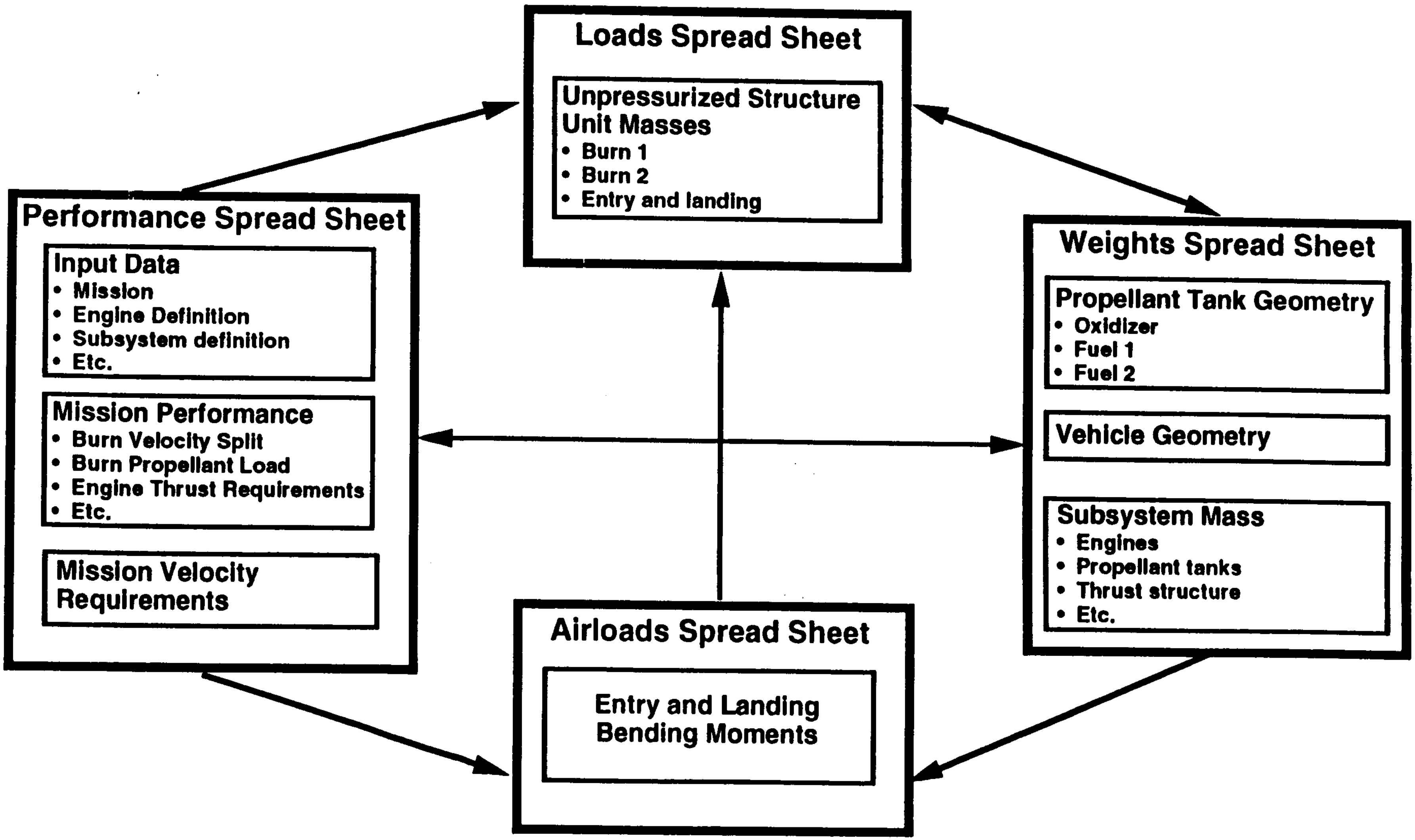
Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

- The side entry VTOL configuration requires some type of pullup maneuver to transition from horizontal gliding flight during entry to vertical for landing
  - Work is underway on designing this maneuver
  - The approximation currently used in the sizing program is that enough propellant is carried to decelerate from the vehicle terminal velocity and then to hover for a user specified amount of time
- The engine restart propellant and the vehicle landing propellant are stored in the OMS propellant tanks
- Propellant tanks are integral (load carrying)
- LOX and hydrocarbon tanks are nested for tripropellant configuration options
- Additional propellant tank layout choices in the weight spreadsheet are:
  - The fore-to-aft arrangement of the propellant tanks are user defined
  - The payload bay may be in the forward or aft intertank
  - The propellant tanks may have separate, nested or common endcaps
  - The propellant tanks may be cylindrical or toroidal

Keith Holden
205-722-4531

Conical Side Entry VTOL Configuration Model



Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

- The weights spreadsheet splits the wetted body area into zones to calculate the TPS mass
- This configuration also has a loads spreadsheet and an airloads spreadsheet
- The airloads spreadsheet calculates the bending moments imposed on the vehicle during entry and sends the information to the loads spreadsheet
  - User consults a vehicle entry trajectory analysis and picks the most heavily loaded part of the entry trajectory
  - At this point, the user inputs Cn and Ca data as a function of vehicle station into the airloads spreadsheet
  - Geometry and mass information are supplied by the weights spreadsheet

Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

- The loads spreadsheet calculates unit axial loads for the unpressurized structures and sends it to the weights spreadsheet
  - Axial loads are calculated on the unpressurized structure based on the mass of everything above it
  - The load conditions that are looked at are during burn one at either the end of the burn or at the point the vehicle reaches maximum acceleration; during burn two at either the end of the burn or at the point the vehicle reaches maximum acceleration and the vehicle setting on the pad (for the aft skirt)
  - The unpressurized structure unit masses are calculated for these load cases (including the bending moments from the airloads spreadsheet)
  - The maximum value of the unpressurized structure unit masses is sent to the weights spreadsheet

Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

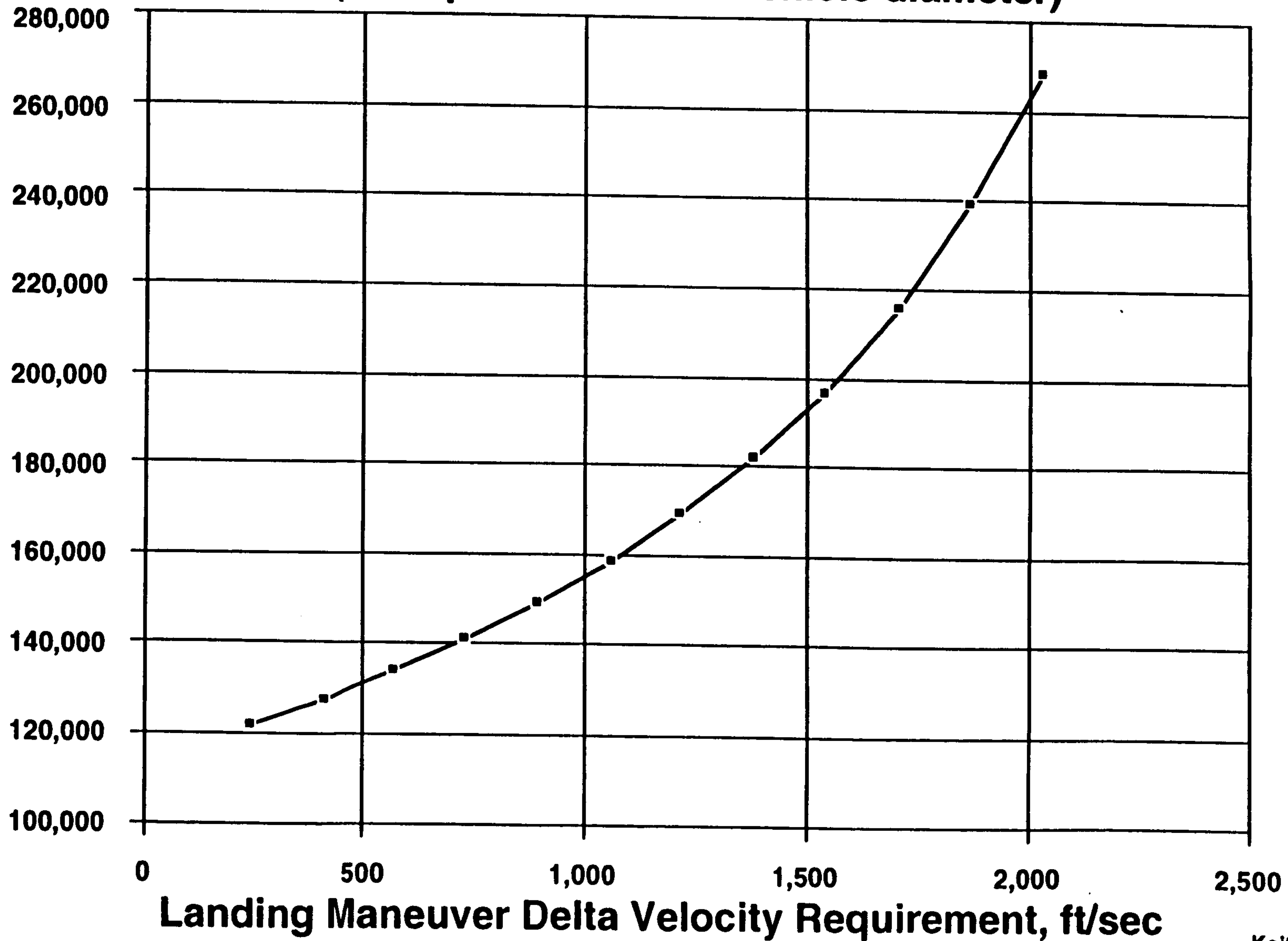
- The following charts show the results on trades on the vehicle dry mass as a function of the landing maneuver velocity requirement, the vehicle base diameter and the vehicle cone half angle
  - A landing maneuver velocity requirement greater than 1,000 ft/sec is a major vehicle dry mass driver

  - The vehicle dry mass is not sensitive to small changes in vehicle diameter or cone half angle

Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

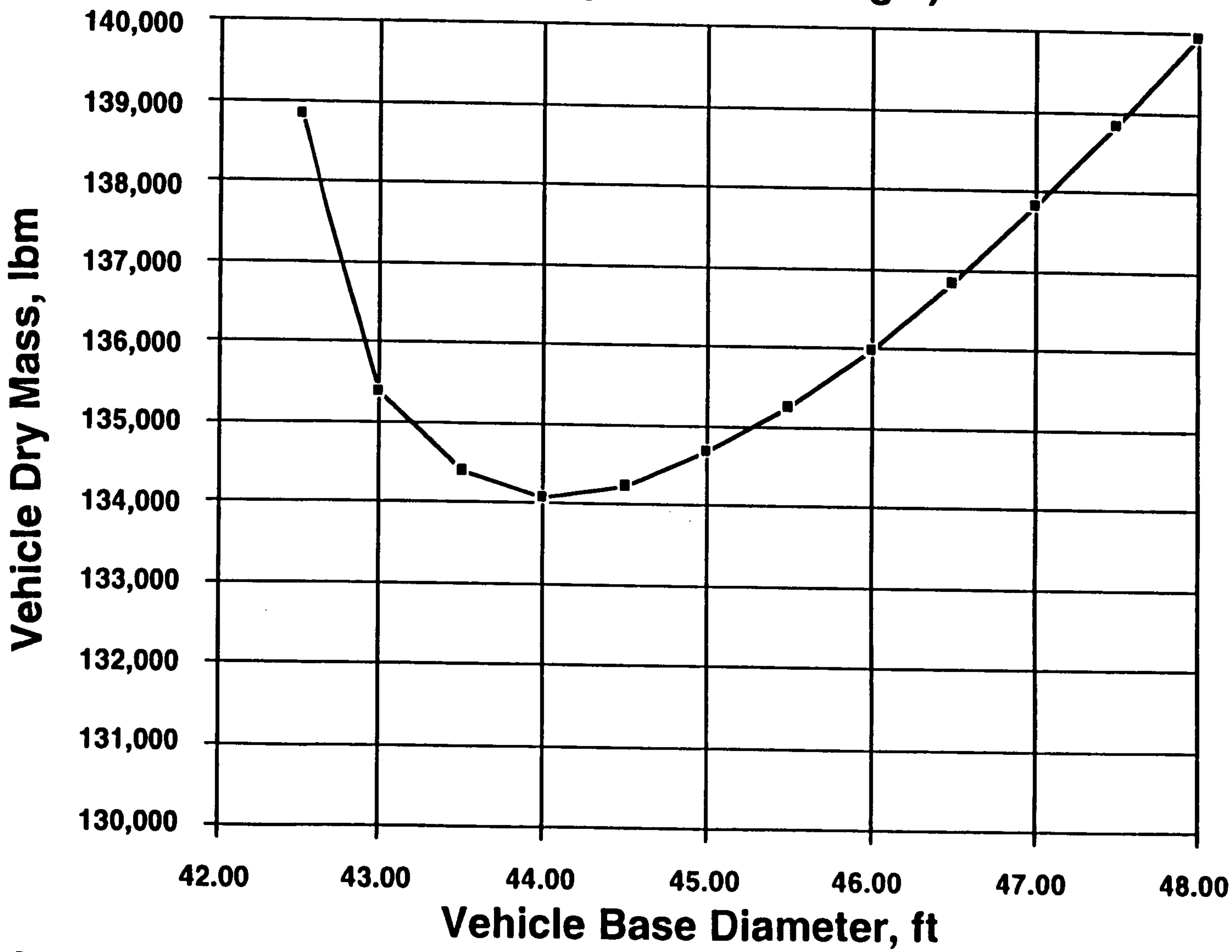
Vehicle Dry Mass as a Function of Landing Maneuver Delta Velocity
(each point is at best vehicle diameter)



Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Continued)

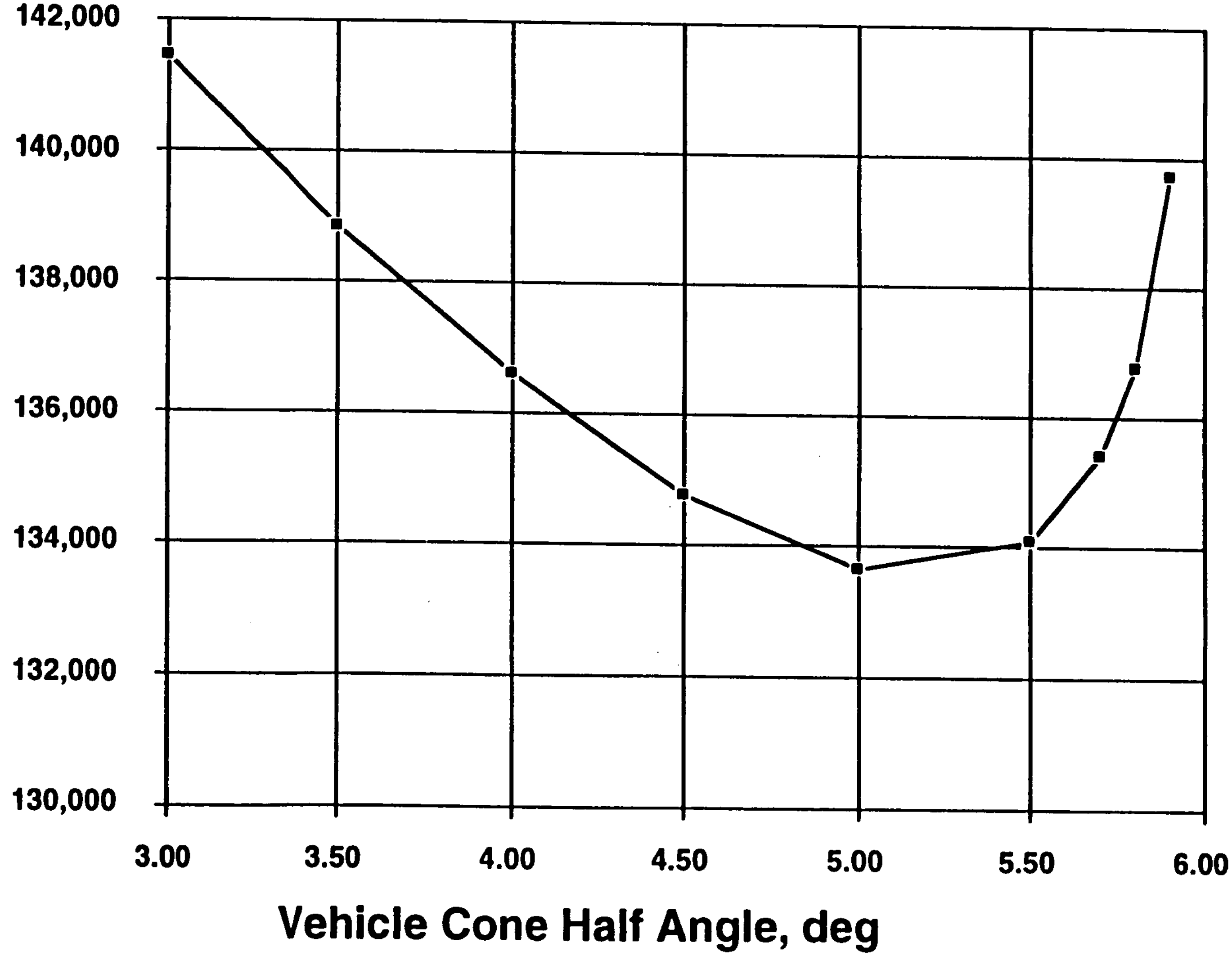
Vehicle Dry Mass as a Function of Vehicle Base Diameter
(for 5.5 deg. cone half angle)



Keith Holden
205-722-4531

VTOL Side Entry Cone Concept Results
(Concluded)

Vehicle Dry Mass as a Function of Vehicle Cone Half Angle



Keith Holden
205-722-4531

VTOL Sizing Conclusions

- The following chart is a bar chart showing the dry mass of the side entry conical VTOL vehicle configurations as a function of the engine and propellant used on the configuration and cut sheets describing each configuration in more detail

- The best propellant is a function of the engine used

- There was a large difference in the dry mass of the two vehicle configurations with different versions of the Russian tripropellant engine (the Option 3 version of the RD-701 engine had less hydrogen flow during the mode one burn than the Pratt version of the RD-704 engine)

- The vehicle configuration with the full flow staged combustion cycle (FFSCC) engine was significantly lighter than the vehicle configuration with the Option 3 version of the evolved SSME

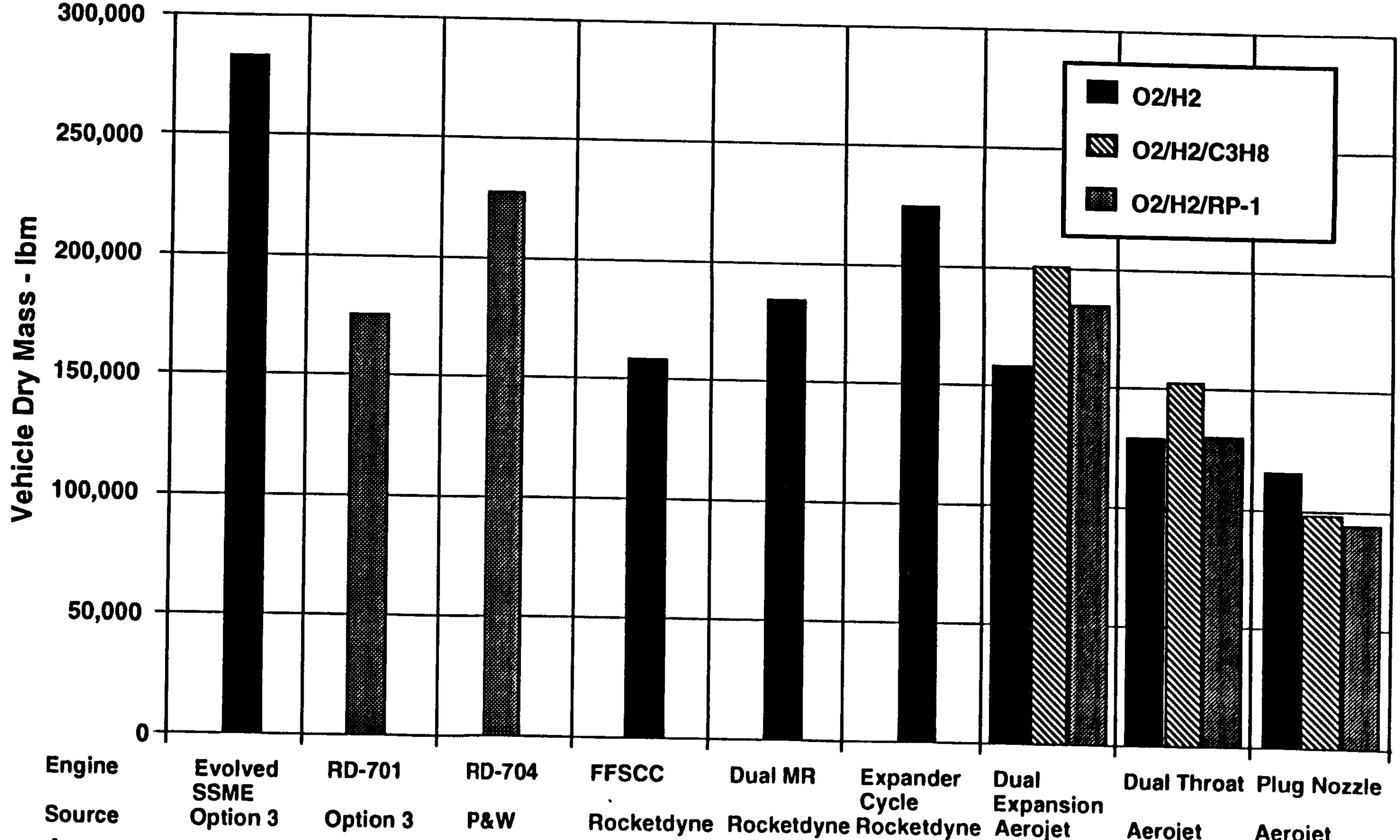
- The vehicle configurations using the dual mixture ratio (MR) engine and the expander cycle engine are heavier than the vehicle configuration using the FFSCC engine

- The lowest dry mass vehicle configuration used a plug nozzle engine

- A vehicle configuration using a tripropellant plug nozzle engine was slightly lighter than a vehicle configuration using an O_2/H_2 plug nozzle engine

Keith Holden
205-722-4531

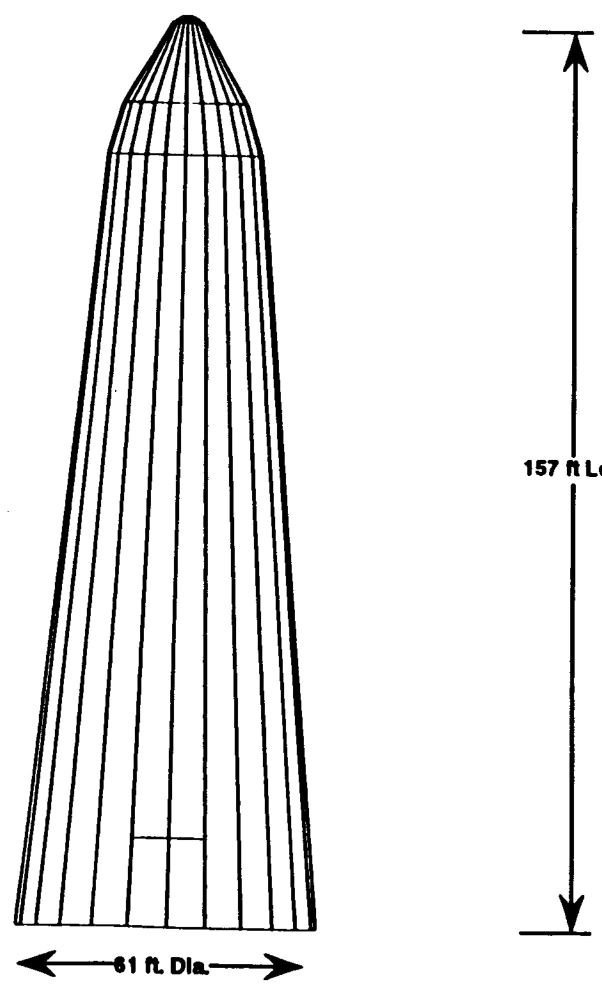
Conical VTOL Vehicle Dry Mass as a Function of Engine and Propellant Used



Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



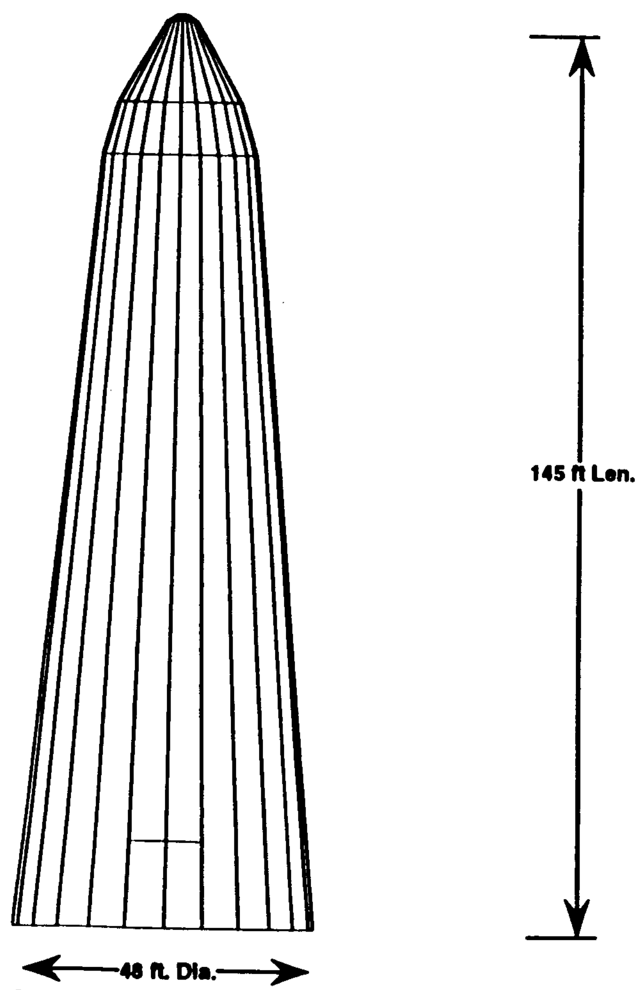
Vehicle Not Drawn to Scale

Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	3,273,748 lbm
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

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

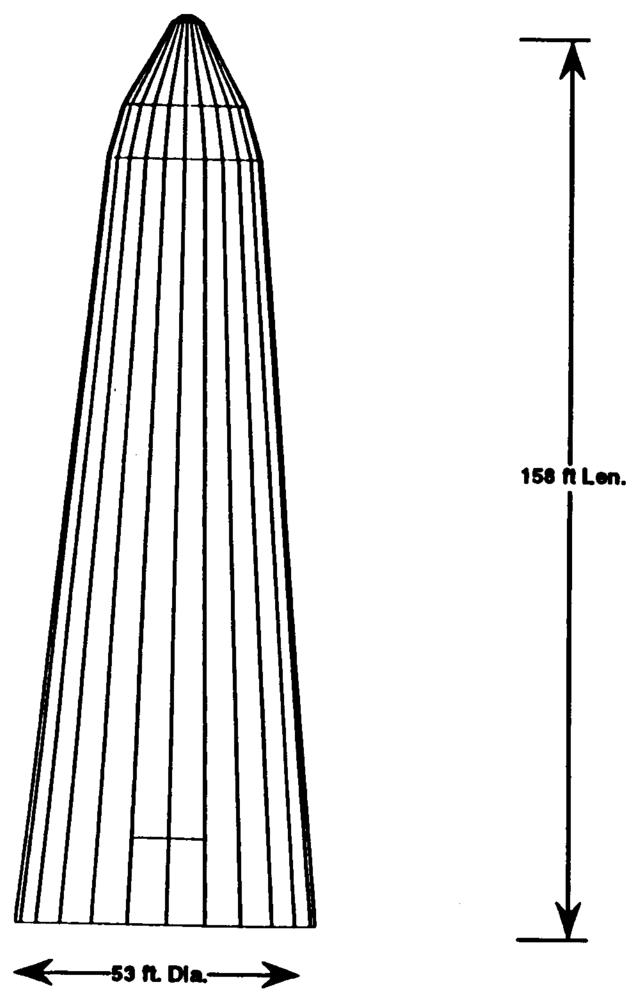
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,409,834 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	206,598 lbf
Vacuum Isp (@100 % RPL):	452.7 sec

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

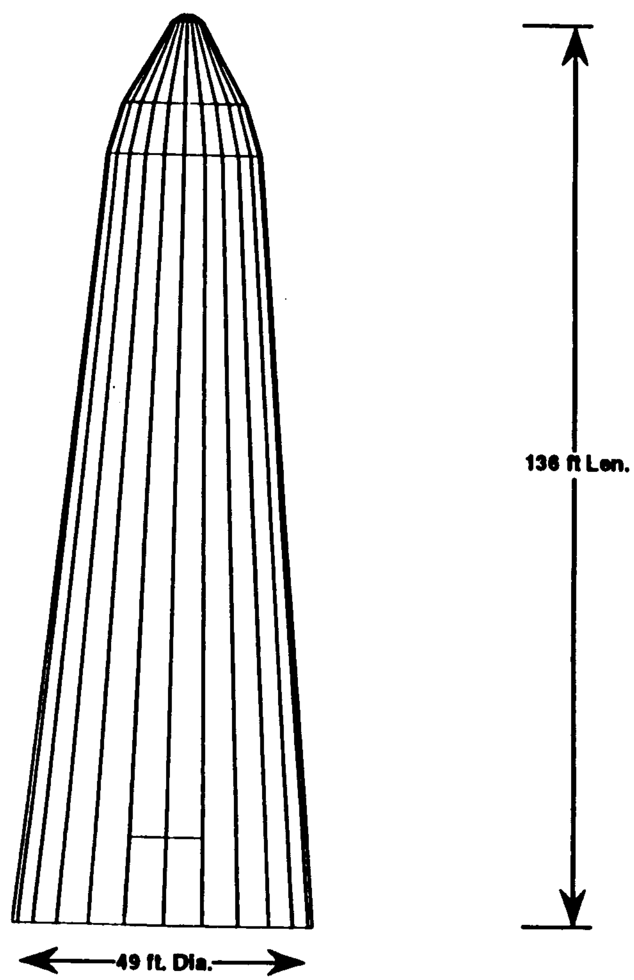
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,876,495 lbm
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

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

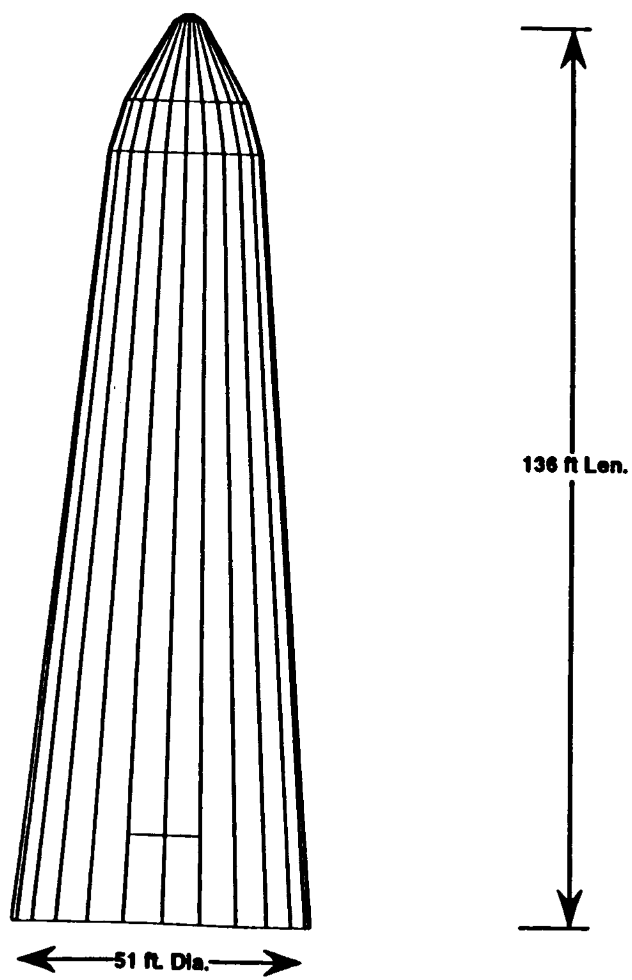
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,826,799 lbm
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

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

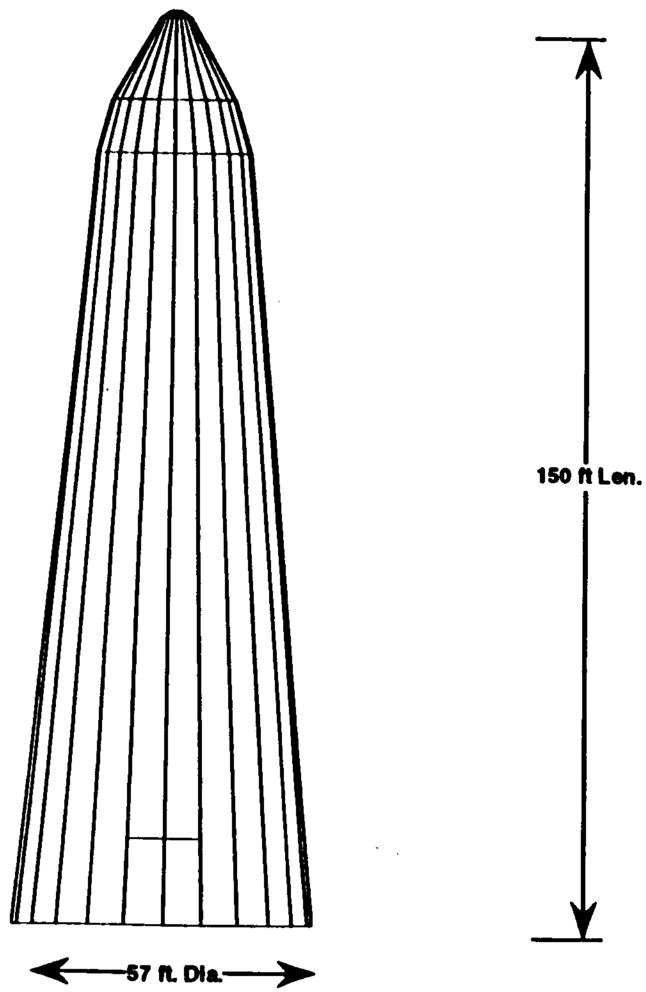
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,352,271 lbm
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

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

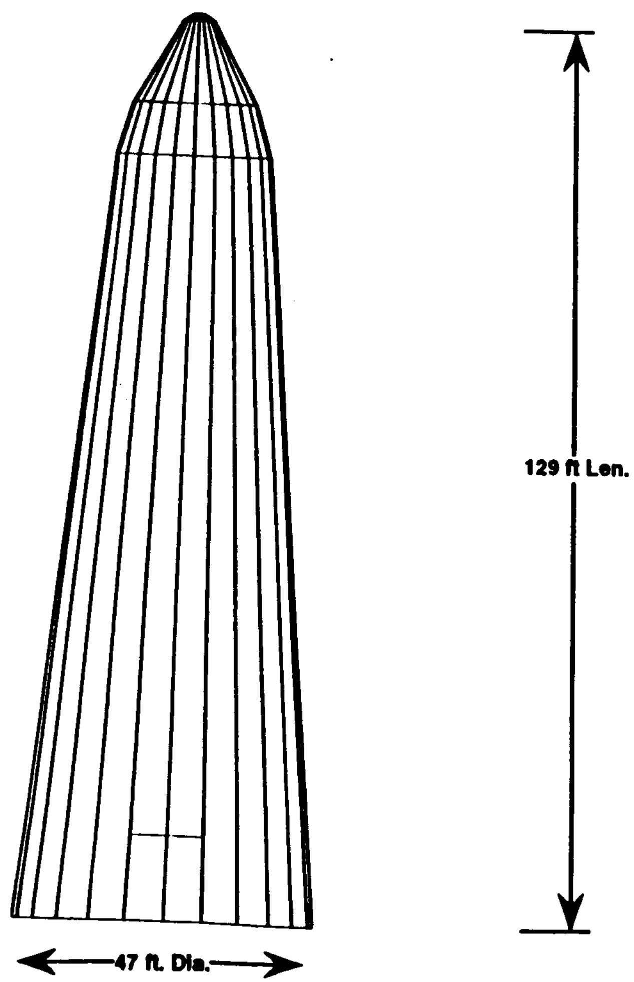
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,706,355 lbm
<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

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

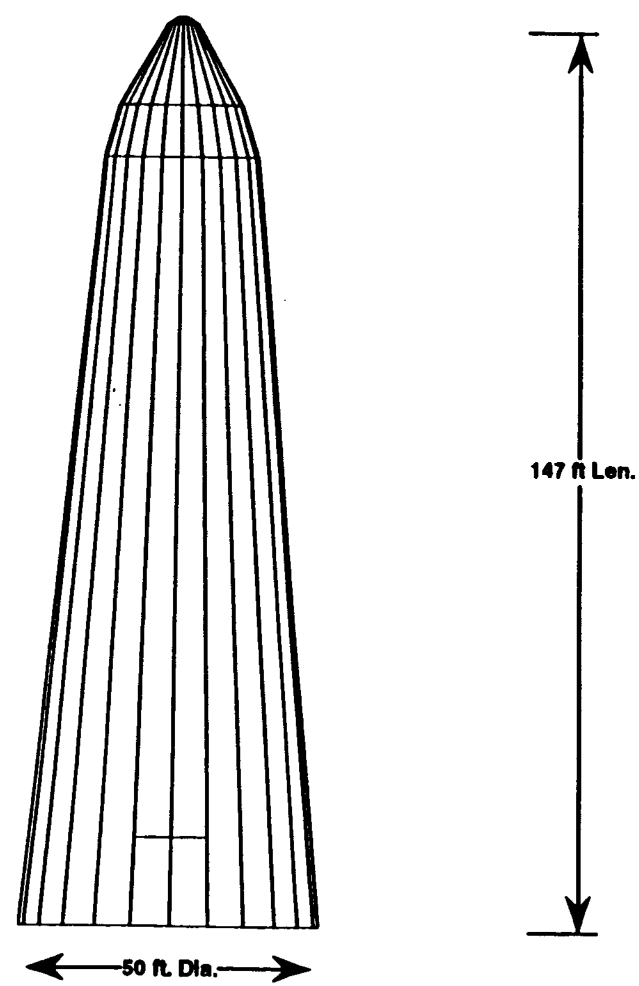
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,836,795 lbm
<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

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

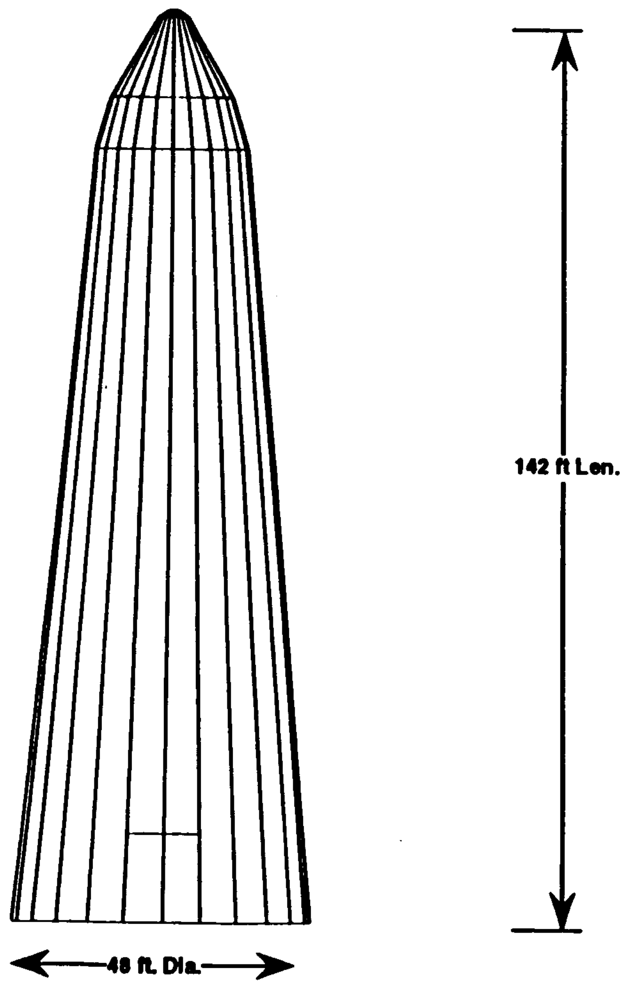
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,681,674 lbm
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

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

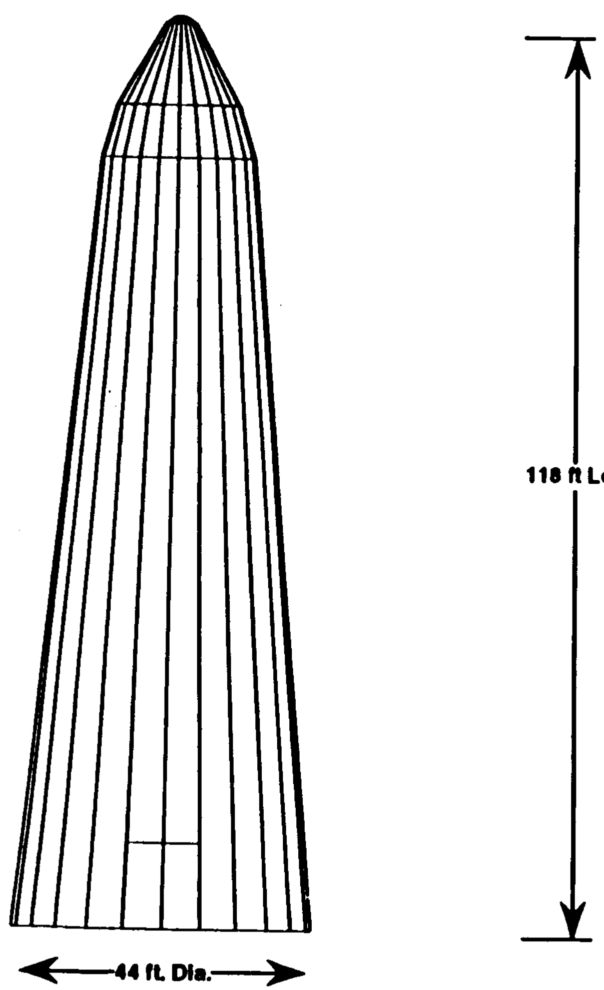
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,495,971 lbm
<u>Vehicle Specifications</u>:	
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
<u>Mode 1 Propulsion Specifications</u>:	
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
<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)	208,182 lbf
Vacuum Isp (@100 % RPL):	462.0 sec

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

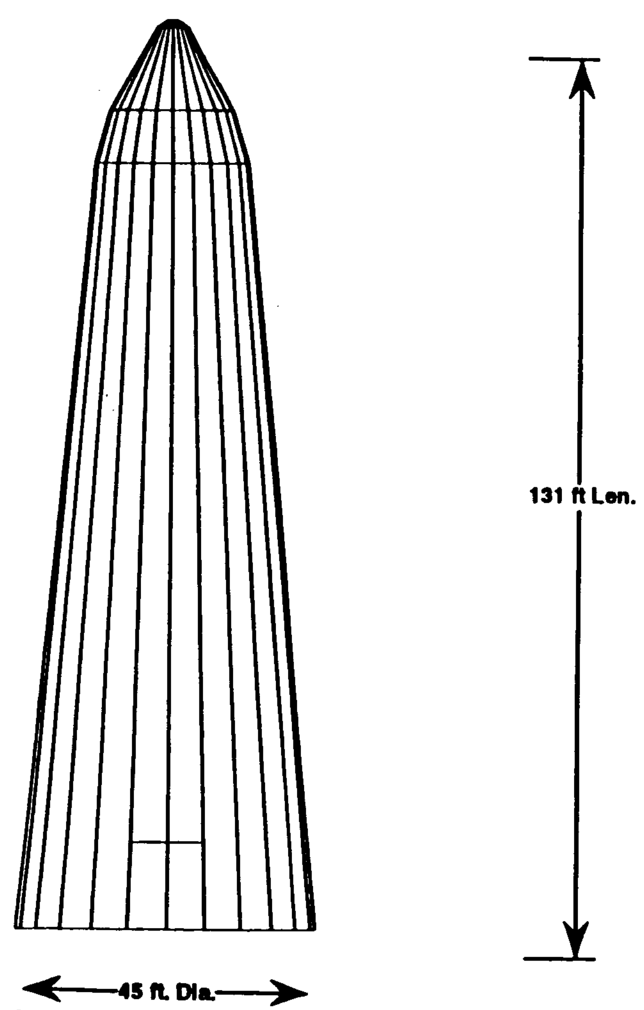
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,594,567 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	1,066,474 lbf
Vacuum Isp (@100 % RPL):	461.0 sec

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

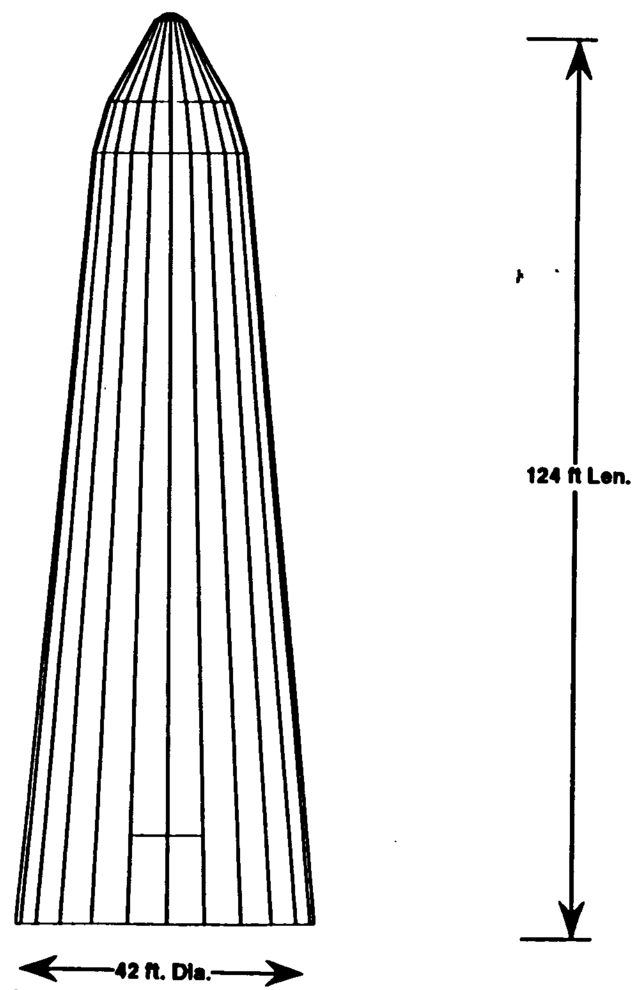
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,117,862 lbm
<u>Vehicle Specifications</u>:	
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
<u>Mode 1 Propulsion Specifications</u>:	
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
<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)	1,242,296 lbf
Vacuum Isp (@100 % RPL):	461.0 sec

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

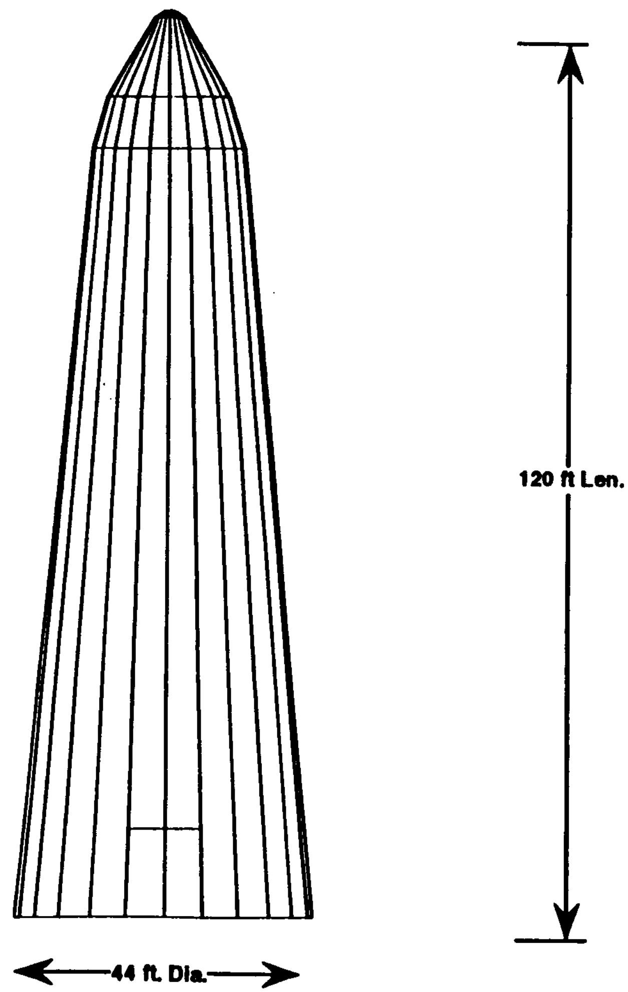
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,806,057 lbm
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

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

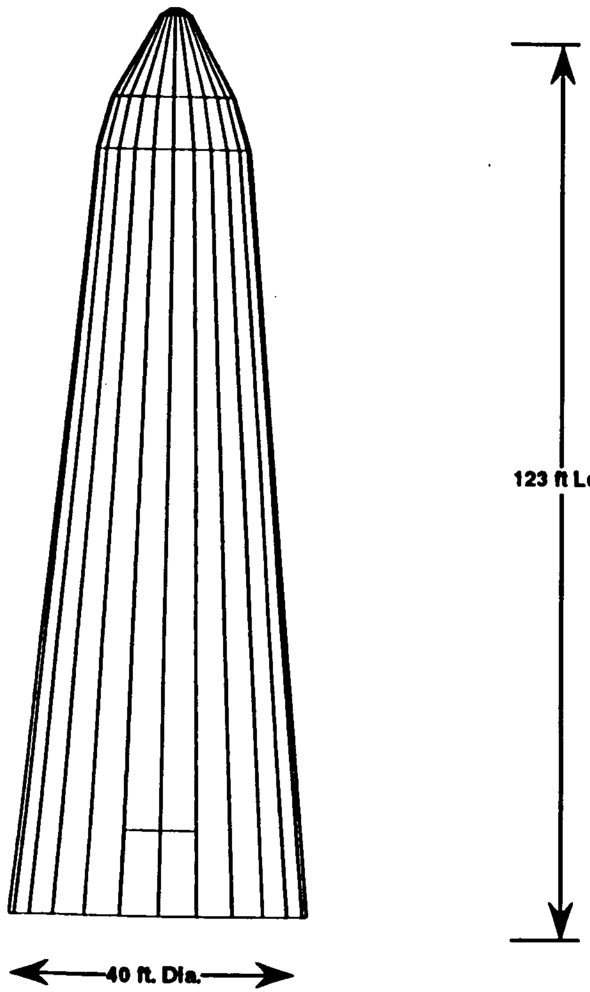
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,428,257 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	982,594 lbf
Vacuum Isp (@100 % RPL):	460.0 sec

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

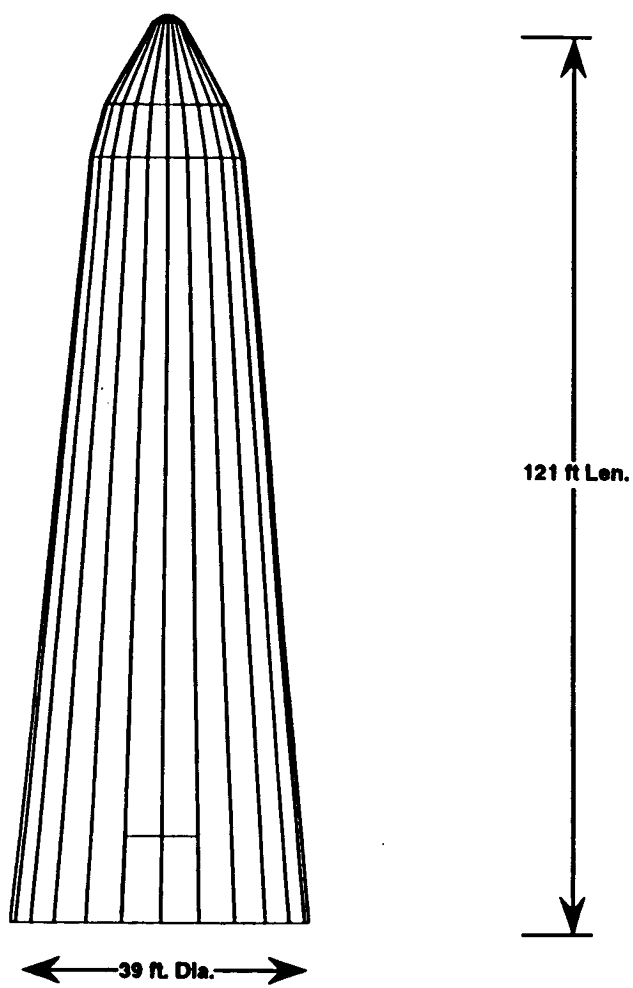
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,388,459 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	861,700 lbf
Vacuum Isp (@100 % RPL):	460.0 sec

Keith Holden
205-722-4531

VTOL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,357,839 lbm
<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

Keith Holden
205-722-4531

VTHL Winged Body Concept Results

- This configuration has no additional technology assumptions

Configuration Specific Sizing Groundrules

- Maximum normal acceleration is 2.5 Gs (sensitivity trade study should be performed)
- Payload bay weight is 5,786 lbm (Option 3 vehicle payload bay mass and mass of the faring 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

Keith Holden
205-722-4531

VTHL Winged Body Concept Results
(Continued)

Configuration Specific Subsystem Mass Relationships

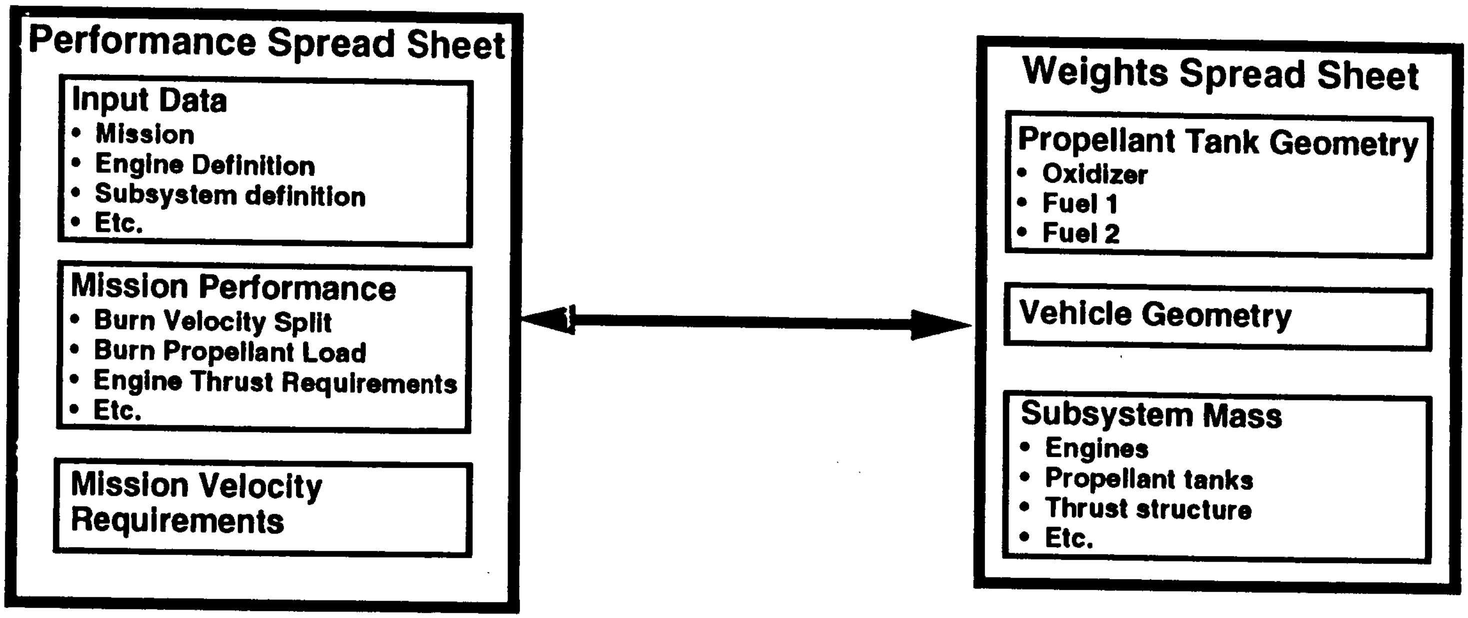
- Unpressurized structure mass is a function of the wetted surface area
- Unpressurized structure unit mass is a function of the maximum normal acceleration and the safety factor (detailed loads/airloads calculations should be added)
- Tip fin mass is a function of the planform area

Additional Propellant Tank Layout Choices in the Weight Spreadsheet

- The fore-to-aft arrangement of the propellant tanks are user defined
- The payload bay may be in the forward or aft intertank
- The propellant tanks may have separate, nested or common endcaps
- The propellant tanks may be cylindrical or toroidal
- There are no additional spreadsheets in this tool

Keith Holden
205-722-4531

Winged Body VTHL Configuration Model



Keith Holden
205-722-4531

VTHL Winged Body Concept Results
(Concluded)

- The weights spreadsheet splits the wetted body and wing areas into zones to calculate the TPS mass

- This configuration was modeled to provide a calibration of the sizing tool vehicle mass estimates against a known vehicle design
  - The sizing tool dry mass estimate for the launch vehicle configuration using the Option 3 RD-701 engine and Gr-Ep LH2 tank is 162,145 lbm
  - The projected dry mass by the Access to Space Advanced Technology vehicle design team Option 3 RD-701 engine and Gr-Ep LH2 tank is 130,218 lbm
  - The sizing tool dry mass estimate for the launch vehicle configuration using the Option 3 evolved SSME and Gr-Ep LH2 tank is 251,480 lbm
  - The projected dry mass by the Access to Space Advanced Technology vehicle design team Option 3 evolved SSME and Gr-Ep LH2 tank is 198,980 lbm
  - This difference in launch vehicle configuration dry masses can be explained by our respective estimates of the thrust structure mass

- No sizing sensitivity trades were performed

Keith Holden
205-722-4531

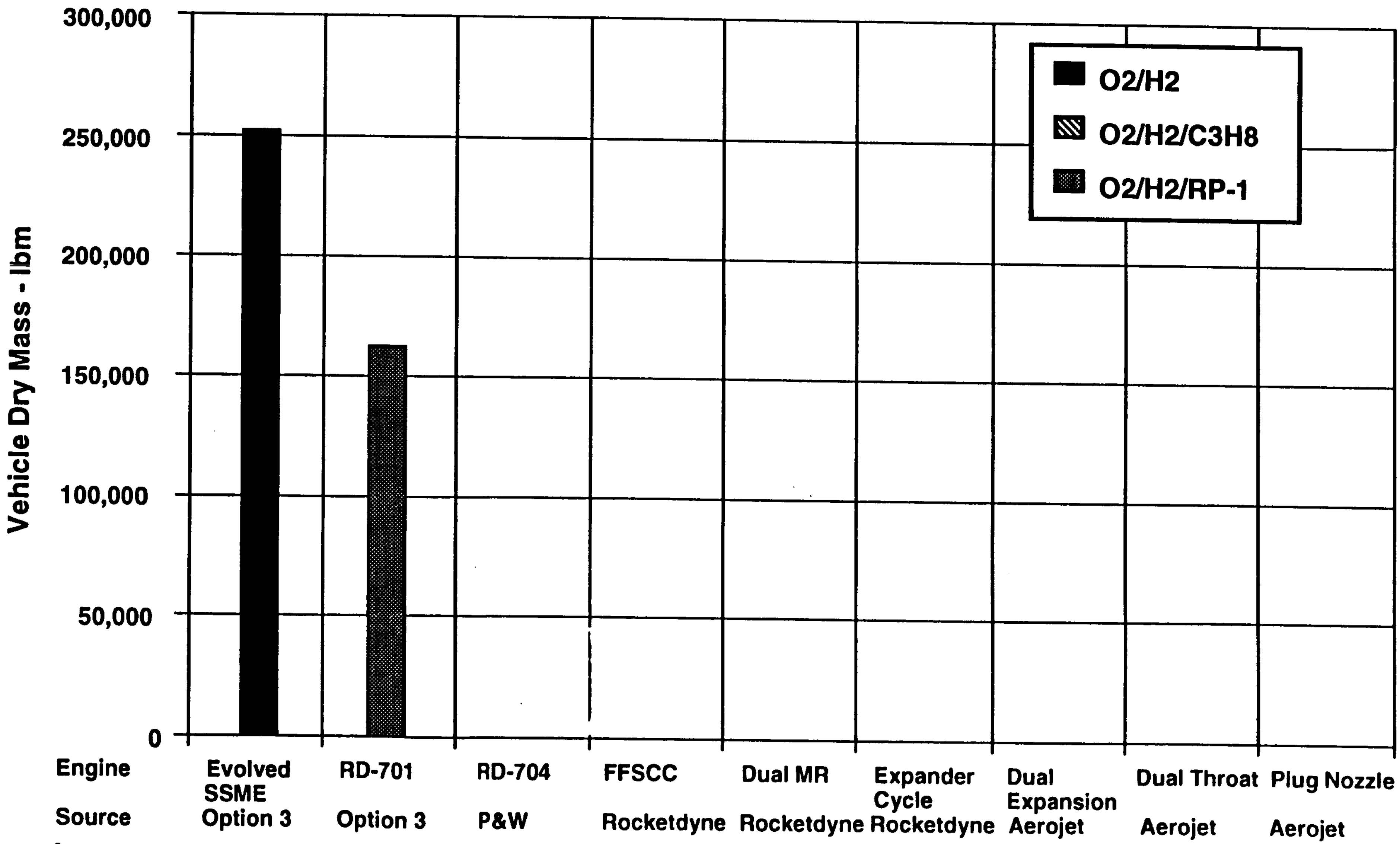
VTHL Winged Body Sizing Conclusions

- The following chart is a bar chart showing the dry mass of the winged body VTHL vehicle configuration using the Option 3 versions of the RD-701 engine and the evolved SSME and cut sheets describing both configurations in more detail

- The dry mass of the vehicle configuration using the Option 3 version of the RD-701 engine is significantly smaller than the dry mass of the vehicle configuration using the Option 3 version of the evolved SSME

Keith Holden
205-722-4531

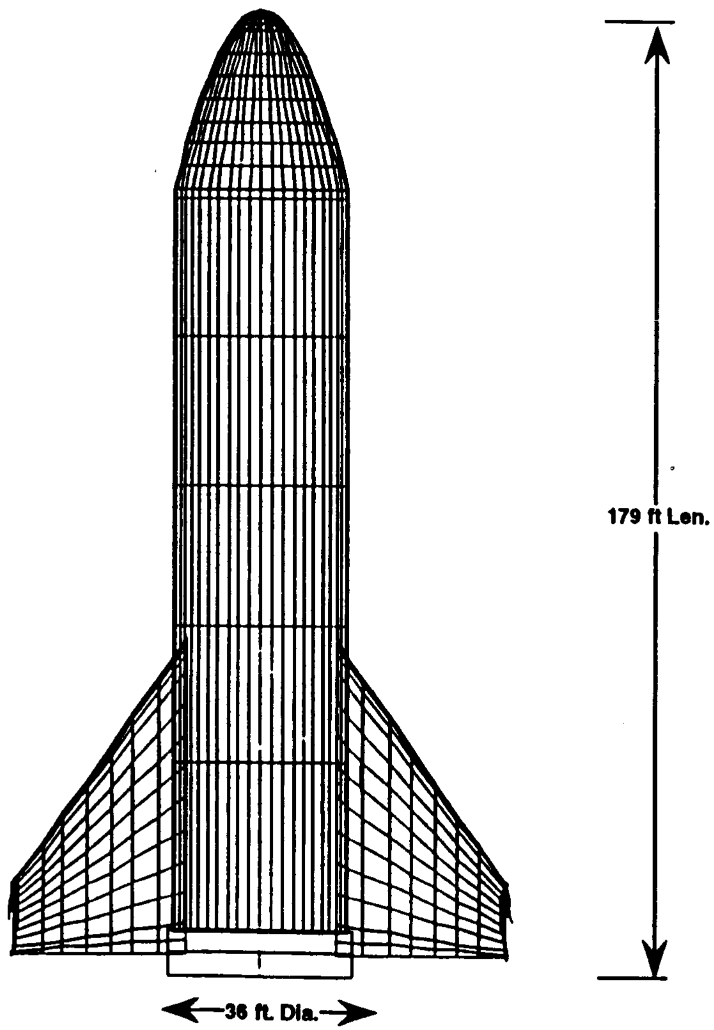
Winged Body VTHL Vehicle Dry Mass as a Function of Engine and Propellant Used



Keith Holden
205-722-4531

VTHL Winged Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

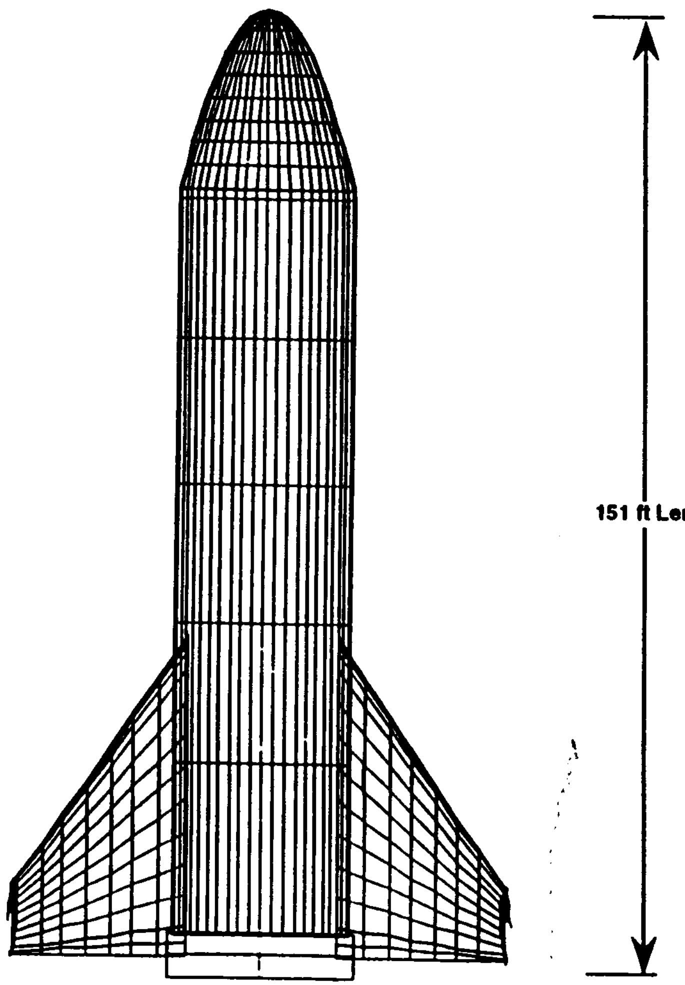
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,647,250 lbm
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

Keith Holden
205-722-4531

VTHL Winged Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,994,088 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	199,296 lbf
Vacuum Isp (@100 % RPL):	452.7 sec

Keith Holden
205-722-4531

VTHL Lifting Body Concept Results

- This configuration has no additional technology assumptions

<u>Configuration Specific Sizing Groundrules</u>

- Maximum normal acceleration is 1.6 Gs (from unconstrained trajectory results)
- 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

Keith Holden
205-722-4531

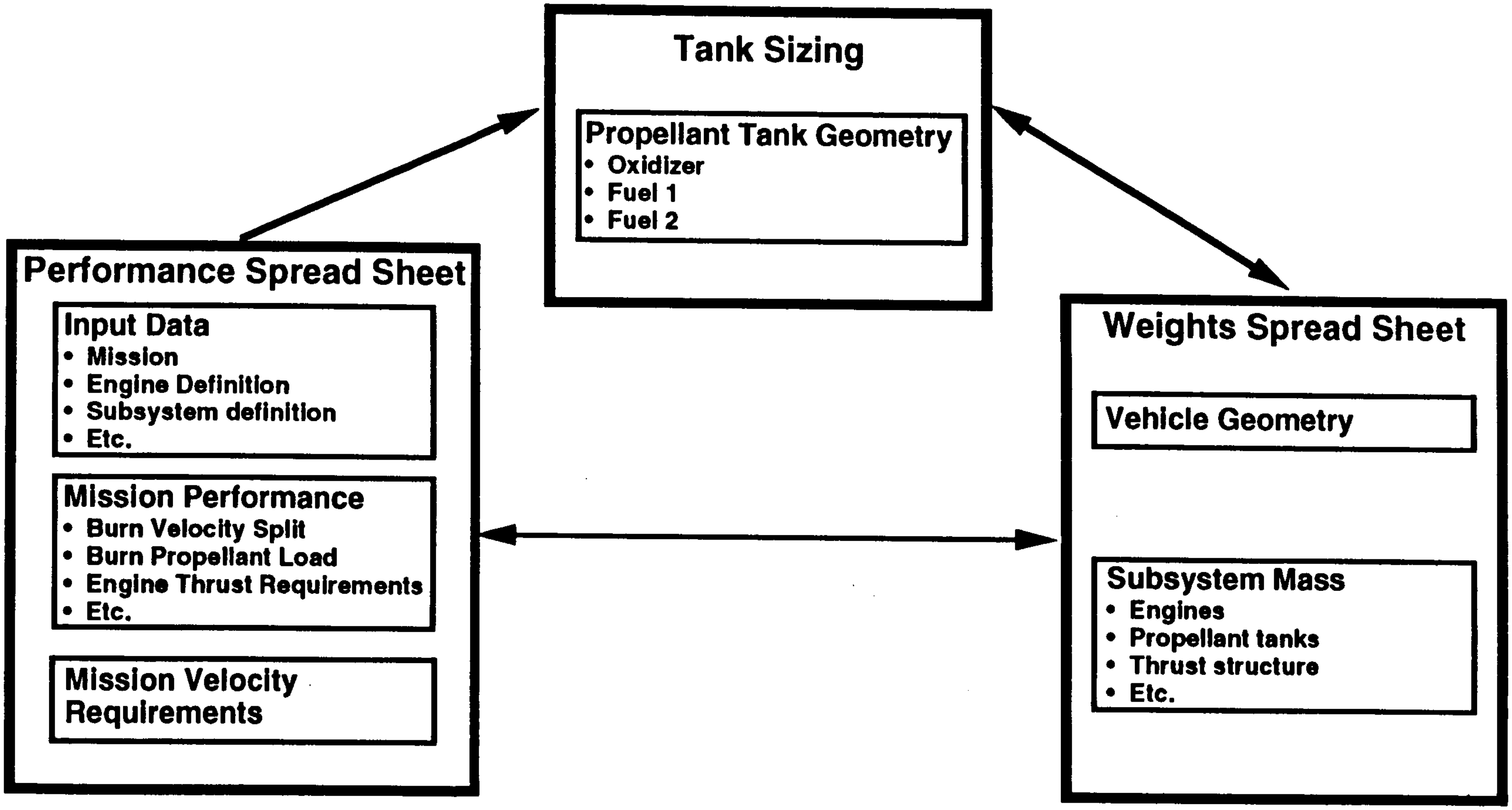
VTHL Lifting Body Concept Results (Continued)

Configuration Specific Subsystem Mass Relationships

- Body aeroshell mass is a function of the wetted aeroshell surface area
- Body unit mass is a function of the maximum normal acceleration and the safety factor (detailed loads/airloads calculations should be added)
- Tip fin mass is a function of the planform area

Keith Holden
205-722-4531

Lifting Body VTHL Configuration Model



Keith Holden
205-722-4531

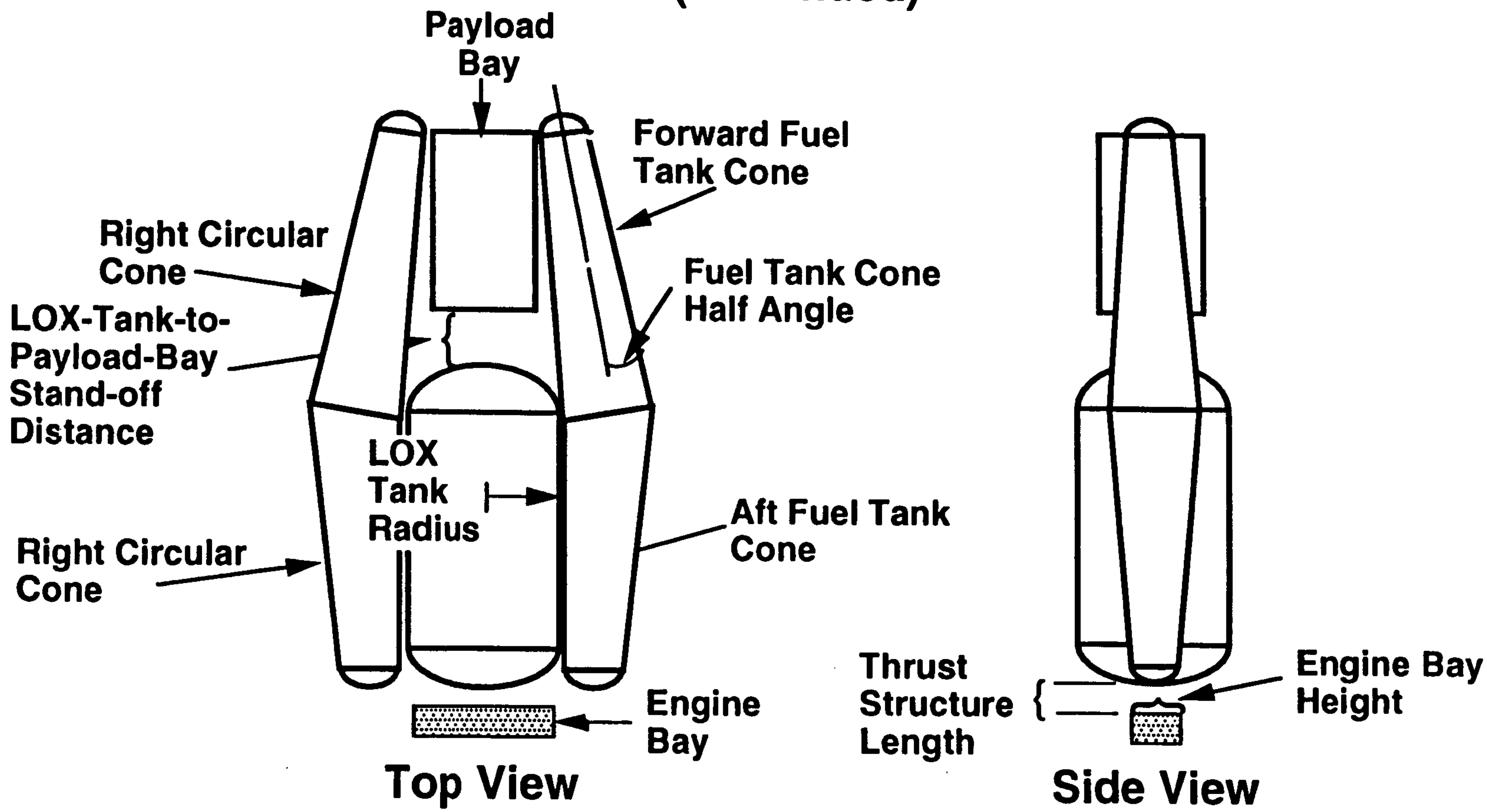
VTHL Lifting Body Concept Results
(Continued)

<u>Propellant Tank Sizing Spreadsheet</u>

- The propellant tank configuration is hardwired into the spreadsheet
- The payload bay is forward of the oxidizer tank
- The oxidizer tank forward and aft radii may be independently varied
- The fuel is split into port and starboard tanks
- The fuel tanks are on the side of the oxidizer tank and payload bay
- If two fuels are used, the fuel one tanks are smaller and forward of the fuel two tanks

Keith Holden
205-722-4531

VTHL Lifting Body Concept Results
(Continued)



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

Keith Holden
205-722-4531

VTHL Lifting Body Concept Results
(Continued)

- The weights spreadsheet,
  - Uses inputs from the tank sizing spreadsheet to calculate a vehicle planform area
  - Uses user defined inputs to go from vehicle planform area to wetted body area
  - Splits the wetted body area into zones to calculate the TPS mass
- The following are the independent parameters that change the tank planform area and therefore the vehicle geometry and mass
  - The forward and aft oxidizer tank radii
  - The half angle of the forward fuel tank cone
  - The engine bay height

Keith Holden
205-722-4531

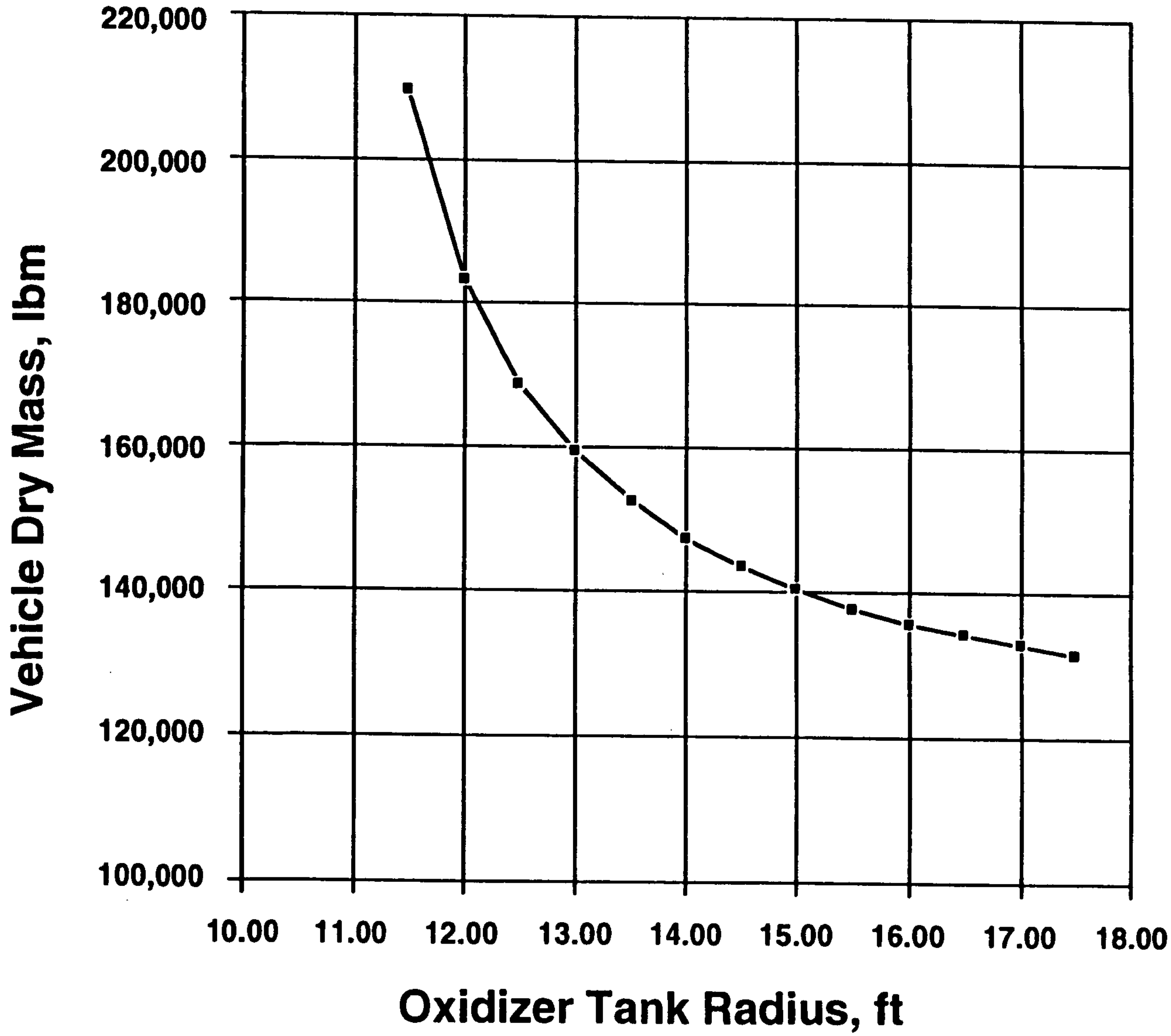
VTHL Lifting Body Concept Results
(Continued)

- The following pages show the trends in the lifting body configuration dry mass as these parameters are changed
  - After a new lifting body geometry is chosen, it will be necessary to generate a new set of aero and a new set of body coefficients
  - The vehicle must then be resized using this new data
  - It will be an iterative process of generating a new set of trends and then verifying the configuration

Keith Holden
205-722-4531

VTHL Lifting Body Concept Results
(Continued)

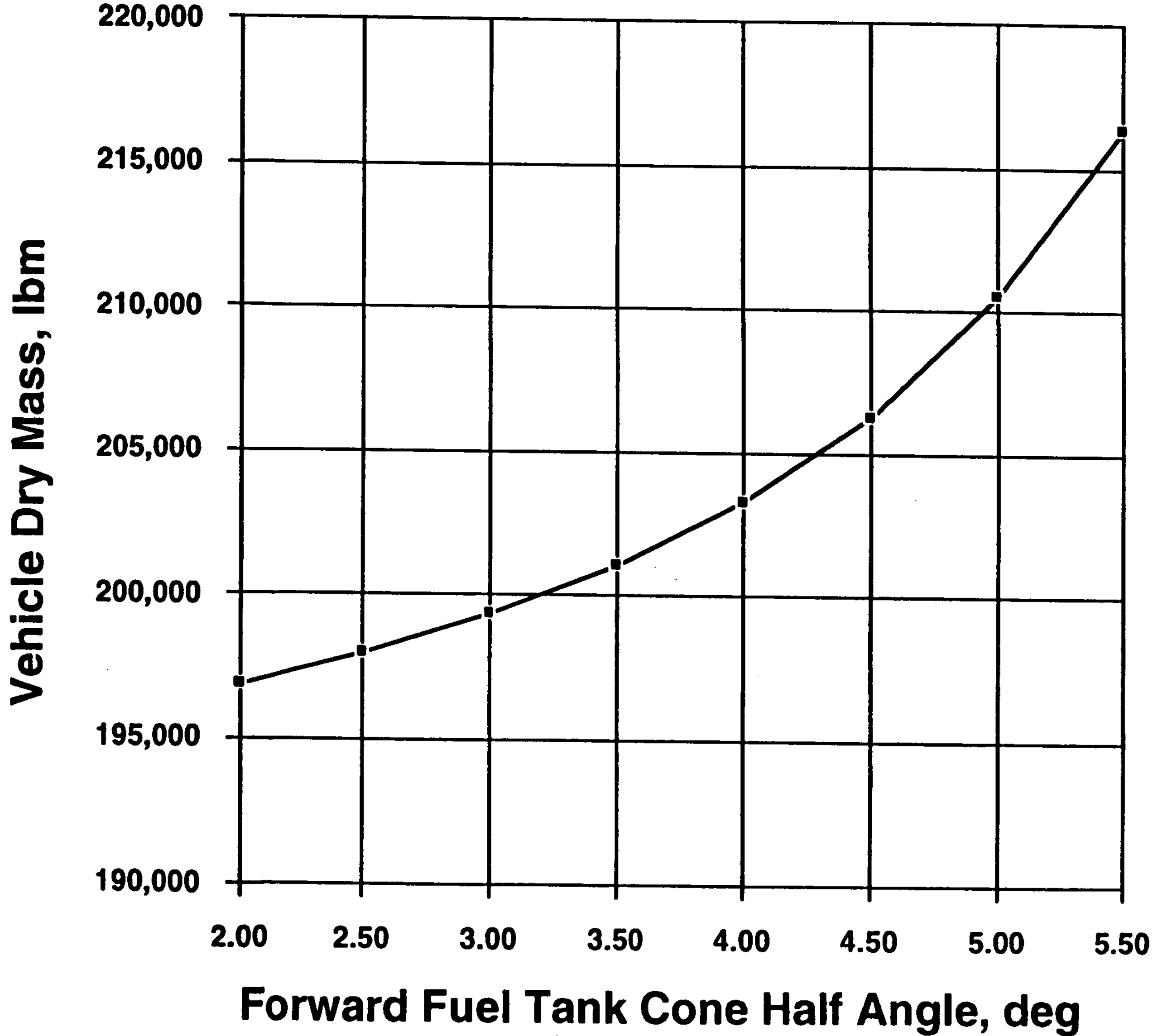
Vehicle Dry Mass as a Function of Oxidizer Tank Radius



Keith Holden
205-722-4531

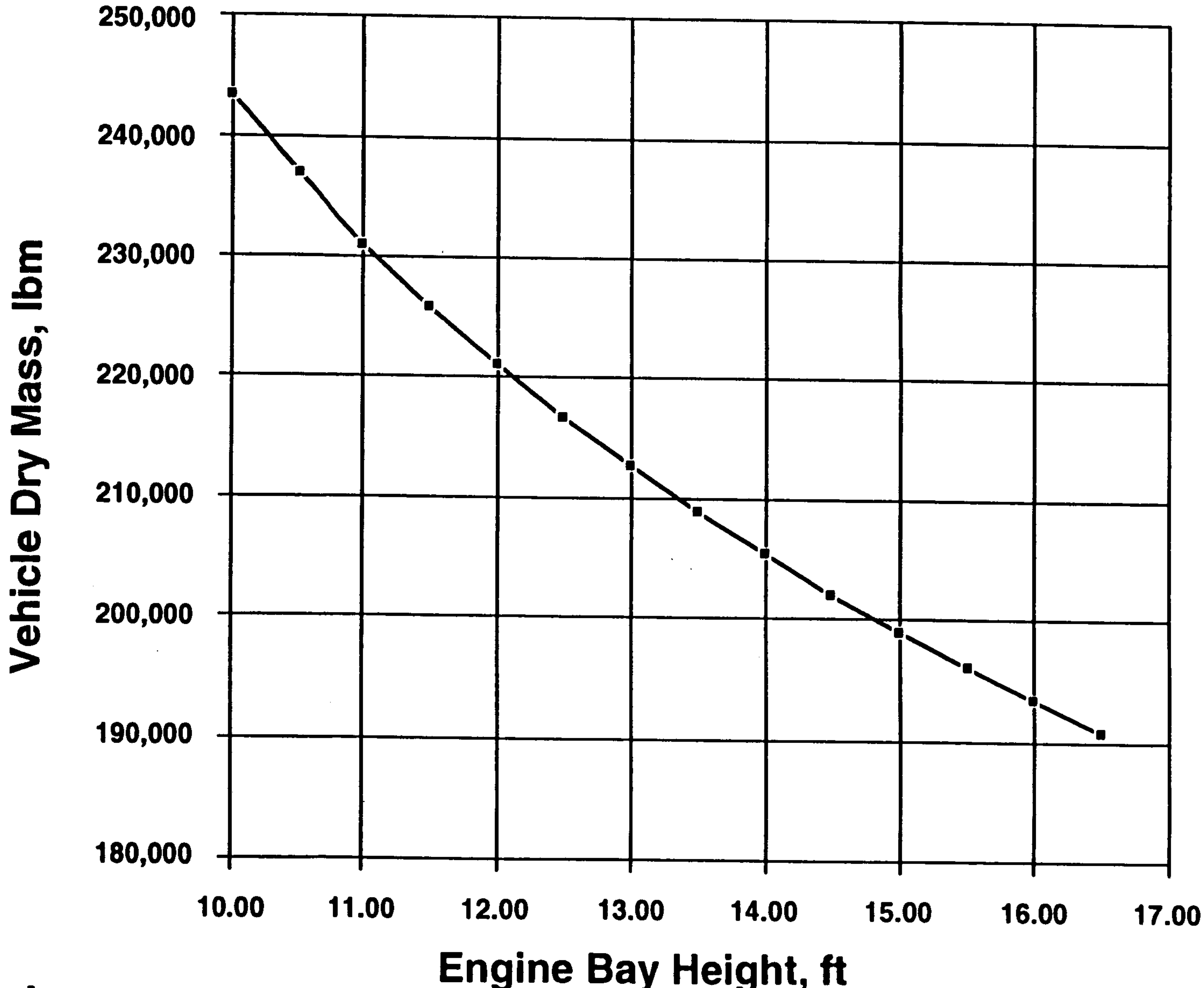
VTHL Lifting Body Concept Results
(Continued)

Vehicle Dry Mass as a Function of the Forward Fuel Tank Cone Half Angle



Keith Holden
205-722-4531

VTHL Lifting Body Concept Results
(Concluded)
Vehicle Dry Mass as a Function of Engine Bay Height



Keith Holden
205-722-4531

VTHL Lifting Body Sizing Conclusions

- Increasing tank diameter shortens aft fuselage, widens fuel tanks, and decreases wetted surface area (and dry weight) but hurts stability margin by moving c.p. forward
  - May need to increase tip fin size to move c.p. aft again
- Fuel tank half angle is not as strong a dry weight and stability influence
- Engine bay height is a stronger influence than fuel tank half angle, increases base area

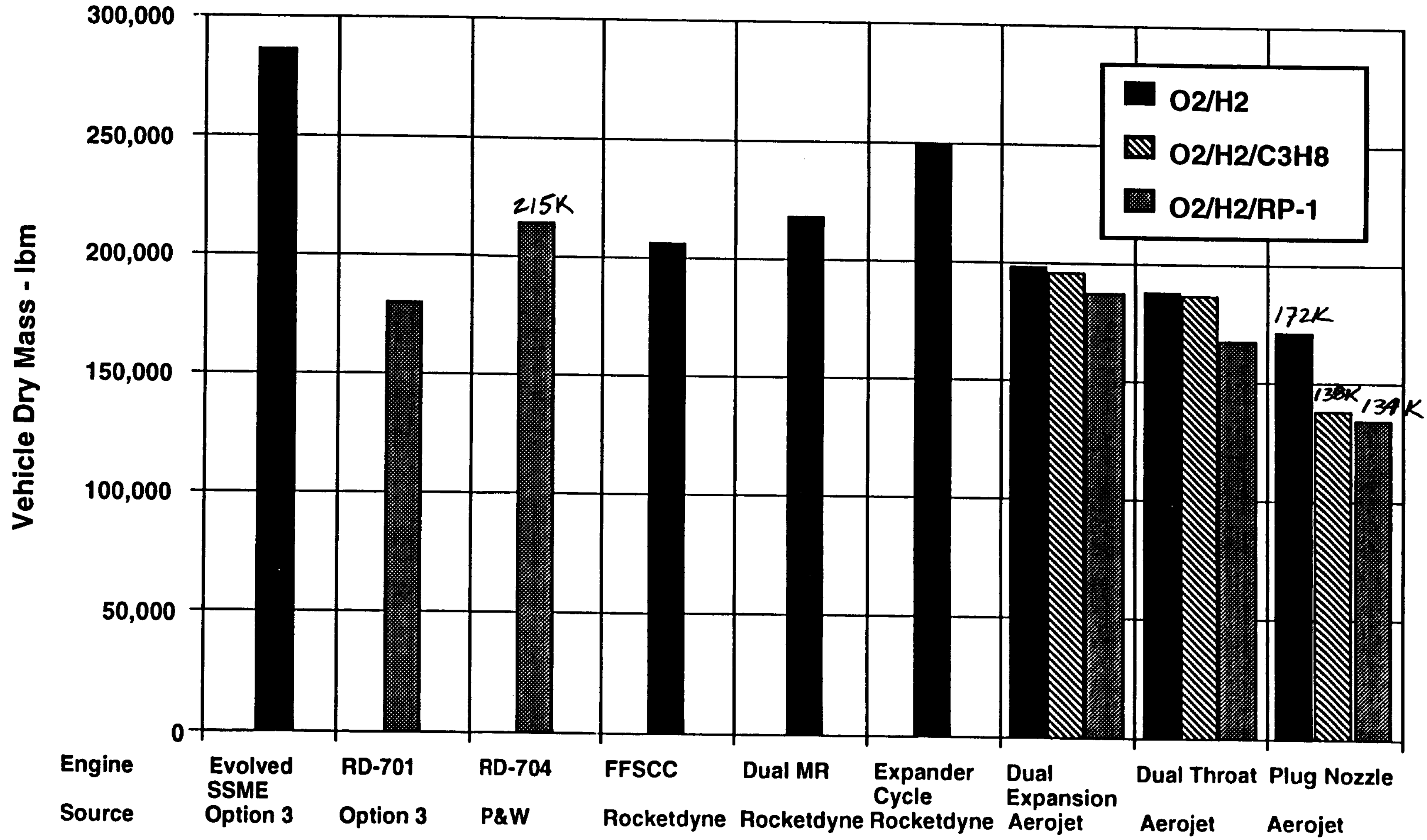
Keith Holden
205-722-4531

VTHL Lifting Body Sizing Conclusions
(Concluded)

- The following chart is a bar chart showing the dry mass of the lifting body VTHL vehicle configurations as a function of the engine and propellant used on the configuration and cut sheets describing each configuration in more detail
- The best propellant is the tripropellant combination of O2/H2/RP-1
- The tripropellant concept payoff is a function of the engine type used
- There was a large difference in the dry mass of the two vehicle configurations with different versions of the Russian tripropellant engine (the Option 3 version of the RD-701 engine had less hydrogen flow during the mode one burn than the Pratt version of the RD-704 engine)
- The vehicle configuration with the full flow staged combustion cycle (FFSCC) engine was significantly lighter than the vehicle configuration with the Option 3 version of the evolved SSME
- The vehicle configurations using the dual mixture ratio (MR) engine and the expander cycle engine are heavier than the vehicle configuration using the FFSCC engine
- The lowest dry mass vehicle configuration used a plug nozzle engine

Keith Holden
205-722-4531

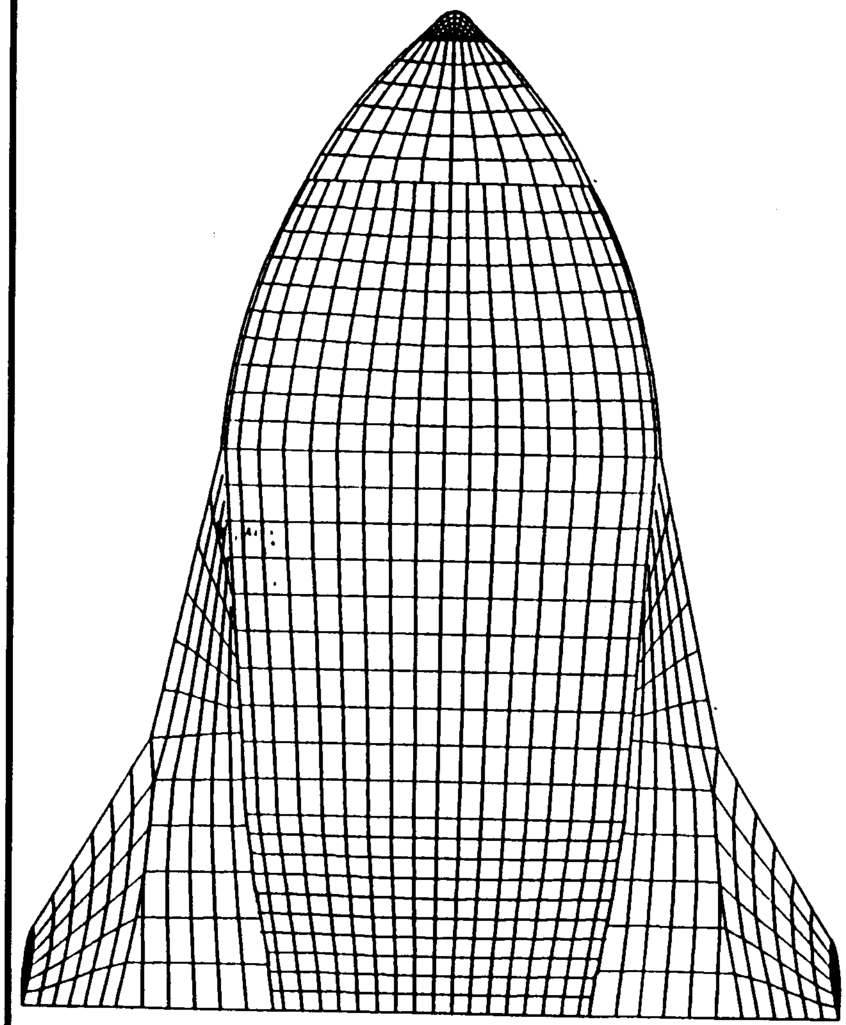
Lifting Body VTHL Vehicle Dry Mass as a Function of Engine and Propellant Used



Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

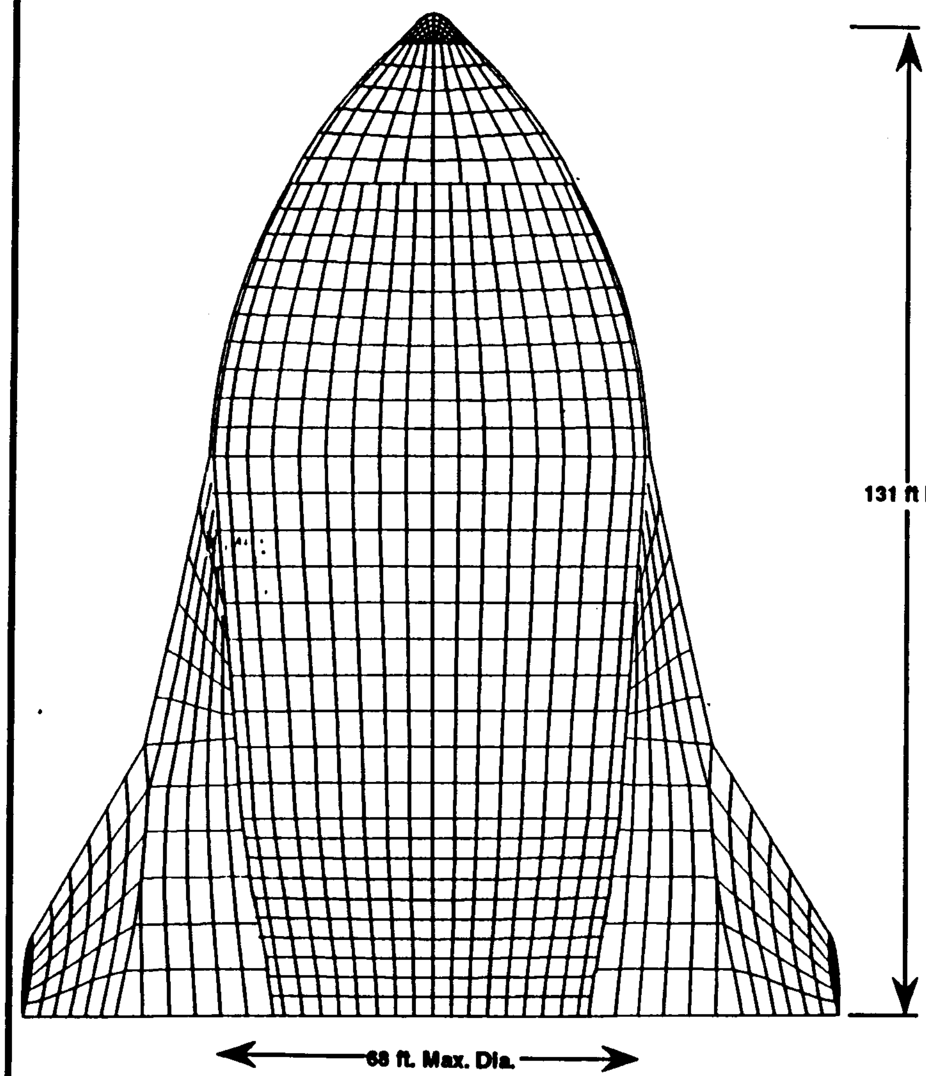
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,919,677 lbm
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

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

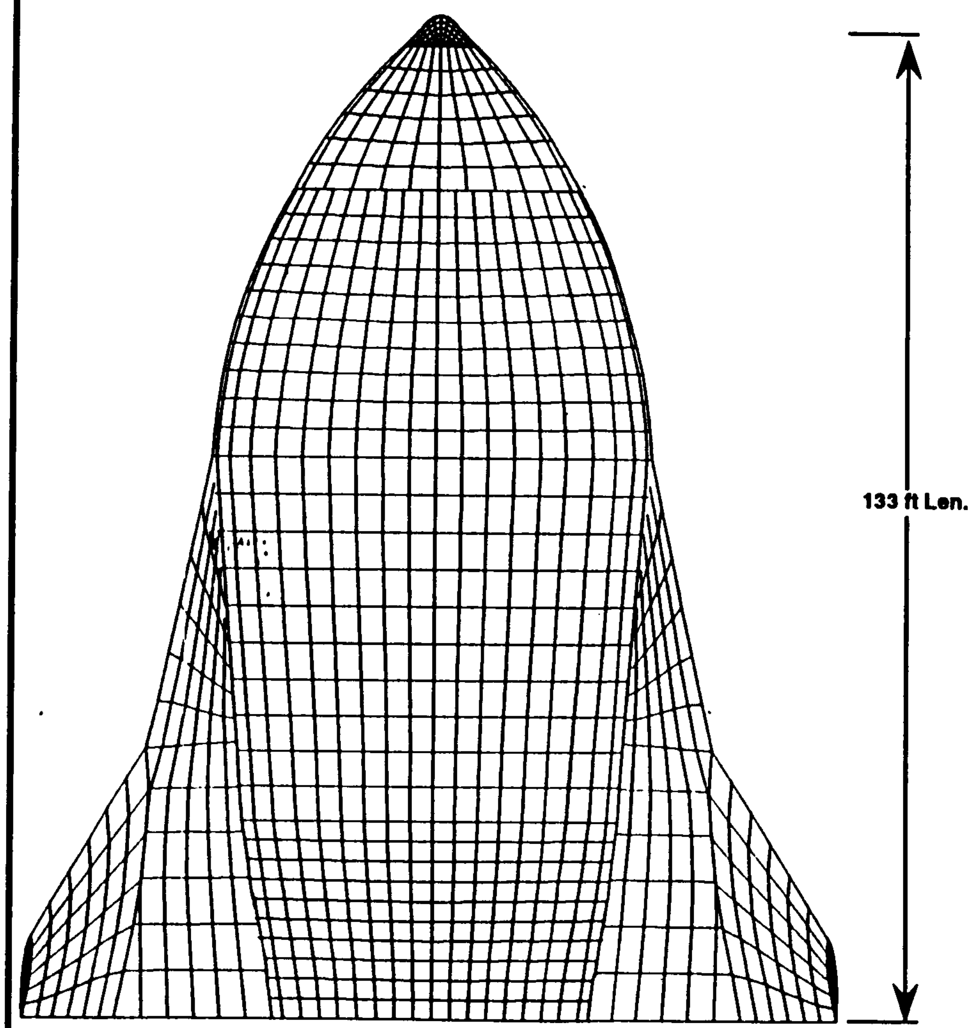
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,143,771 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	257,285 lbf
Vacuum Isp (@100 % RPL):	452.7 sec

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

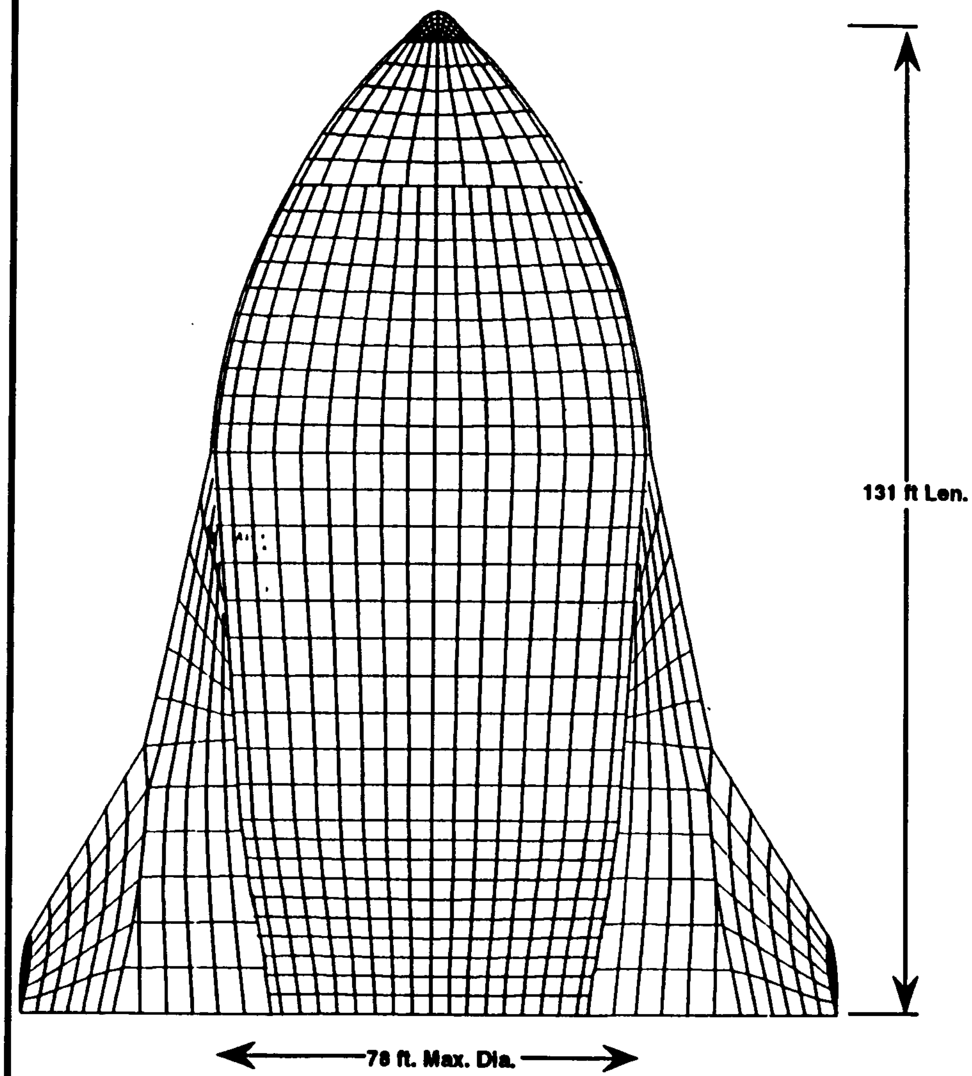
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,388,519 lbm
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

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

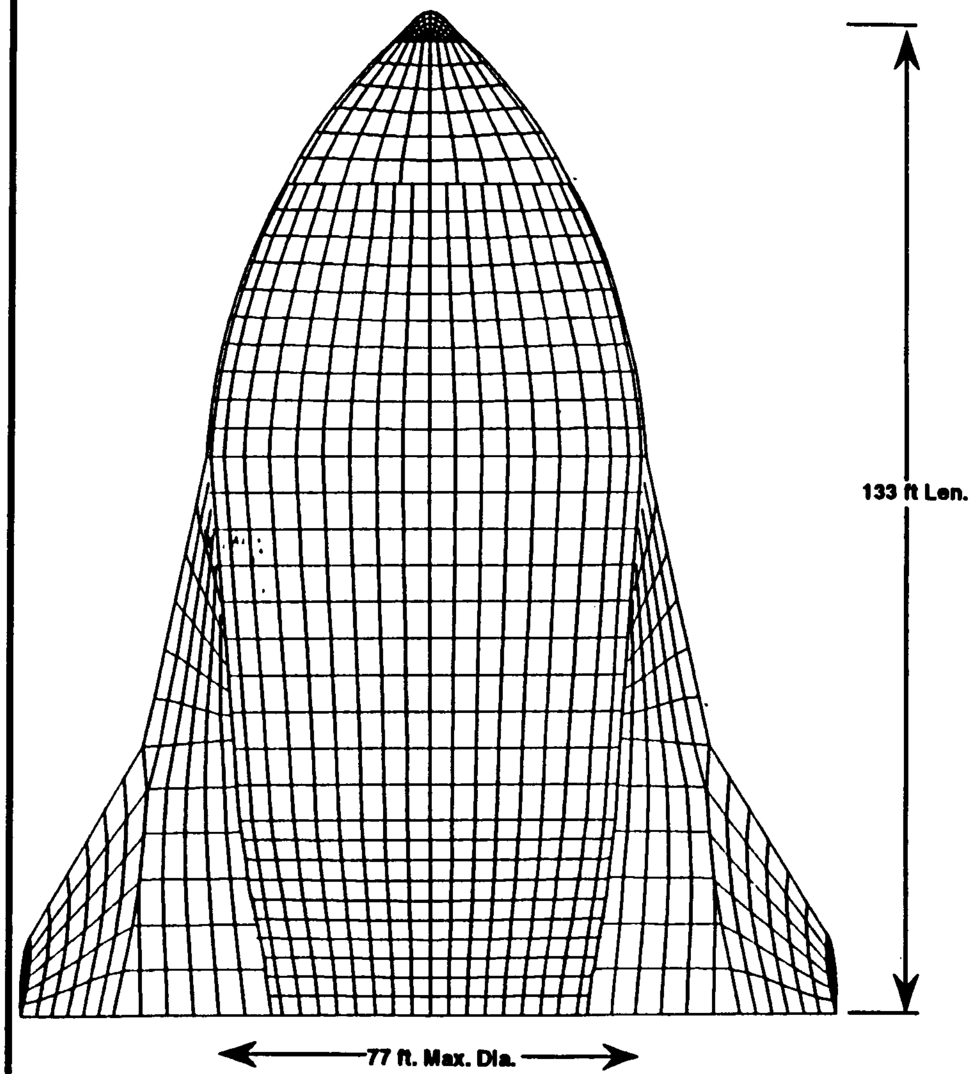
GLOW:	2,039,569 lbm
Vehicle Specifications:	
Vehicle Dry Mass @ Liftoff:	206,769 lbm
Usable Propellant Mass (including FPR):	
--Mode 1	1,037,595 lbm
--Mode 2	738,814 lbm
Propellant Combination:	
--Mode 1	LOX/LH2/Kerosene
--Mode 2	LOX/LH2
Ascent Residuals	9,276 lbm
OMS & RCS Propellant	22,116 lbm
Landing Propellant	N/A
Landing Specific Impulse	N/A
Main Engine Type/No.:	FFSCC/5
Mode 1 Propulsion Specifications:	
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
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)	280,553 lbf
Vacuum Isp (@100 % RPL):	460.3 sec

Keith Holden
205-722-4531

ATSS TEAM
Advanced Transportation Systems

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

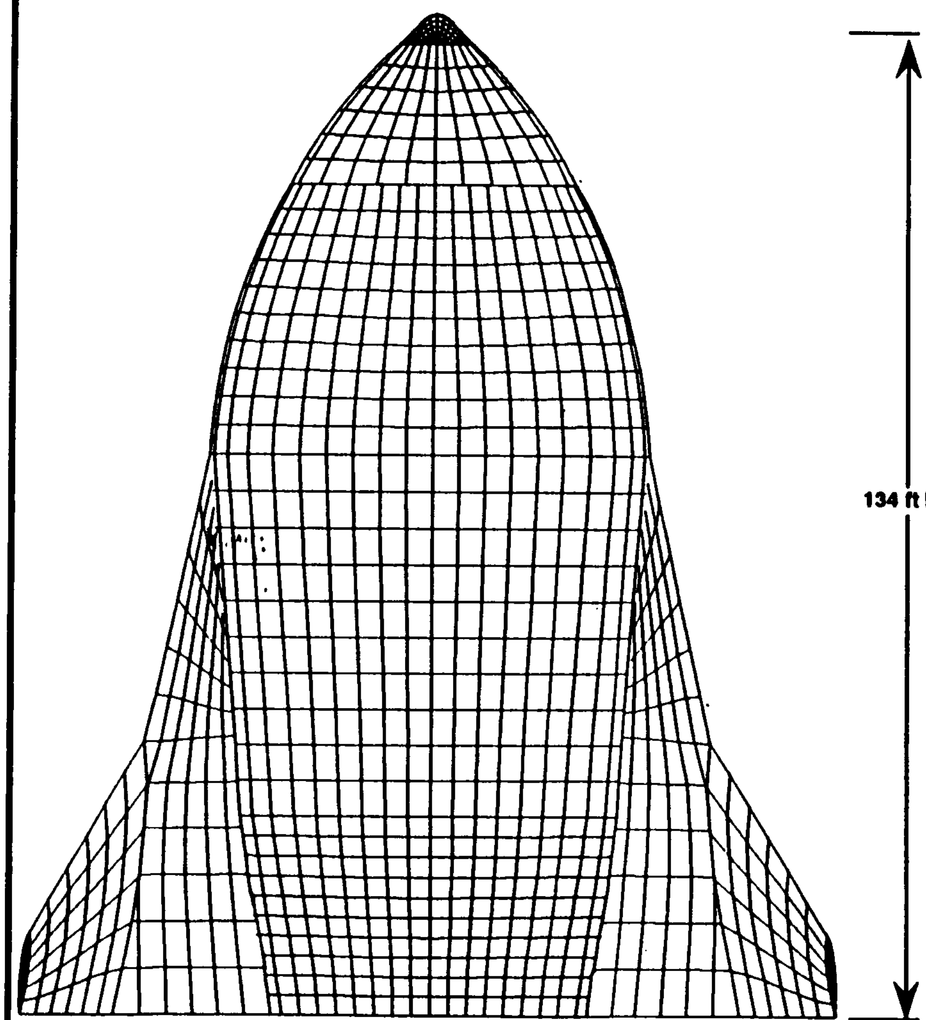
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,386,748 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	301,862 lbf
Vacuum Isp (@100 % RPL):	455.5 sec

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

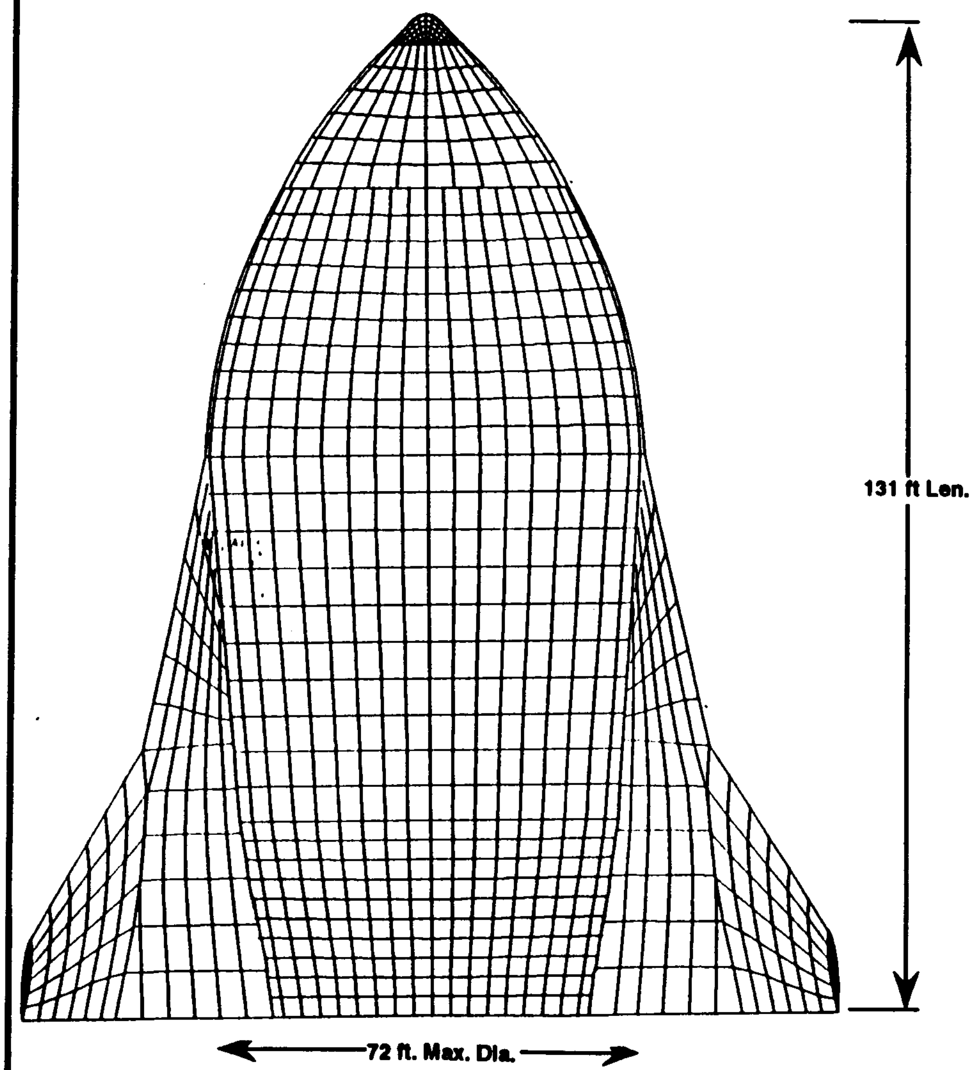
GLOW:	2,604,058 lbm
Vehicle Specifications:	
Vehicle Dry Mass @ Liftoff:	249,513 lbm
Usable Propellant Mass (including FPR):	
--Mode 1	1,390,464 lbm
--Mode 2	900,739 lbm
Propellant Combination:	
--Mode 1	LOX/LH2
--Mode 2	LOX/LH2
Ascent Residuals	12,147 lbm
OMS & RCS Propellant	26,195 lbm
Landing Propellant	N/A
Landing Specific Impulse	N/A
Main Engine Type/No.:	Dual Expander Cycle/5 EXPANDER CYCLE
Mode 1 Propulsion Specifications:	
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
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)	339,806 lbf
Vacuum Isp (@100 % RPL):	444.8 sec

Keith Holden
205-722-4531

ATSS TEAM
Advanced Transportation Systems

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

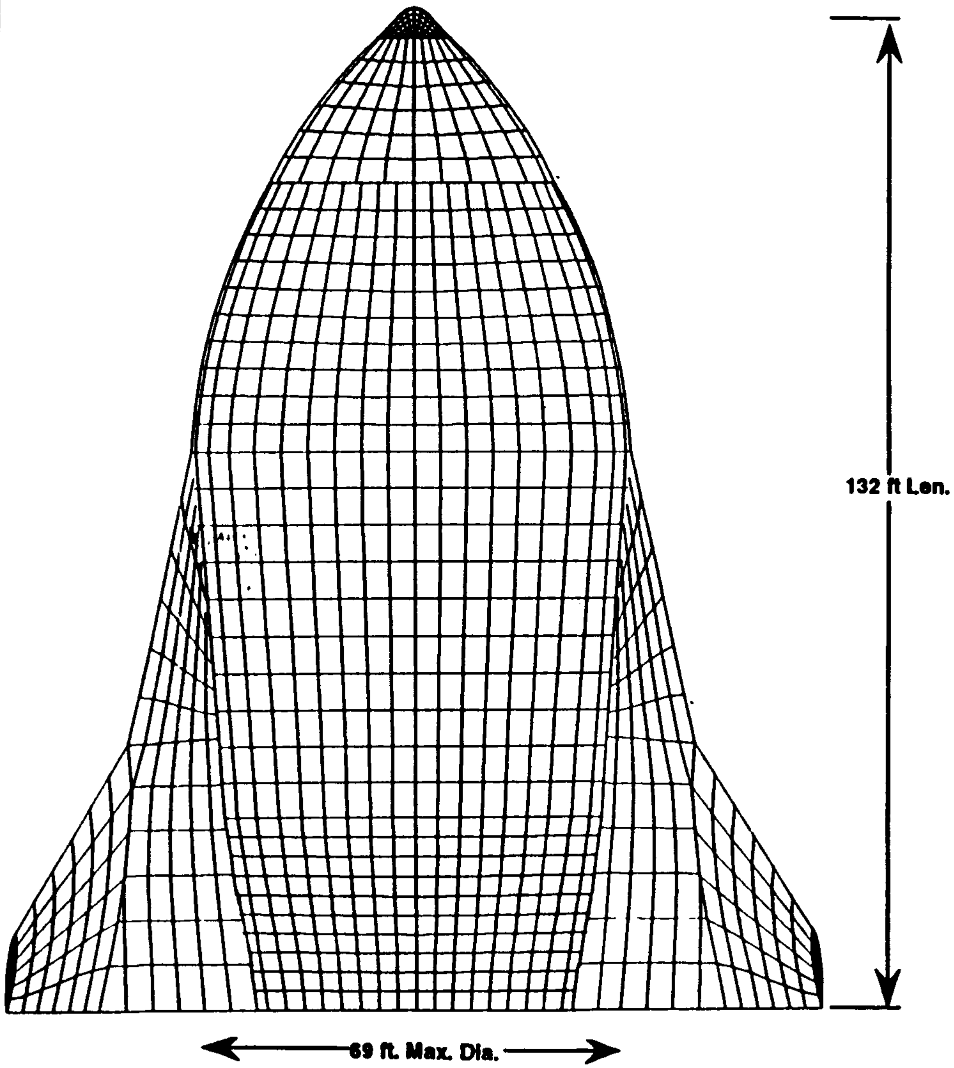
GLOW:	1,969,088 lbm
Vehicle Specifications:	
Vehicle Dry Mass @ Liftoff:	199,104 lbm
Usable Propellant Mass (including FPR):	
--Mode 1	1,018,482 lbm
--Mode 2	695,892 lbm
Propellant Combination:	
--Mode 1	LOX/LH2
--Mode 2	LOX/LH2
Ascent Residuals	9,225 lbm
OMS & RCS Propellant	21,385 lbm
Landing Propellant	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

Lockheed

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

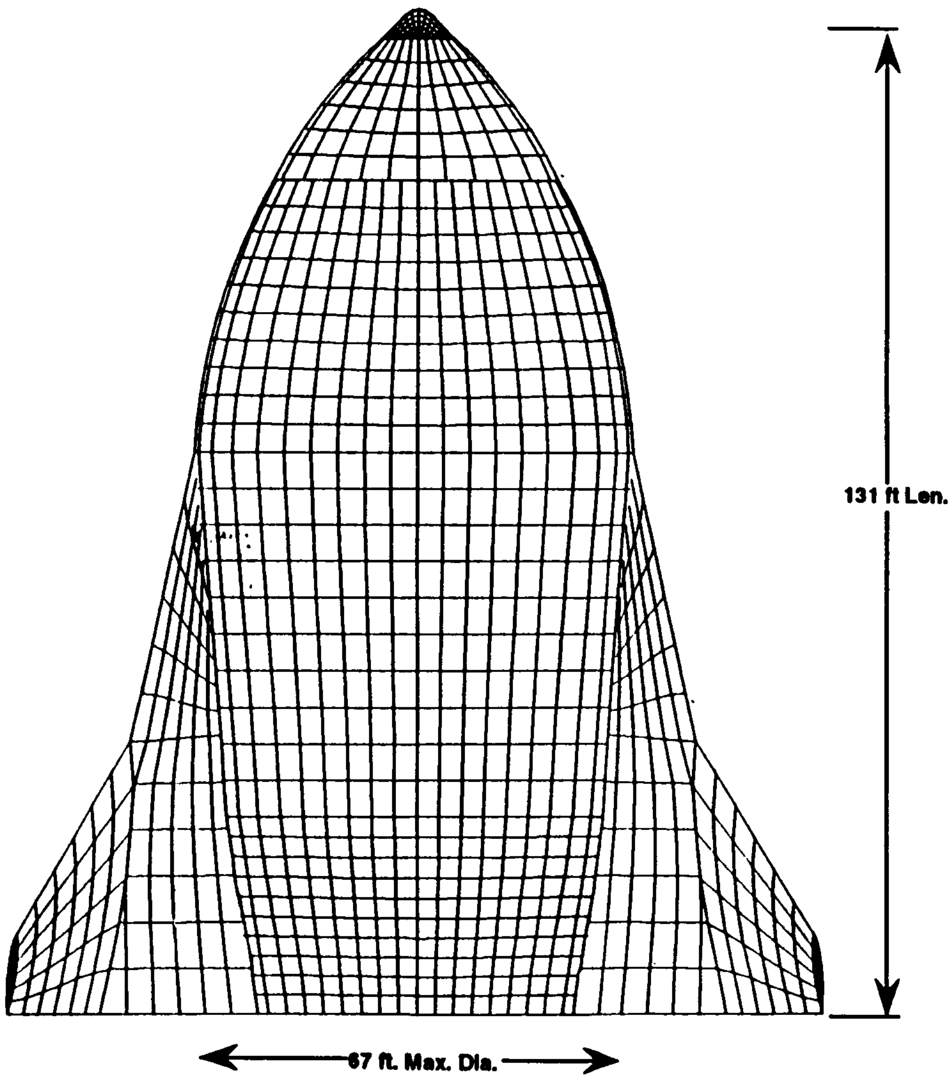
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,304,262 lbm
<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

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

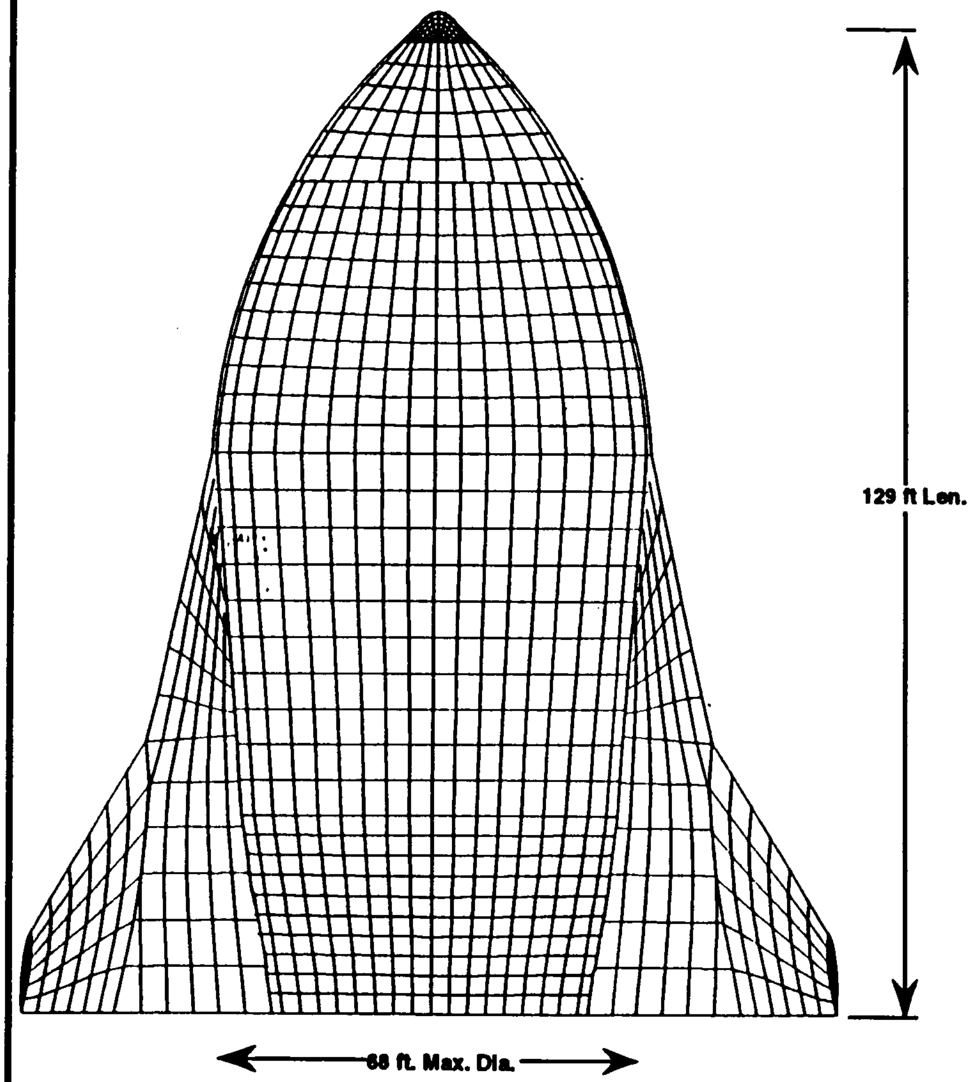
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,218,593 lbm
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

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

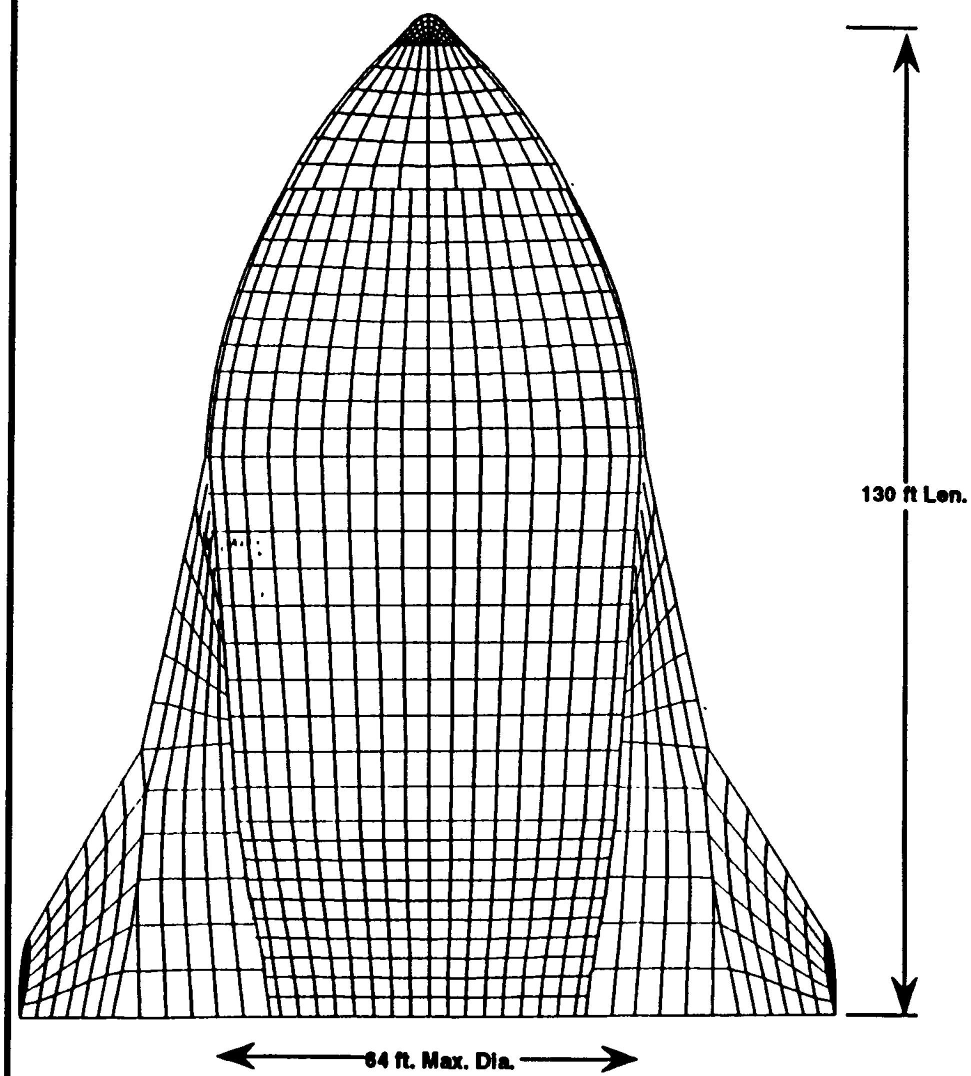
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,939,209 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	259,552 lbf
Vacuum Isp (@100 % RPL):	461.0 sec

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

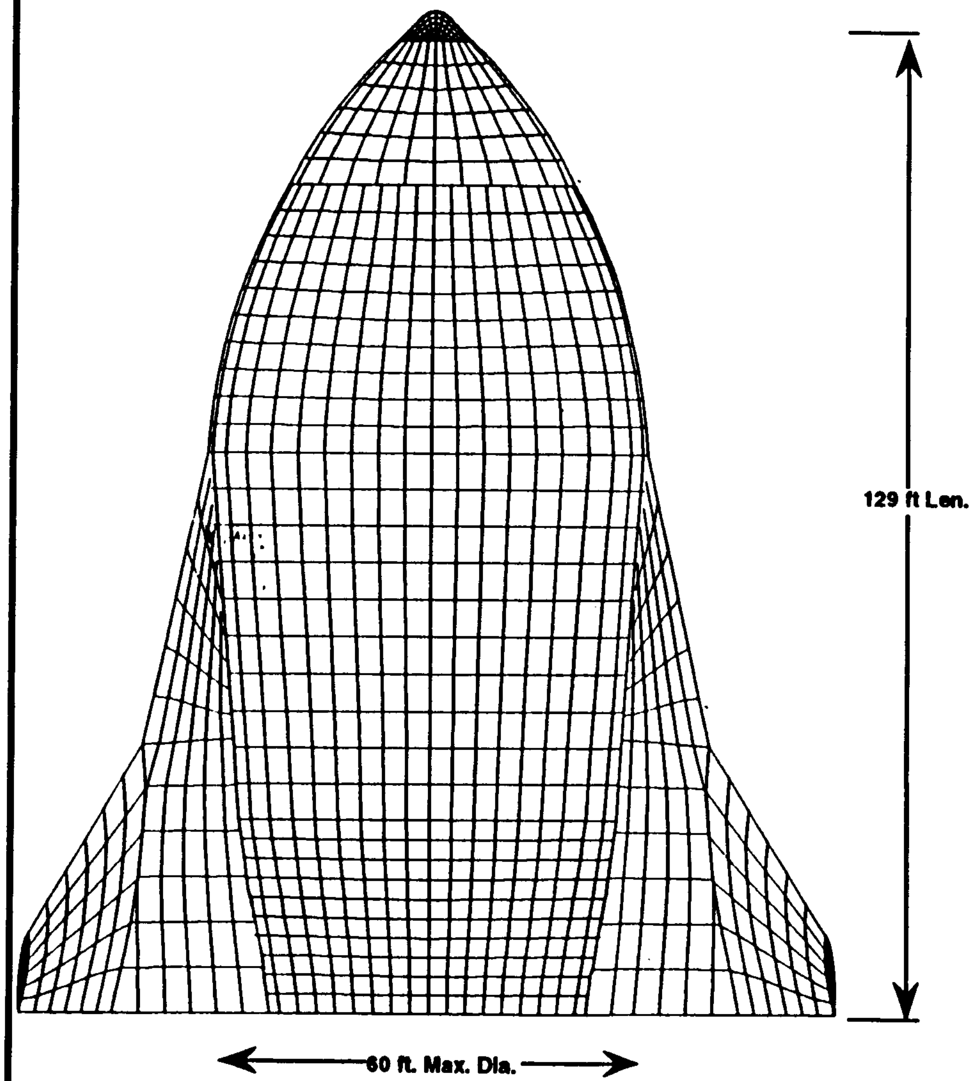
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	2,202,834 lbm
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

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

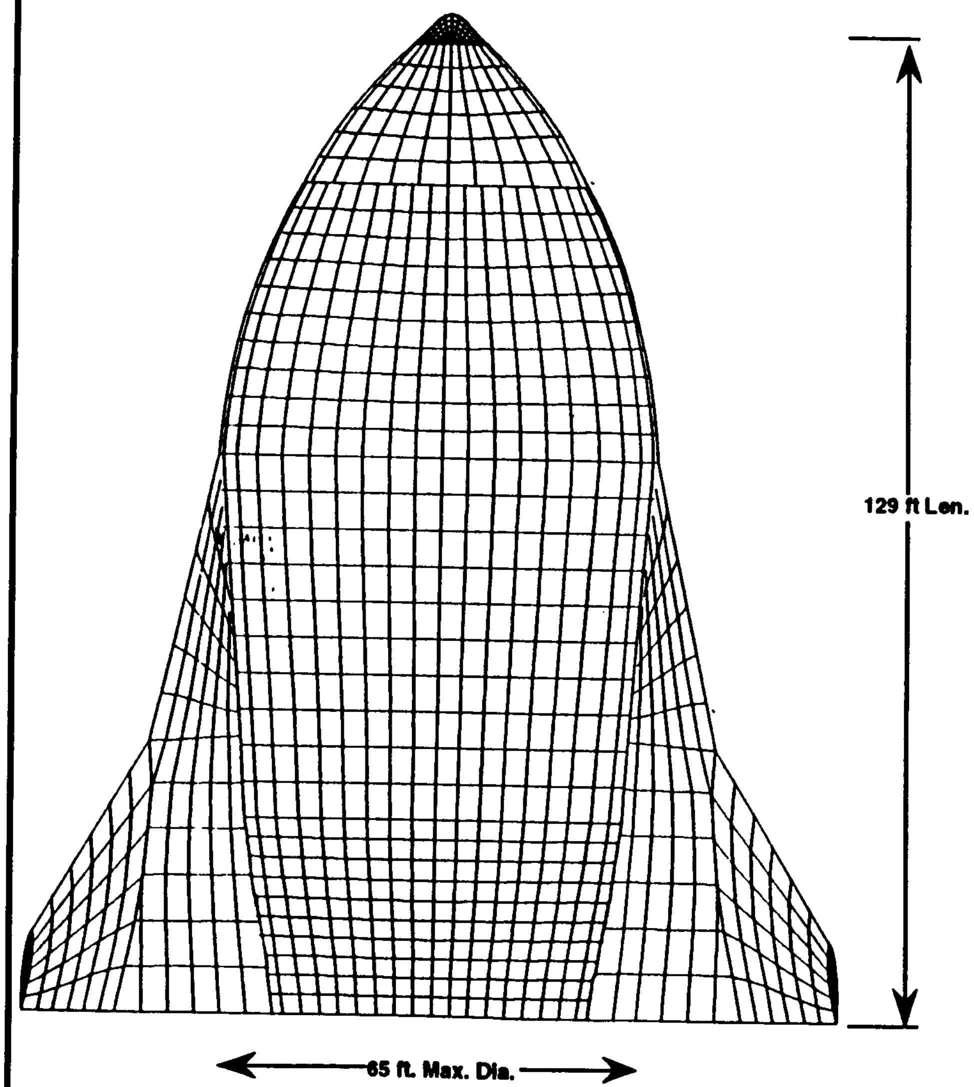
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,962,870 lbm
<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

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

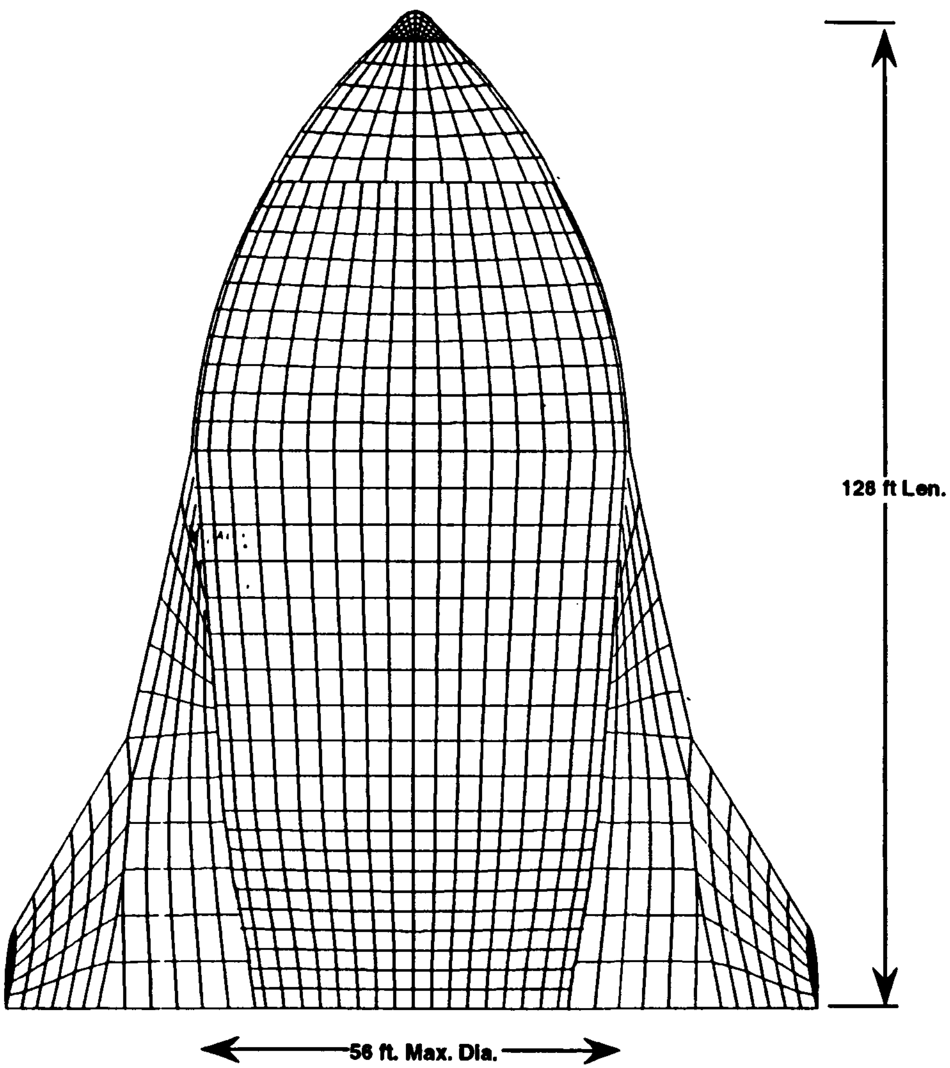
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,741,133 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	239,714 lbf
Vacuum Isp (@100 % RPL):	460.0 sec

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

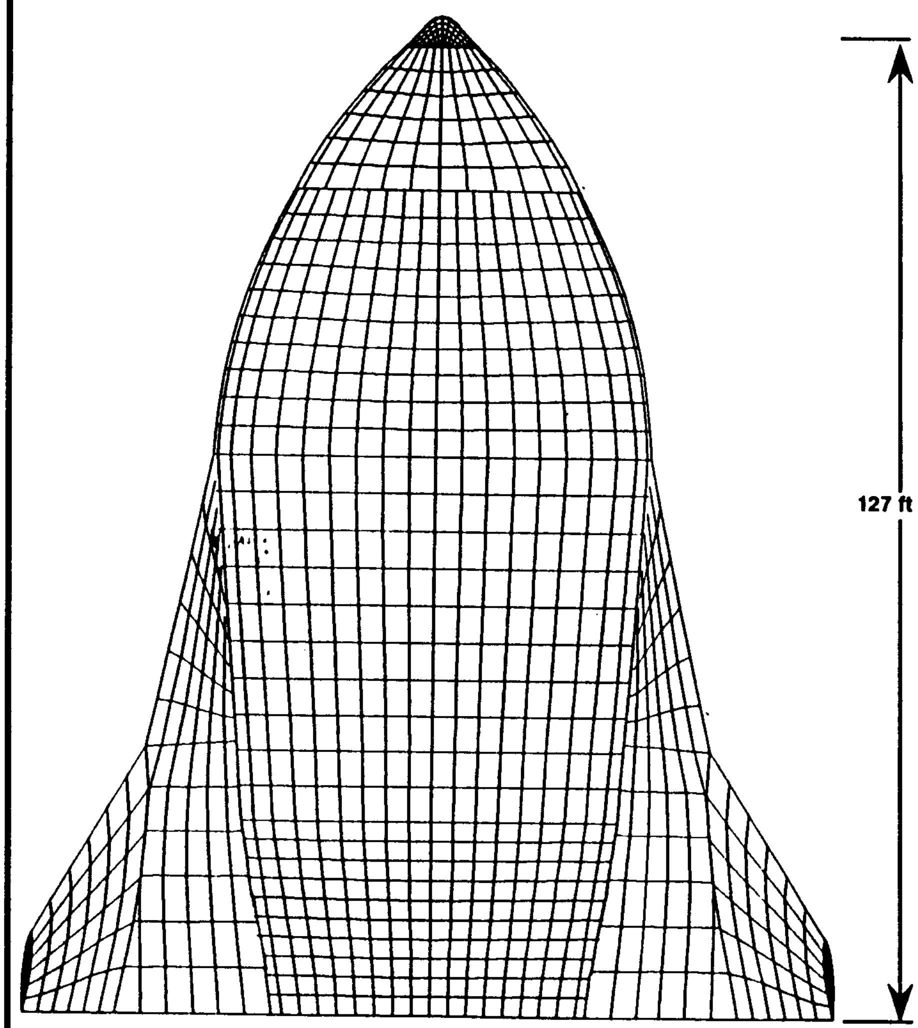
Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,606,145 lbm
Vehicle Specifications:	
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
Mode 1 Propulsion Specifications:	
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
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)	199,347 lbf
Vacuum Isp (@100 % RPL):	460.0 sec

Keith Holden
205-722-4531

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,000 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

GLOW:	1,580,604 lbm
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

Keith Holden
205-722-4531

Study Conclusions

- The winged body VTHL vehicle configuration is the lightest vehicle configuration studied, using the Option 3 technologies
  - Based on comparing the configuration dry masses of the Option 3 versions of the evolved SSME and the RD-701 engine
- The side entry conical VTOL vehicle configuration will be the lightest vehicle configuration if the landing maneuver velocity requirement drops below 900 ft/sec
- The lifting body VTHL vehicle configuration, as defined in this study, is the heaviest configuration
  - This configuration does not show as much improvement in vehicle dry mass for use of high thrust-to-weight, high performance engines as the side entry conical VTOL configuration
  - There are more possible design solutions, allowing more flexibility in reducing vehicle dry mass, than for the other two configurations
- The lifting body configuration dry mass will more favorably compare with the winged body configuration if a horizontal payload bay is used on the winged body configuration

Keith Holden
205-722-4531

Study Conclusions
(Concluded)

- The Option 3 version of the evolved SSME is not a satisfactory engine with respect to the vehicle dry mass as a design metric
- Plug nozzle engines gave the lowest dry mass vehicle configurations
- The dry mass payoff of the tripropellant concept is a function of both the vehicle type and the engine used
- The dry mass payoff of using high thrust-to-weight, high performance engines is different for the side entry conical VTOL configuration and the lifting body VTHL configuration
  - It is likely that the winged body VTHL configuration will also have a different dry mass payoff from using high thrust-to-weight, high performance engines

Keith Holden
205-722-4531

Simulation Results

- Tool Description
- VTVL Concept Results
- VTHL Wing-Body Concept Results
- VTHL Lifting Body Concept Results

K. D. Sagis
205-722-4532

Simulation Results

Tool Description

- Simulation and Optimization of Rocket Trajectory Program (SORT) was used for ascent and entry simulations

Ascent Groundrules

- KSC launch
- KSC atmosphere/no winds
- 50 x 100 nautical mile MECO
- 51.6° orbital inclination
- Maximum acceleration of 3 Gs
- 1% Delta V reserves for flight performance reserves

Kevin D. Sagis
(205) 722-4532

Simulation Results

Ascent Groundrules (concluded)

- 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

Kevin D. Sagis
(205) 722-4532

Simulation Results

Entry Groundrules

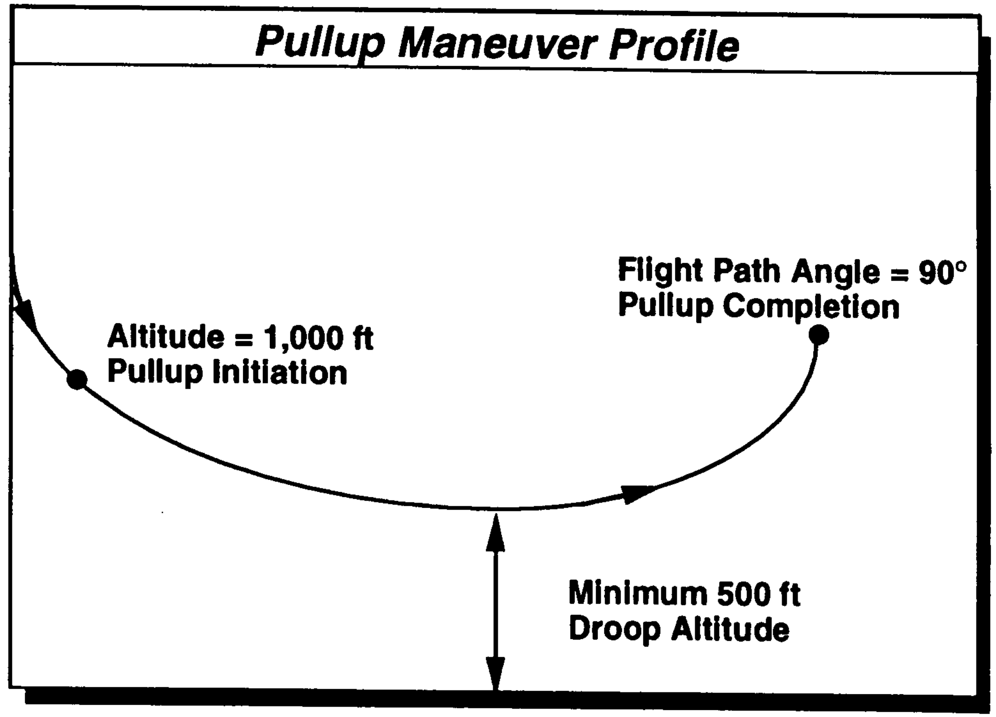
- 1962 U.S. standard atmosphere (non-site-specific)
- Maximum lift trajectory (minimum heat rate) transitioning into maximum range trajectory
- Bank angle and angle of attack used for heat rate modulation for the winged & lifting body configurations and conical configurations, respectively
- Bank angle and sideslip angle used for cross range capability for the winged & lifting body configurations 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

Kevin D. Sagis
(205) 722-4532

Simulation Results

Entry Groundrules (concluded)

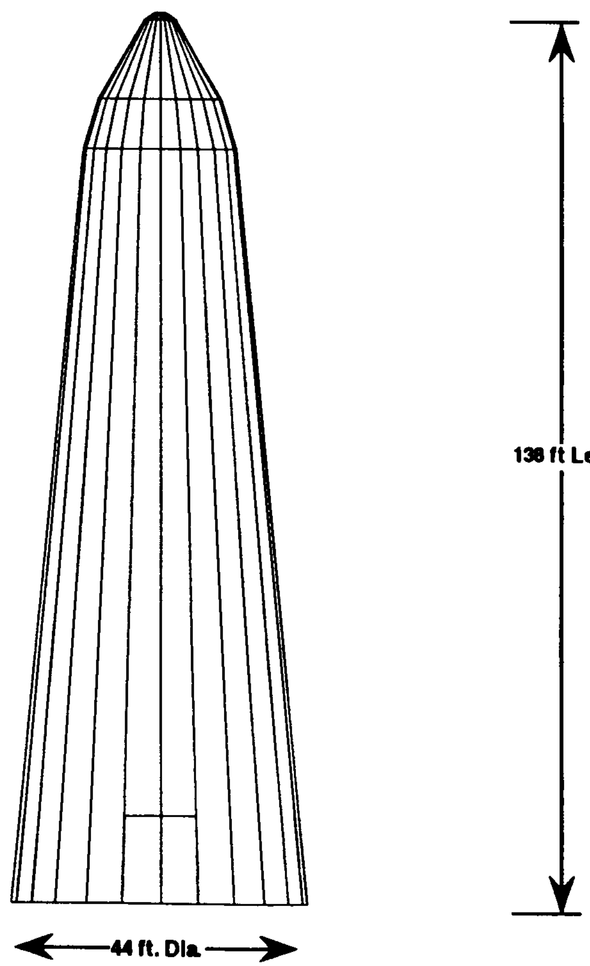
- Conical landing maneuver initiated at 1,000 ft altitude with a 500 ft minimum droop altitude
- 2 G constant normal load during propulsive pullup maneuver to 90 degree flight path angle
- Strive for minimum H-dot value and minimum safe altitude at end of pullup maneuver to minimize propellant requirements



Kevin D. Sagis
(205) 722-4532

VTVL SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

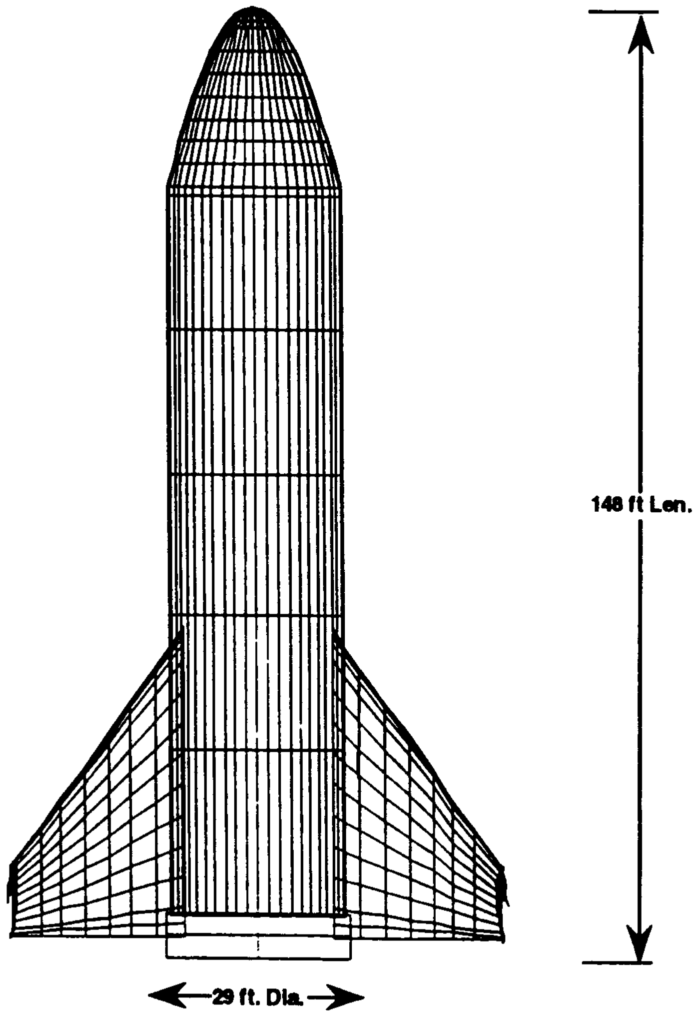
Payload:	24,800 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

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 Propulsion 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):	452.7sec

K. D. Sagis
205-722-4532

VTHL Wing-Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,300 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

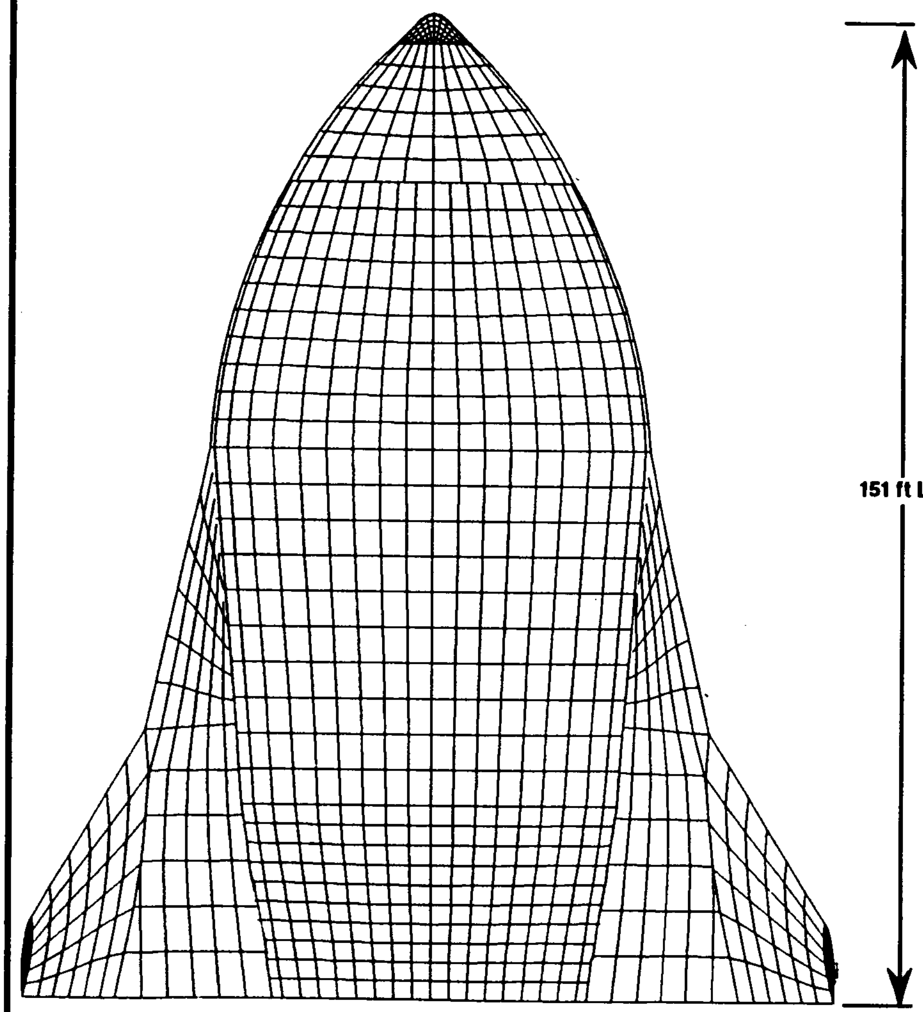
GLOW:	1,857,714 lbm
Vehicle Specifications:	
Vehicle Dry Mass @ Liftoff:	148,884 lbm
Usable Propellant Mass (including FPR):	
--Mode 1	1,061,829 lbm
--Mode 2	596,618 lbm
Propellant Combination:	
--Mode 1	LOX/LH2/Kerosene
--Mode 2	LOX/LH2
Ascent Residuals	8,790 lbm
OMS & RCS Propellant	16,593 lbm
Landing Propellant	N/A
Landing Specific Impulse	N/A
Main Engine Type/No.:	RD-701A/6
Mode 1 Propulsion 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):	452.7sec

Lockheed

K. D. Sagis
205-722-4532

VTHL Lifting Body SSTO Concept Summary

Preliminary Concept



Vehicle Not Drawn to Scale

Payload:	25,200 lbm
Final Position:	220 nm Circ. Orbit, i= 51.6 deg

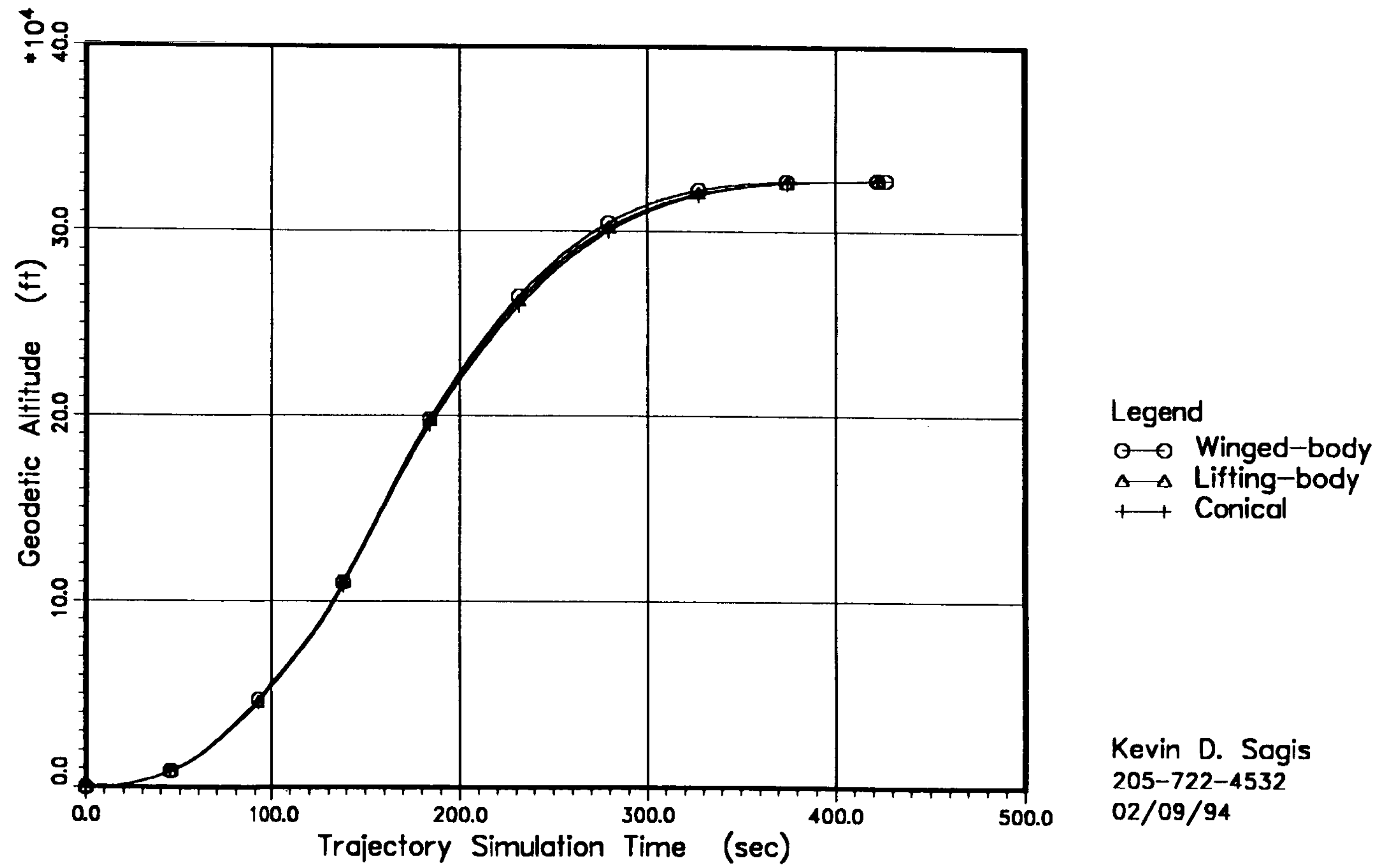
GLOW:	2,439,123 lbm
Vehicle Specifications:	
Vehicle Dry Mass @ Liftoff:	209,569 lbm
Usable Propellant Mass (including FPR):	
--Mode 1	1,394,274 lbm
--Mode 2	776,386 lbm
Propellant Combination:	
--Mode 1	LOX/LH2/Kerosene
--Mode 2	LOX/LH2
Ascent Residuals	11,511 lbm
OMS & RCS Propellant	22,384 lbm
Landing Propellant	N/A
Landing Specific Impulse	N/A
Main Engine Type/No.:	RD-701A/5
Mode 1 Propulsion 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):	452.7sec

Lockheed

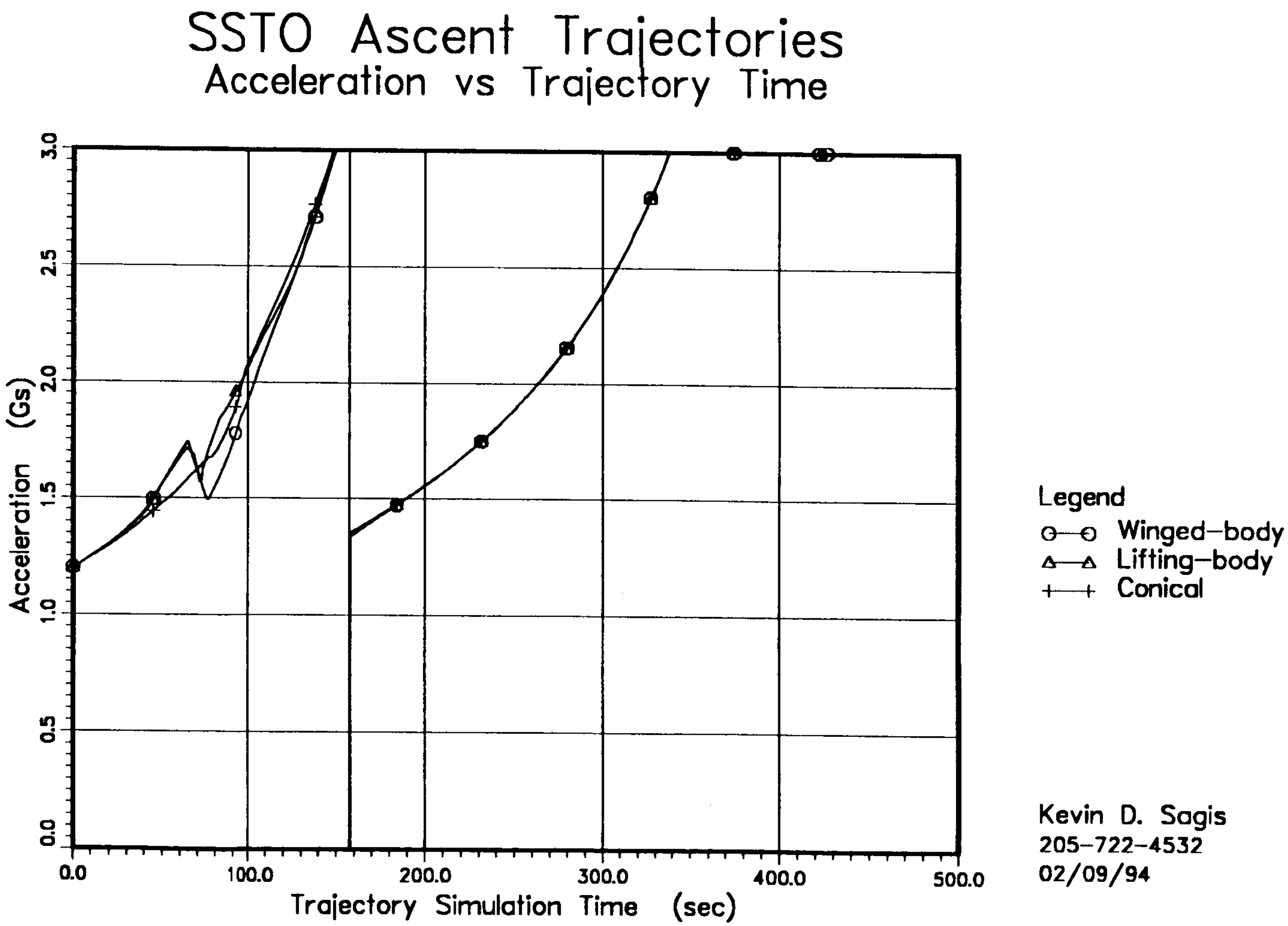
K. D. Sagis
205-722-4532

Simulation Results

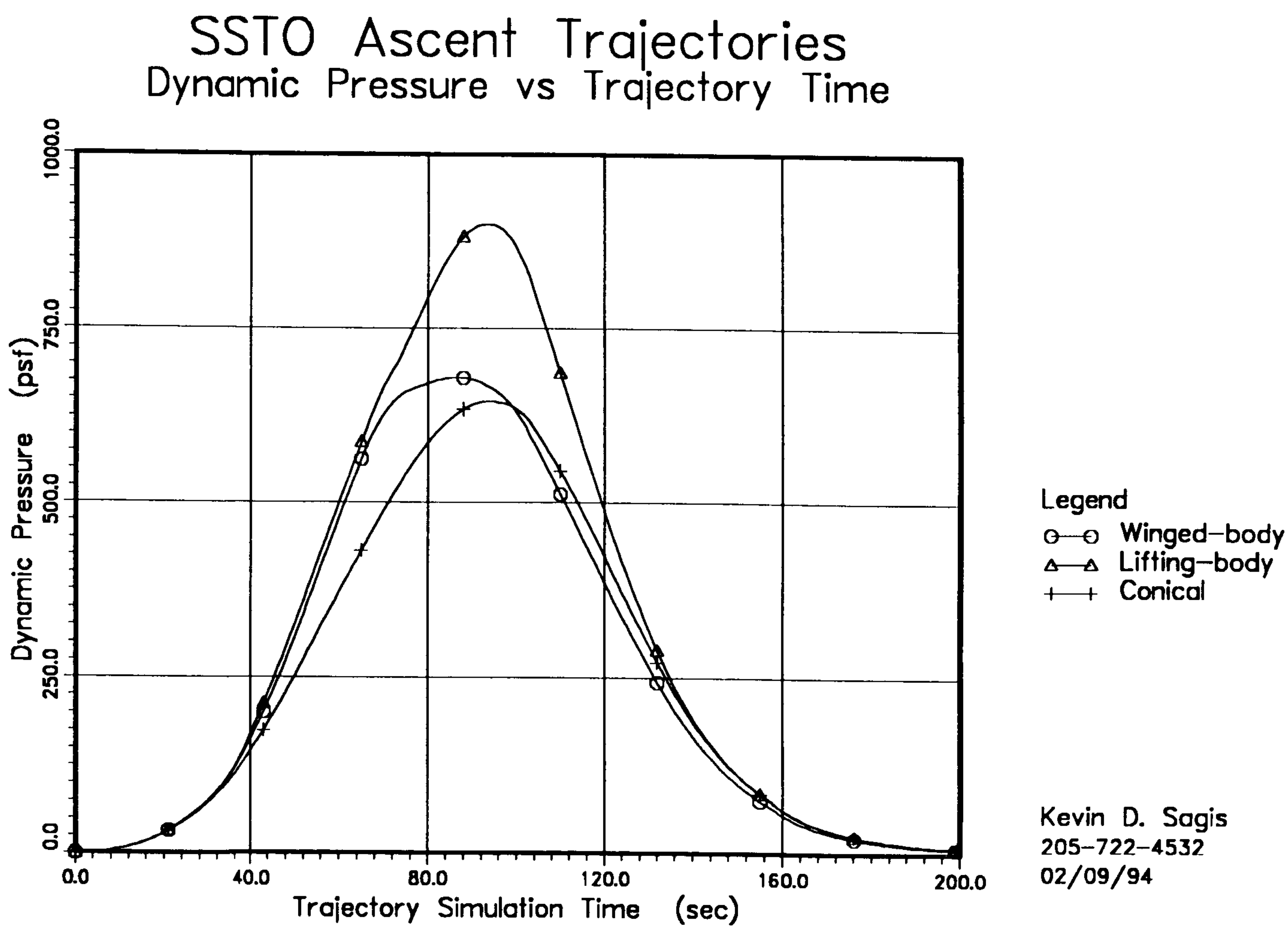
SSTO Ascent Trajectories
Geodetic Altitude vs Trajectory Time



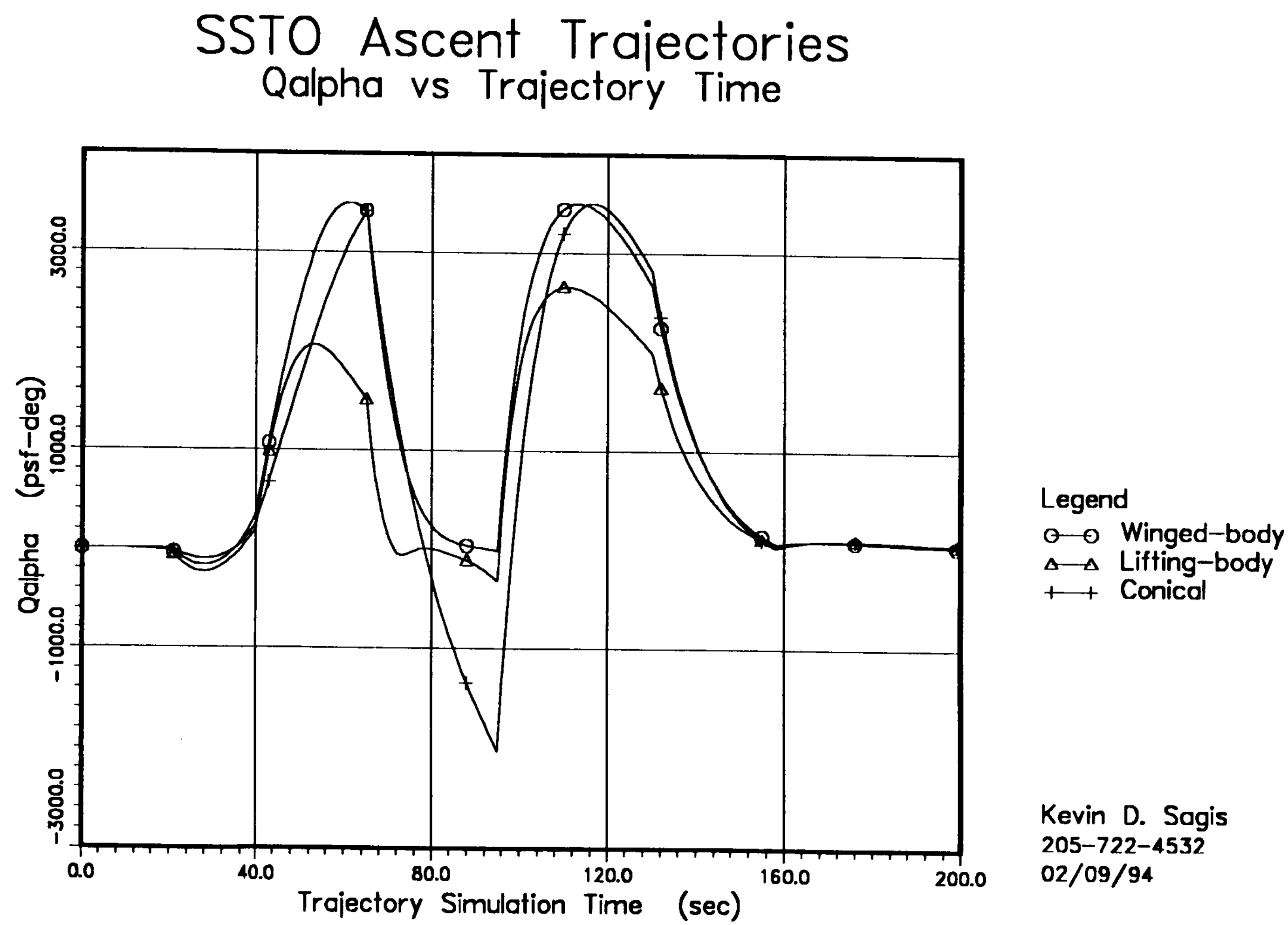
Simulation Results



Simulation Results

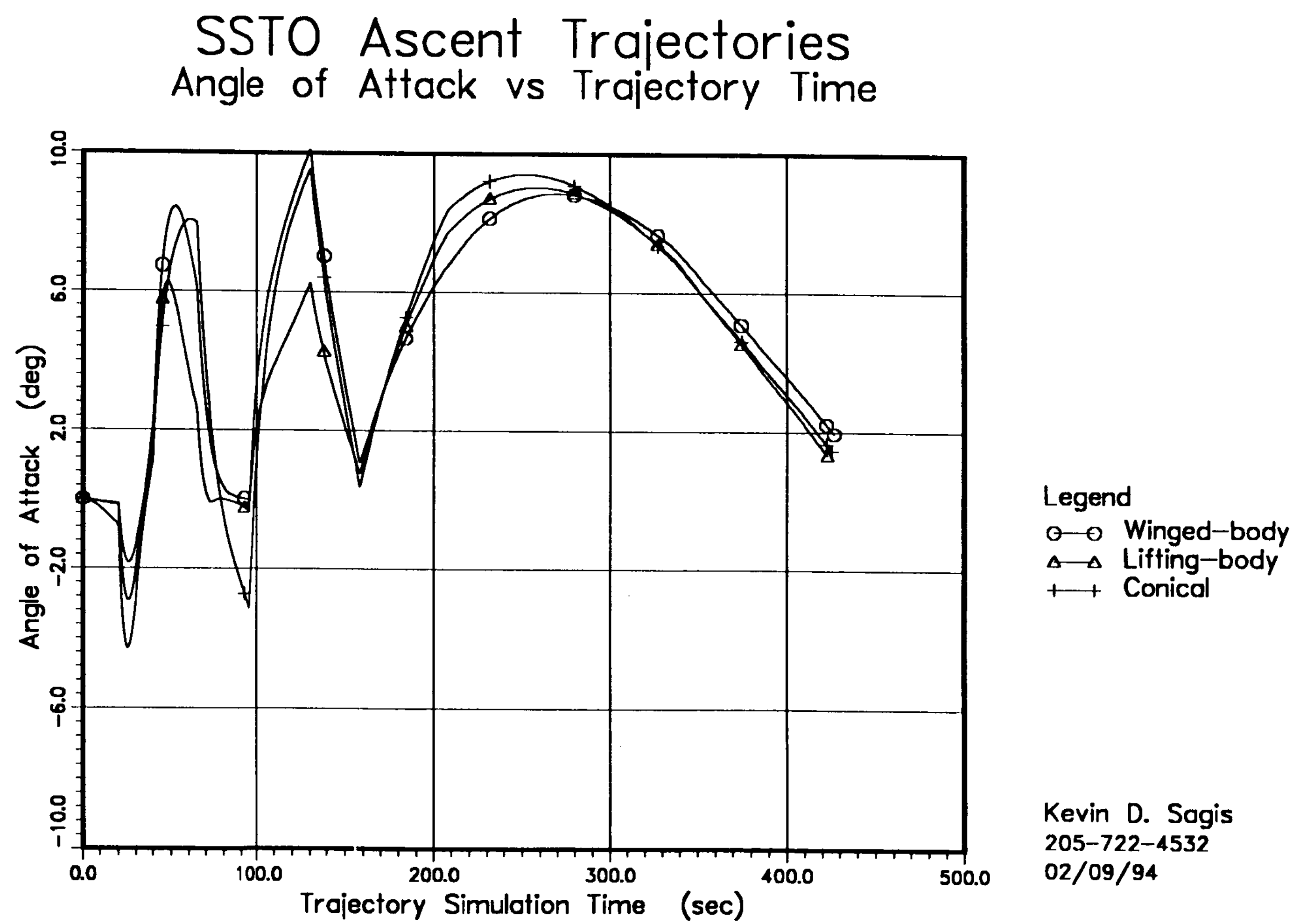


Simulation Results



Kevin D. Sagis
205-722-4532
02/09/94

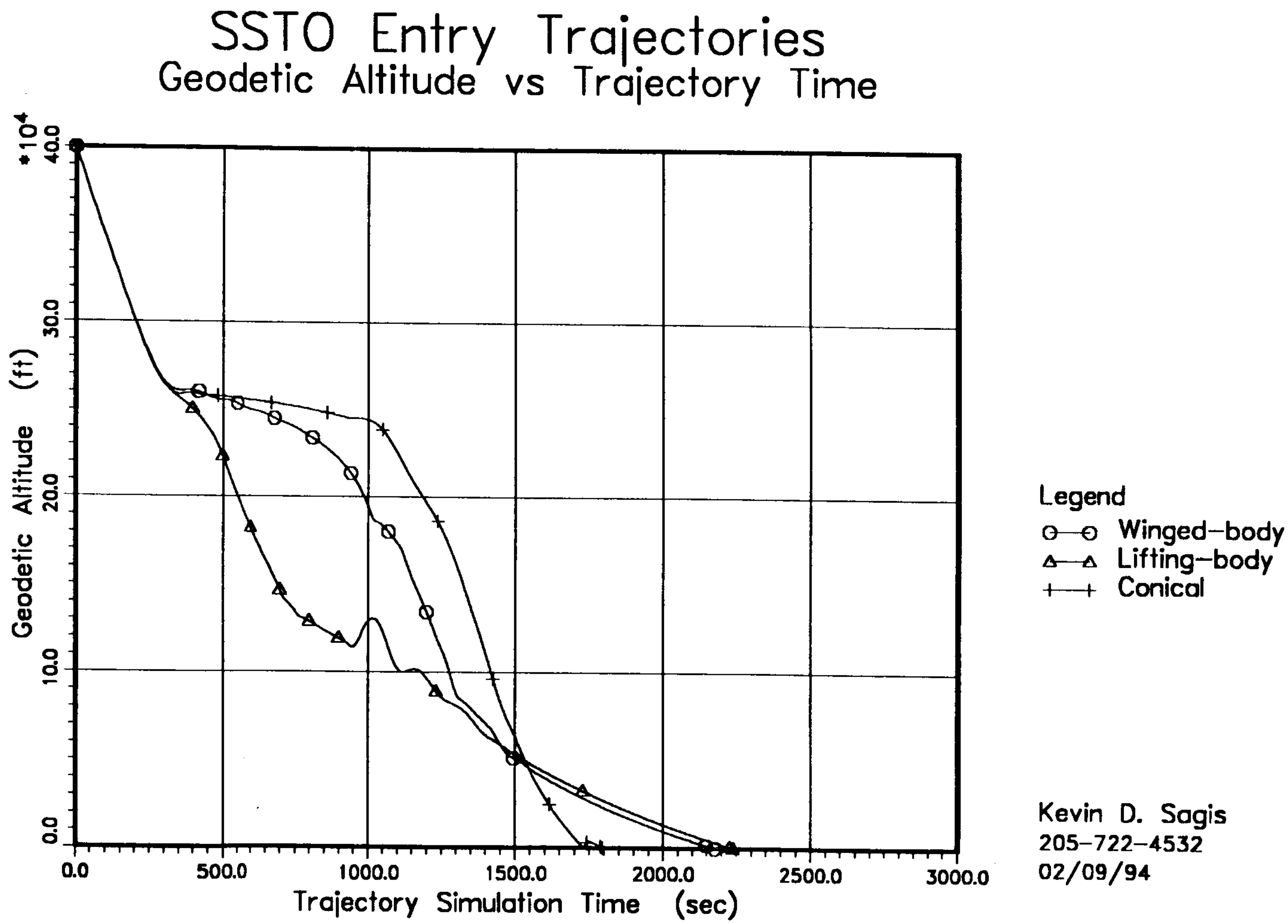
Simulation Results



Kevin D. Sagis
205-722-4532
02/09/94

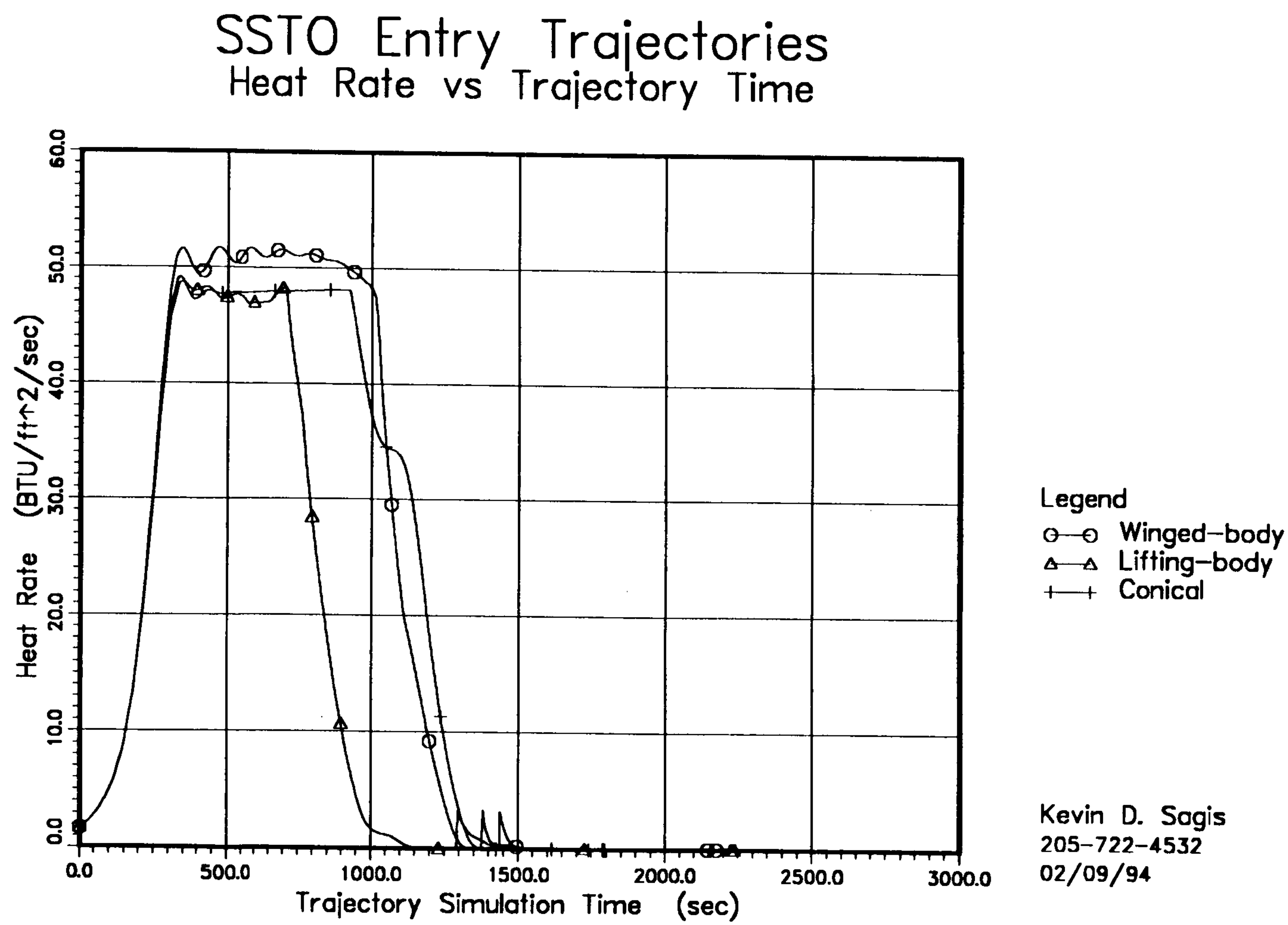
Lockheed

Simulation Results

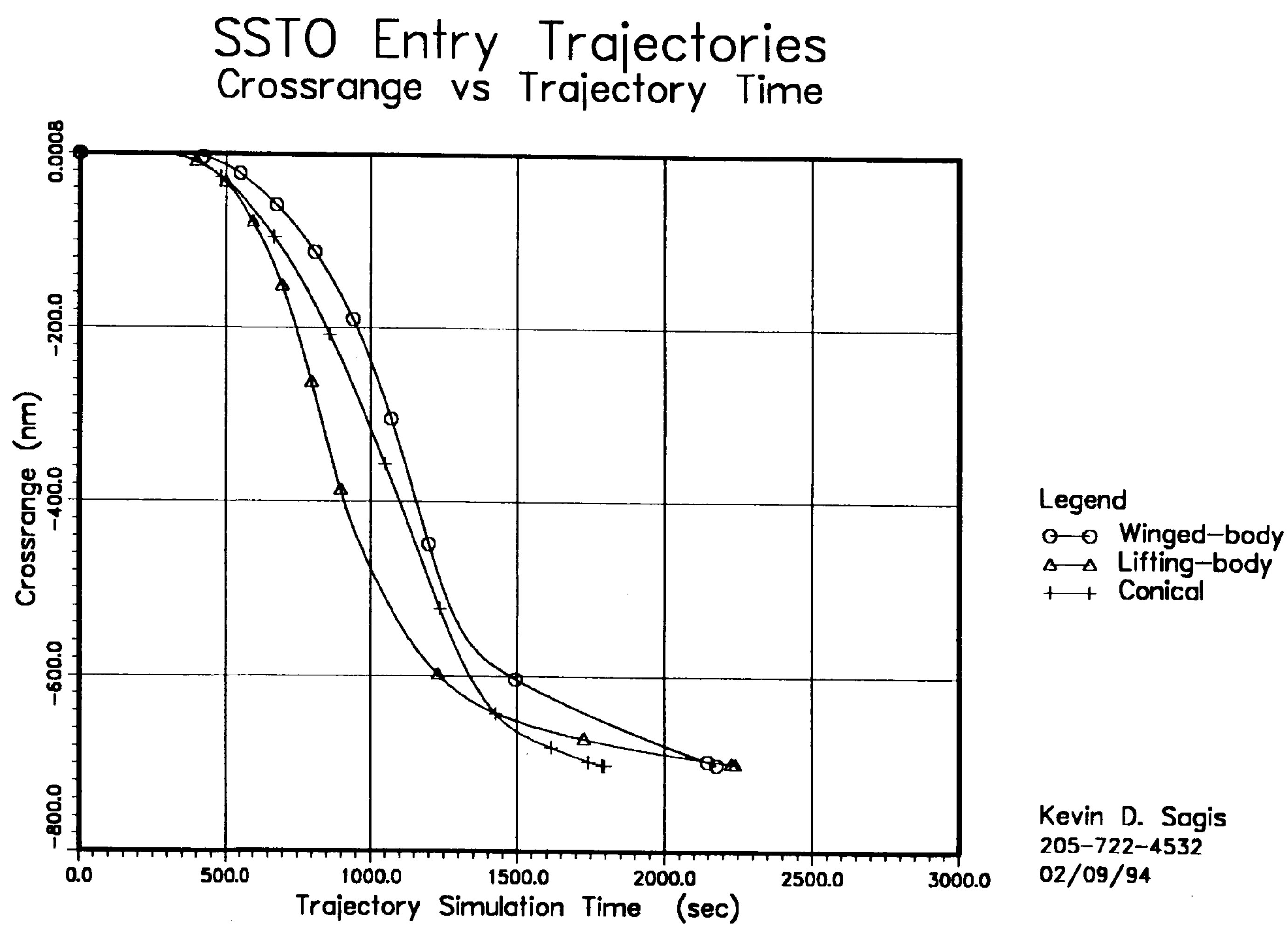


Kevin D. Sagis
205-722-4532
02/09/94

Simulation Results

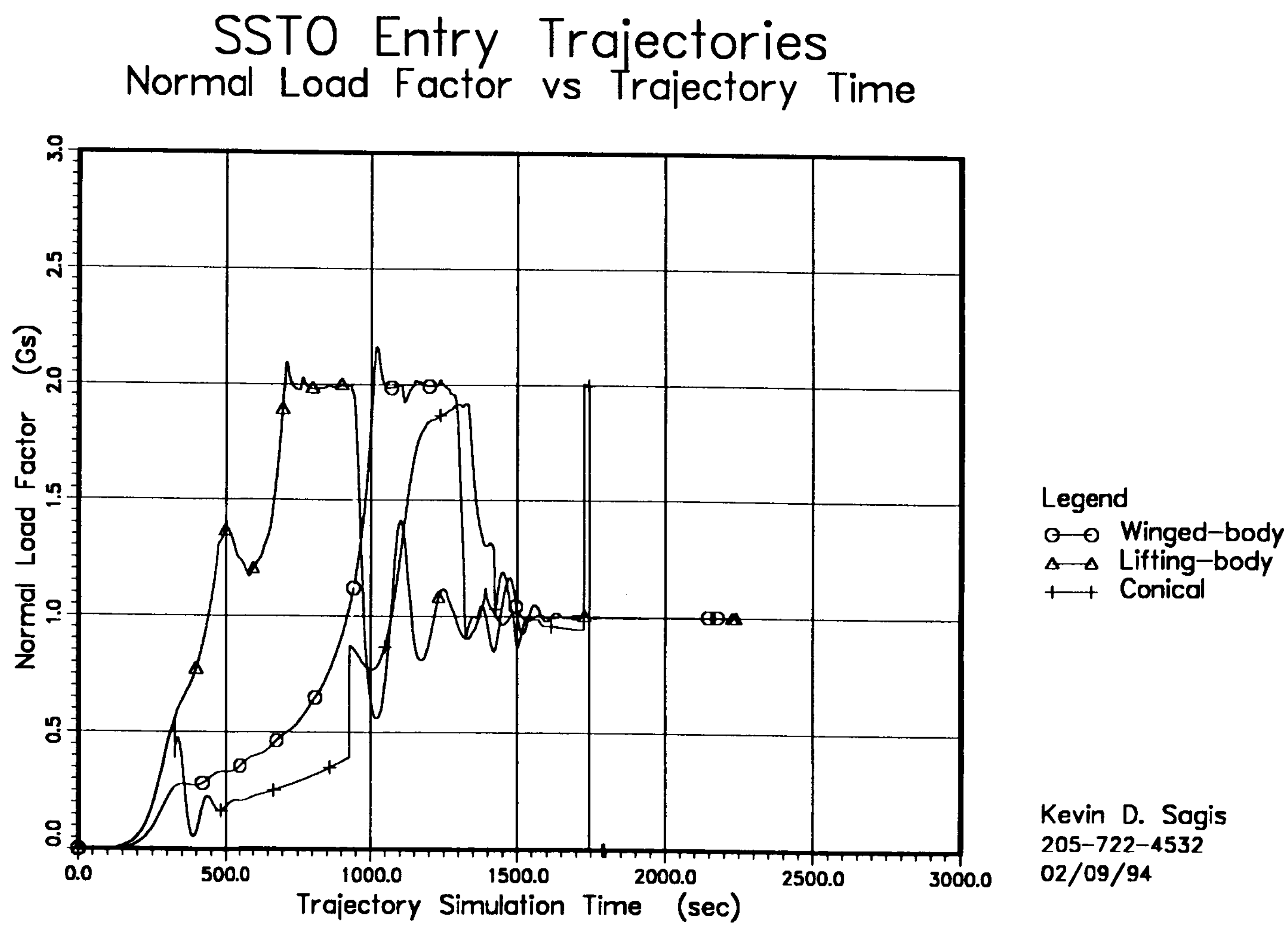


Simulation Results



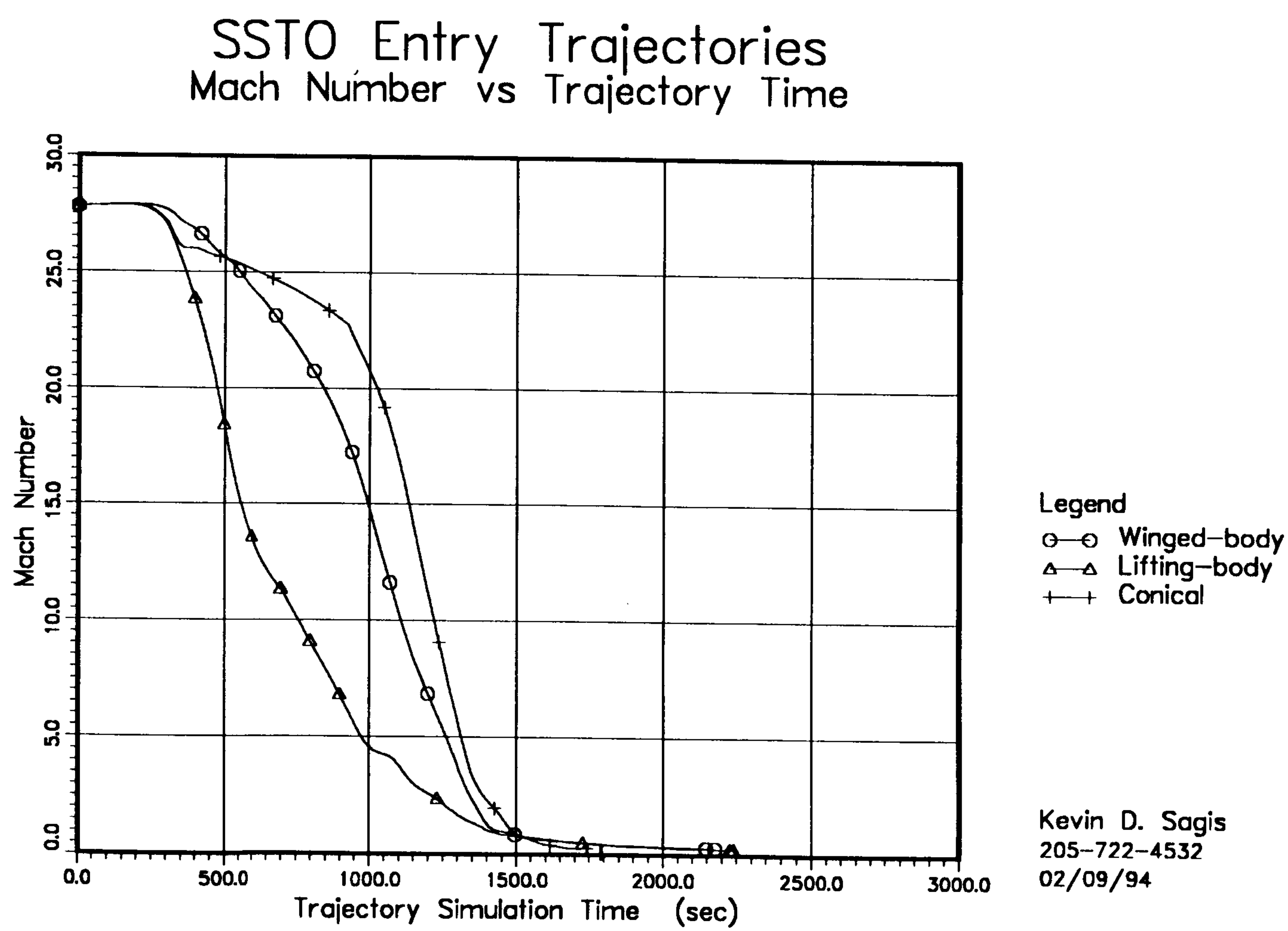
Kevin D. Sagis
205-722-4532
02/09/94

Simulation Results

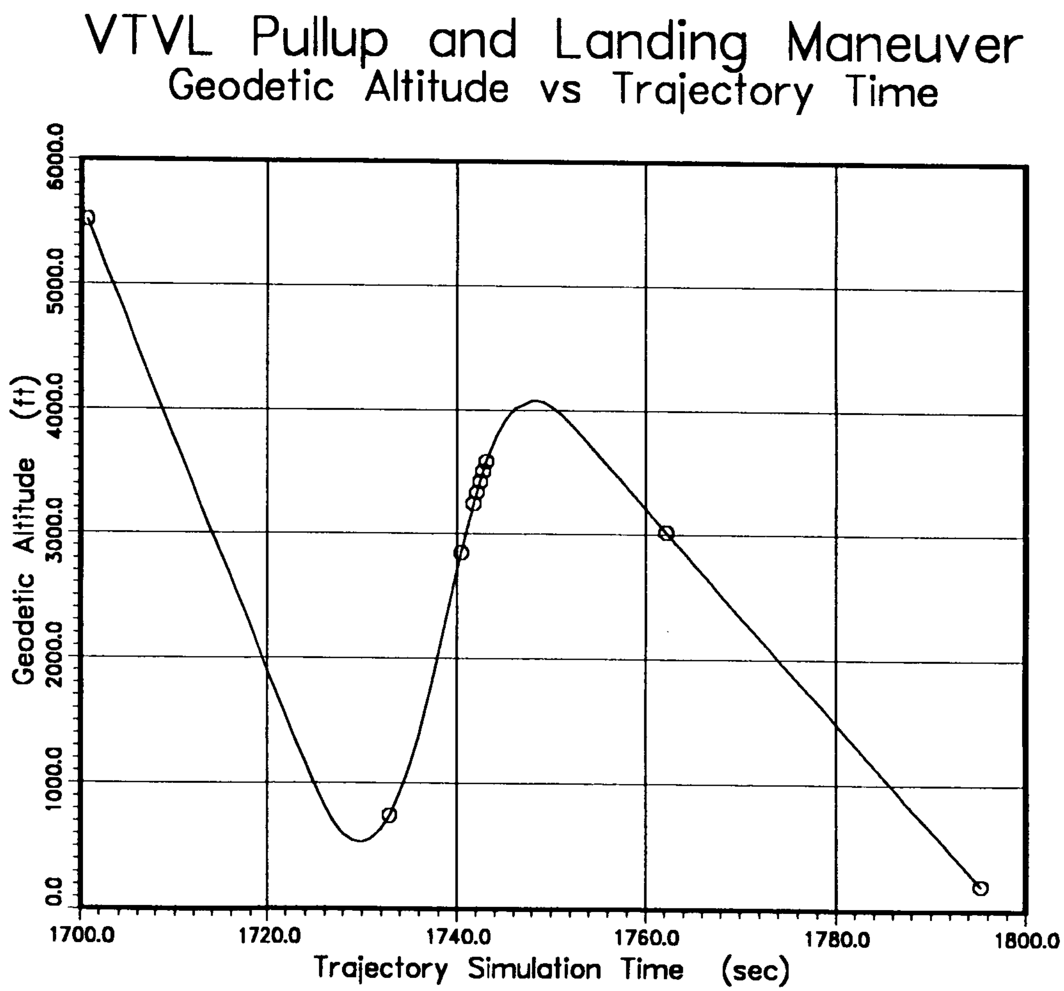


Kevin D. Sagis
205–722–4532
02/09/94

Simulation Results



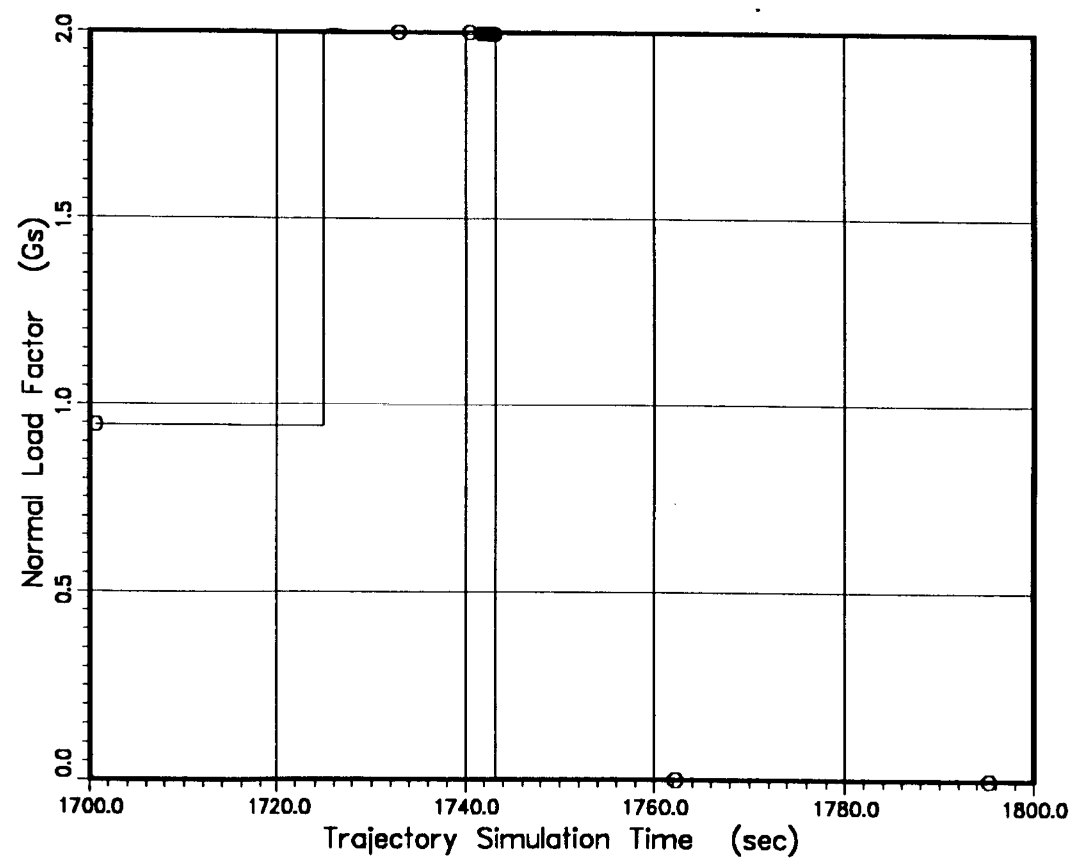
Simulation Results



Kevin D. Sagis
205–722–4532
02/09/94

Simulation Results

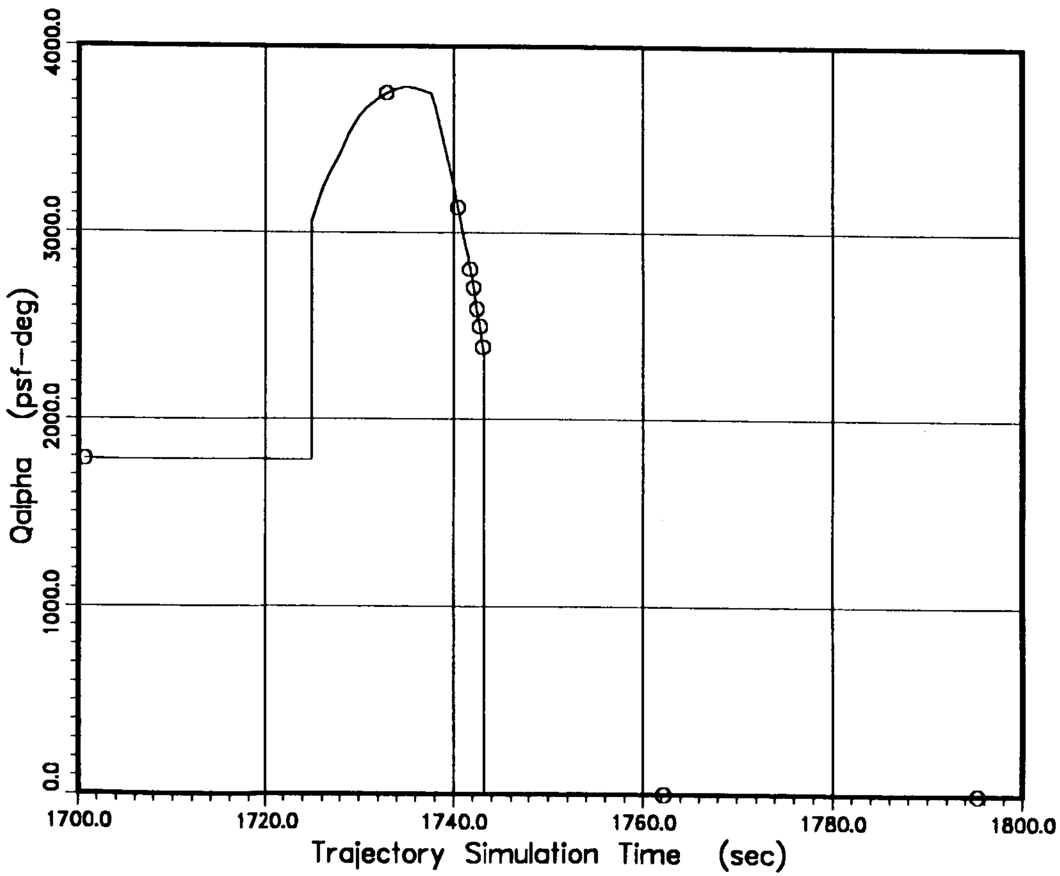
VTVL Pullup and Landing Maneuver
Normal Load Factor vs Trajectory Time



Kevin D. Sagis
205-722-4532
02/09/94

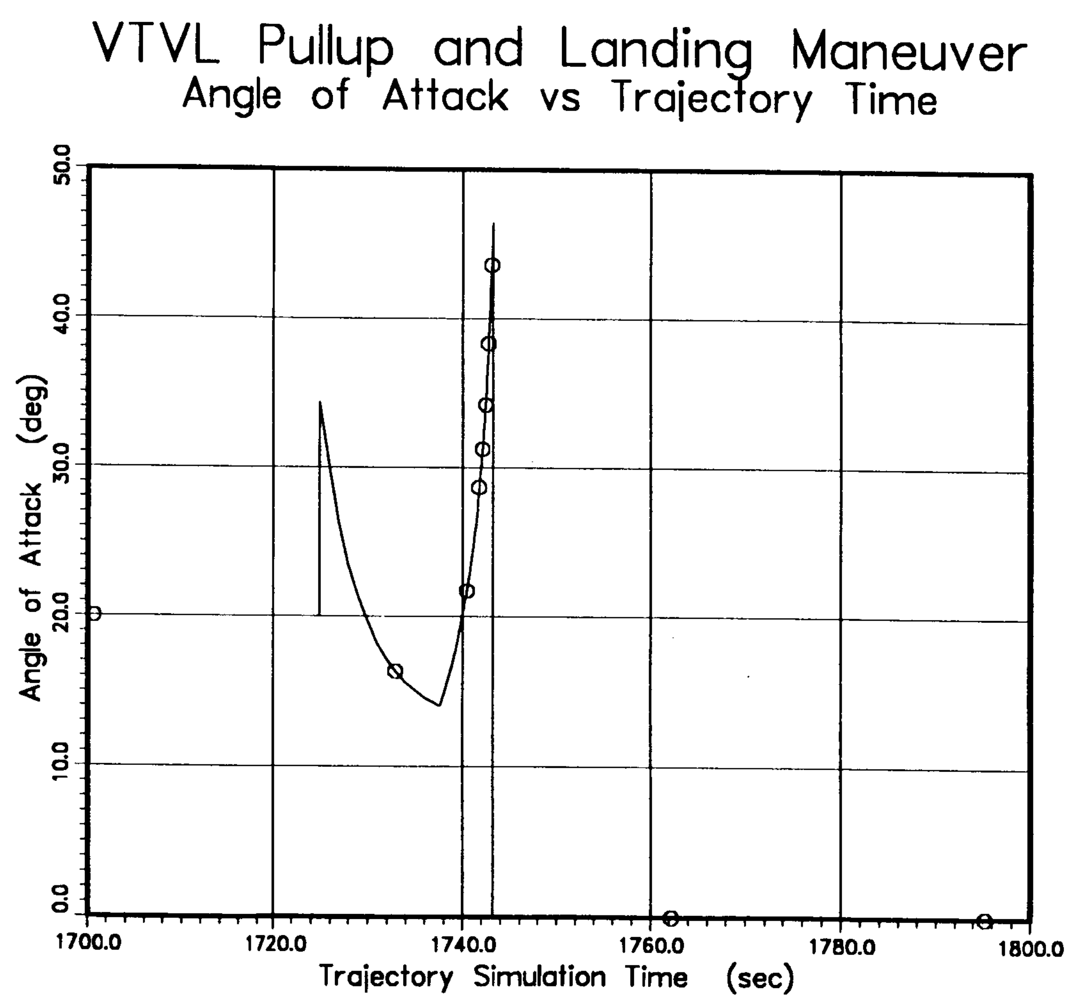
Simulation Results

VTVL Pullup and Landing Maneuver
Qalpha vs Trajectory Time



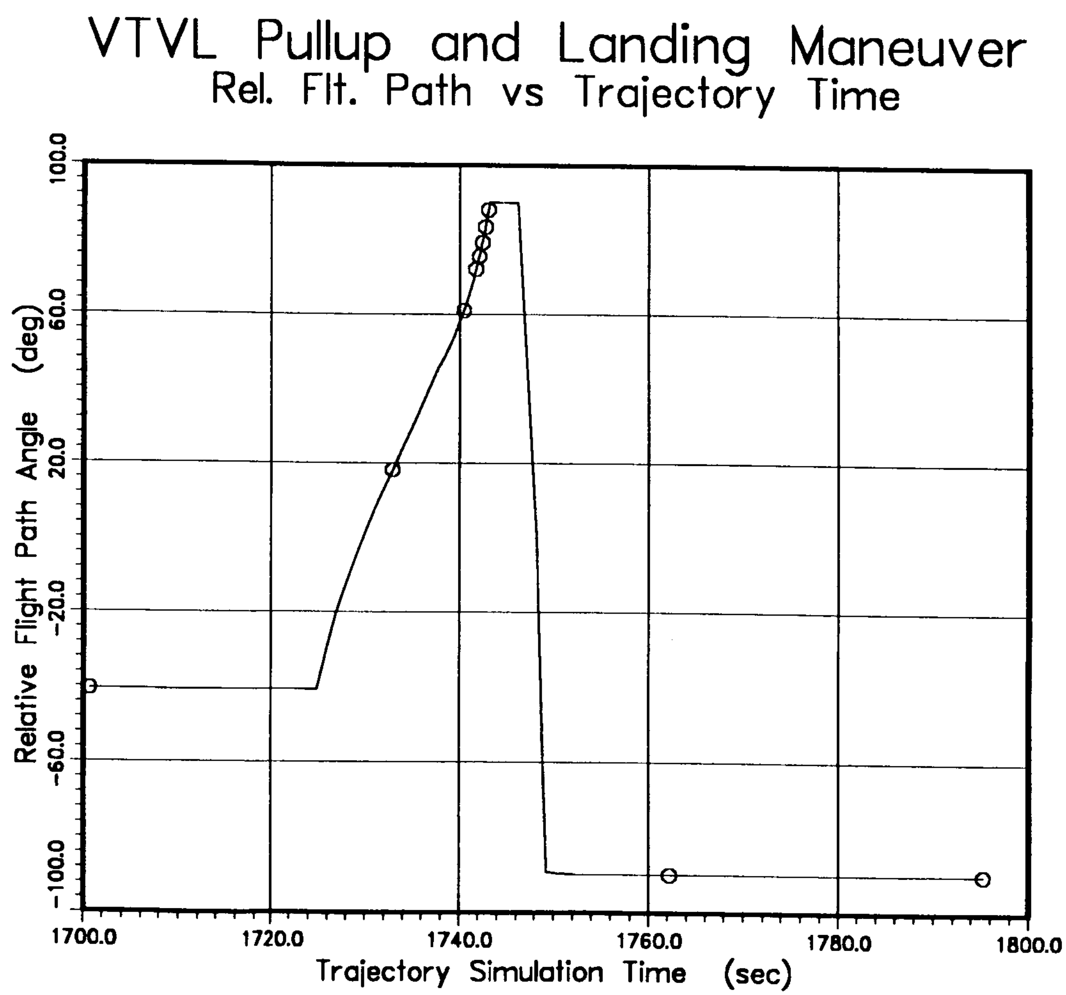
Kevin D. Sagis
205-722-4532
02/09/94

Simulation Results



Kevin D. Sagis
205-722-4532
02/09/94

Simulation Results



Kevin D. Sagis
205-722-4532
02/09/94

Lockheed

Simulation Results

Ascent Trajectory Conclusions

- All three vehicles exhibit similar profiles due to similar sizing assumptions such as T/W ratios and specific impulses
- Lifting body experiences higher qbar due to favorable lift-to-drag ratios at low angles of attack resulting in a lifting trajectory
- Further first stage optimization could smooth the Qalpha profile for all three vehicles

Entry Trajectory Conclusions

- All three vehicles exhibit similar maximum heat rates (approximately 50 BTU/ft^2/sec)
- The lifting body VTHL experiences much smaller total heat load

Kevin D. Sagis
(205) 722-4532

Simulation Results

Entry Trajectory Conclusions

- The VTVL pullup maneuver is feasible with pure thrust control, but a more optimal solution would include control surface deflection contributions

Kevin D. Sagis
(205) 722-4532

Technology Requirements Summary

J. B. McCurry
205-722-4509

Technology Requirements Summary

- Recall that SSTO cost effectiveness can be measured as:

  (infrastructure, W.O.D.B.) (materials, manufacturing)

  Mission Cost = recurring fixed cost + cost due to size

  (complexity, integr., degree of reuse, refurb., test & checkout, etc.)

  + cost due to technology/design + DDT&E amortization

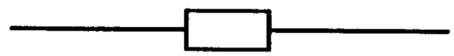
  + . . .

- A program level philosophy is first established to bound the solution set for affecting "cost due to technology/design"
  - Intact abort protection important to protect the investment of a small fleet of relatively expensive (unit cost) SSTO vehicles
  - Launch probability and mission success are important keys to attracting payload customers
  - Short mission cycle time is an important key to attracting payload customers
  - If achieve "routine" cheap turnaround of the vehicle between flights, have intact abort capability, and have ability for rapid changeout of failed LRUs, then mission success not as vital to payload customer if price to achieve high success is great

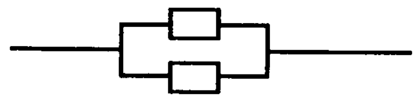
J. B. McCurry
205-722-4509

Technology Requirements Summary (Continued)

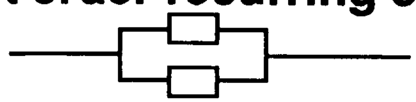
- One candidate program philosophy is the following:
  - Minimize number of Criticality (Crit) 1 items on the vehicle



  - Have dual redundancy of Crit 1 items (that can reasonably be made redundant) to achieve an intact abort at all times; i.e., Crit 1R/2



  - Have dual redundancy of Crit 2 items to achieve mission success; i.e., Crit 2R/2 with priority and choice of redundancy (multiple path or physical) based on first order recurring cost drivers



  - Have no redundancy on Crit 3 items but ensure design as accessible LRUs

- A development program should be used to characterize the operating envelop and associated characteristics of the integrated vehicle
  - Emphasis on developing profile of characteristics for Criticality 1 items
  - Develop sufficient confidence in MTBF to lead to "one-time certification"
  - Shift focus of Crit 2R/2 and Crit 3 subsystem development from MTBF determination to low-cost changeout

J. B. McCurry
205-722-4509

Technology Requirements Summary (Concluded)

- Technology development program should strive for development and demonstration of highly modular subsystems
  - Quick/easy LRU changeout without complex ground support equipment
  - Balance demonstration of MTBF with design for "graceful failure" paths
- Because of SSTO subsystem function commonality with Shuttle subsystems, use advanced development and technology requirements section of Option 1 final report as a first roadmap
  - The Shuttle evolution technologies were independent of advanced vehicle development decisions

J. B. McCurry
205-722-4509

Conclusions

J. B. McCurry
205-722-4509

Conclusions

Sizing Tools

- Second order fidelity (subsystem level) in sizing tool achieved and benchmarked with LaRC Option 3 results
  - Some differences in subsystem sizing assumptions remain to be resolved but have been isolated

- Sizing of three major SSTO configuration types can be accomplished with commonality of groundrules and assumptions
  - Effect of different technology assumptions can be introduced

- A good matrix of propulsion options has been developed for future trade studies

- Completed tool and associated documentation to be provided in March timeframe (TA-2 completion)

J. B. McCurry
205-722-4509

Conclusions (Continued)

General Vehicle Observations

- Vehicle stability during unpowered flight is a first order driver on outer moldline and internal subsystem and structure layout

- Ascent flight mechanics and performance are fundamentally the same for similar thrust-to-weight profiles between the three configurations

- Entry flight mechanics and performance are different between the configurations, similar heat rates, different heat loads, cross range capabilities adequate

VTOL

- Will involve large diameter propellant tanks (>27 ft) raising manufacturability issues
  - May need innovative design concepts such as multi-cell

- Flight mechanics of landing maneuver is complex and affects structure requirements (loads)
  - Optimized entry profile remains to be developed

- Aerospike or "high" number of engines may be required to overcome ascent base drag

J. B. McCurry
205-722-4509

Conclusions (Continued)

Wing-Body VTHL

- Less robust ascent profile due to wing bending/shear/torsion loads coverage
- Requires more high temperature because of more small radii
- Entry heating load slightly better than VTOL but worse than lifting body

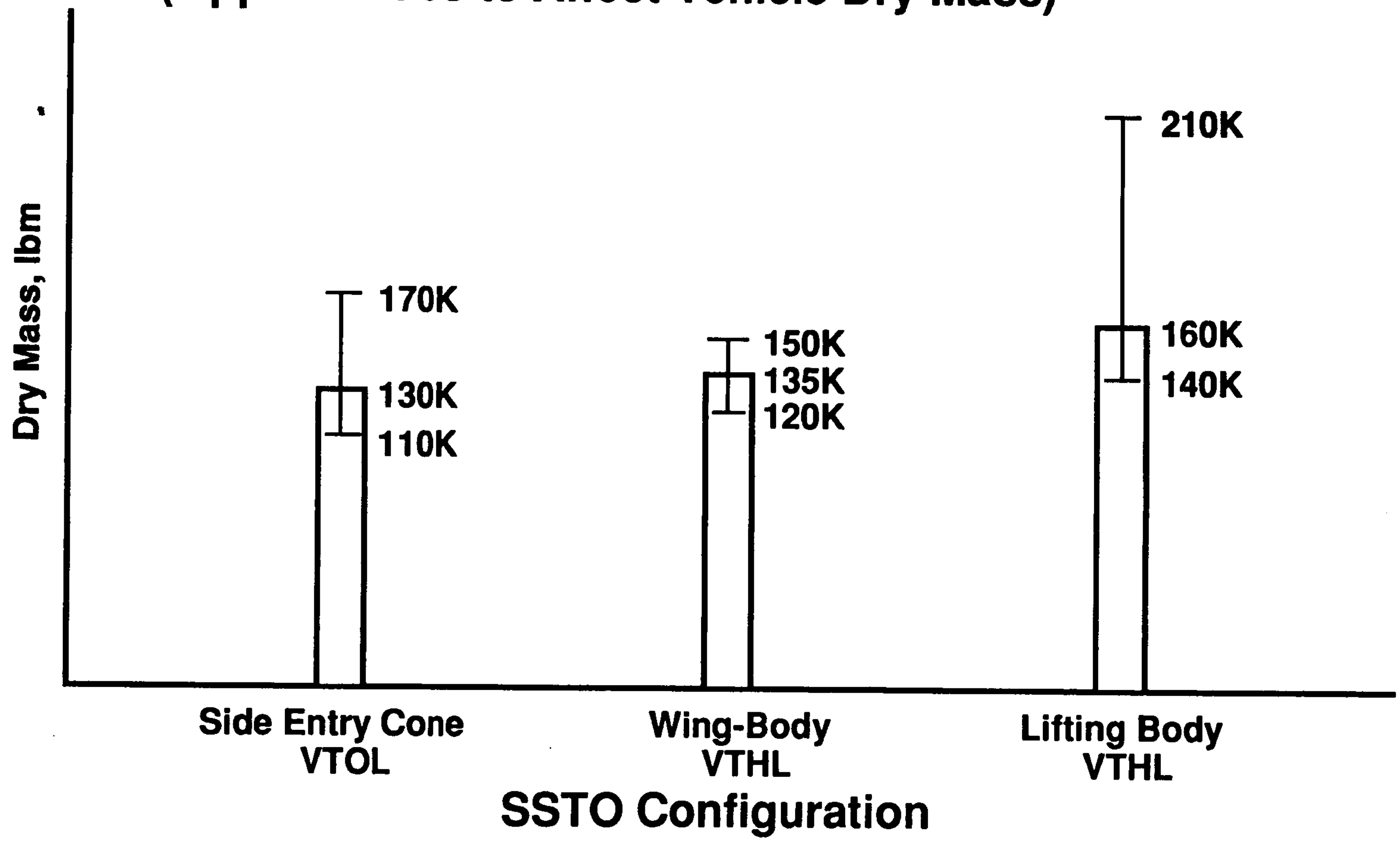
Lifting Body VTHL

- Robust load path and inherent stiffness at price of added dry weight
- Entry heat load significantly better than other two configurations
- Significantly more design options available
- Greater dry weight sensitivity to aerodynamic/stability considerations

J. B. McCurry
205-722-4509

Conclusions (Concluded)

Current Dry Mass Range Due to Configuration Variance (Opportunities to Affect Vehicle Dry Mass)



J. B. McCurry
205-722-4509

13.0 First Lunar Outpost Heavy Lift Launch Vehicle Design and Assessment Report

This section contains a copy of the final report documenting the preliminary results of the First Lunar Outpost (FLO) Heavy Lift Launch Vehicle (HLLV) design activity that was principally performed by personnel from the Marshall Space Flight Center, Johnson Space Center, and Kennedy Space Center during the period of January through May of 1992. The FLO subteam assessed the requirements for, and definition of, HLLV concepts that would perform the single-launch FLO mission. LMSC's TA-2 team also contributed HLLV design results to the report, and the TA-2 study manager performed the editing and production of the report at the request of Gene Austin of the Marshall Space Flight Center.

14.0 Russian Propulsion Technology Assessment Reports

This section provides a list of the contract deliverables that Pratt & Whitney produced under subcontract to LMSC's TA-2 contract that assessed the technologies and performance of NPO Energomash's RD-170 and RD-180 LOX/kerosene main engines, as well as the conceptual tripropellant (LOX/LH_2/kerosene) engine RD-701/RD-704. Energomash was under subcontract to Pratt to provide the engine technology and performance data. Pratt's subcontract consisted of a basic three-month effort from March through May of 1992 to provide preliminary performance data on the RD-170 engine. Two additional contract amendments, Amendment A and B, were funded by MSFC and the Ballistic Missile Defense Organization, respectively, to provide additional details on performance, technologies, and production costs for the RD-170 and RD-180 (Amendment A), and the RD-701 (Amendment B).

During the Heavy Lift Launch Vehicle (HLLV) studies of 1993, it became clear that there was a very limited number of candidate domestic main propulsion elements that could be used or developed. The result was to also consider the use of Russian main propulsion elements. An additional factor in considering the use of Russian rocket engines was the manner in which the engines would be manufactured; either in Russia, or in the U.S. through a licensing agreement. Commercial launch applications did not pose a problem for the use of Russian-built engines, thereby allowing the leveraging of the significantly lower labor costs in Russia. U.S. Government launch applications did, however, pose a perceived conflict, with organizations such as the Air Force advocating the licensing of production by a U.S. propulsion vendor as being the only acceptable solution.

The interest in the RD-701 was for application to Single Stage to Orbit (SSTO) launch vehicle concepts. Due to the fact that the RD-701 design was developed by Energomash to meet Russian SSTO requirements, NASA requested that an RD-701 concept be defined that met the Access to Space Option 3 team's SSTO mission requirements. To avoid confusion regarding the difference between the subsequent two tripropellant engine concepts, Energomash chose to identify the engine concept based on NASA requirements as the RD-704. An additional contract deliverable that was provided by Pratt was a preliminary set of data corresponding to the results of prototype tripropellant injector hot-fire tests that were performed by NPO Energomash.

LMSC utilized the RD-170 and RD-180 performance data in the assessment of candidate heavy lift and medium lift, expendable vehicles, and the RD-704 data in the assessment of candidate SSTO launch vehicles under the TA-2 charter. The Pratt deliverables were also contractually provided to the TA-2 COTR for internal NASA use. Because of the proprietary nature of the data contained in Pratt's deliverables, any request for copies of said deliverables should be made to the TA-2 COTR, Gary Johnson, of the Marshall Space Flight Center.

The following deliverables were provided by Pratt & Whitney, under subcontract to LMSC's TA-2 contract, with the associated statement-of-work task titles indicated:

Basic Subcontract Period (March-May 1993)

- Preliminary Assessment of Russian Propulsion Systems (Doc. No. FR 22861-1)
  - Task 1 RD-170 Manufacturing Location Assessment
  - Task 2 RD-170 U.S. Production Cost Identification
  - Task 3 RD-170 CIS Production Cost Identification
  - Task 4 RD-170 Performance and Operational Regime Specification
  - Task 5 RD-170 Test Requirements
  - Task 6 RD-170 Performance Enhancement
  - Task 7 RD-170 Launch Site Operations

Basic Subcontract Period (March-May 1993) (Concluded)

- Task 8 RD-701 Characterization and Performance Identification
- Task 9 RD-170 Existing Test Information
- Task 10 Final Report

Amendment A Subcontract Period (May 1993-May 1994)

- Preliminary Assessment of Russian Propulsion Systems Amendment A Volume I Executive Summary (Doc. No. FR 23379)

- Preliminary Assessment of Russian Propulsion Systems Amendment A Doc. No. FR 23365)
  - Task 1 RD-170 Acquisition and Detailed Test Assessment
  - Task 2 RD-170 Technology Assessment
  - Task 3 RD-701 (RD-704) Technology Assessment (injector test data delivered via addendum)
  - Task 4 Expander Cycle Rocket Engine Technology

Amendment B Subcontract Period (May 1992-May 1994)

- Preliminary Assessment of Russian Propulsion Systems Amendment B (Doc. No. FR 23317-2)
  - Task 1 Detailed RD-170 Manufacturing Location Assessment
  - Task 2 Detailed RD-170 U.S. Production Cost Identification
  - Task 3 (deleted)
  - Task 4 RD-180 Development, Performance, and Operation Information
  - Task 5 (deleted)
  - Task 6 (deleted)
  - Task 7 Detailed Assessment of RD-170 Existing Test Information
  - Task 8 Detailed RD-701 Characterization and Performance Information

NASA

EXPLORATION
PROGRAMS OFFICE

First Lunar Outpost Heavy Lift Launch Vehicle Design and Assessment

Preliminary Status Report

May 1992

Submitted
to the
Exploration Programs Office

By the
First Lunar Outpost
Heavy Lift Launch Vehicle Assessment Team

ACKNOWLEDGEMENTS

The First Lunar Outpost Heavy Lift Launch Vehicle Assessment Team wishes to acknowledge the contributions of the following people in the development of this status report:

Gene Austin, MSFC	HLLV Assessment Team Program Manager
Ed Svrcek, ExPO	HLLV Assessment Team Deputy Program Manager
Darryl Kendrick, ExPO	Requirements
Uwe Hueter, MSFC	Project Lead-HLLV
Buddy Pattison, MSFC	Project Lead-Trans-Lunar Injection Stage
Steve Cook, MSFC	Lead Engineer-HLLV
John Carlisle, MSFC	Lead Engineer-Trans-Lunar Injection Stage
Charlie Teixeira, JSC	Lead Engineer-Saturn V Derived Concept Definition
Mike Berning, Lockheed	NLS Derived Concept Definition
Bobby Brothers, MSFC	Mass Properties
Jack Chapman, MSFC	Propulsion
Gene Comer, MSFC	Thermal
Karen Doyle, SSC	Propulsion Test Facilities
John Fikes, MSFC	On-Orbit Controls
Woolsey Finnell, MSFC	Avionics
John Frassanito, JF&A	Vehicle Graphic Renditions
Tom Goodrick, MSFC	Aerodynamics
Allen Green, MSFC	Operations
Charlie Hall, MSFC	Ascent Stability and Control
Richard Hage, MSFC	Structural Analysis
Jim Hays, MSFC	Performance
Keith Holden, Lockheed	NLS Derived Concept Definition
Gary Johnson, MSFC	Report Editing
Dave Lanier, MSFC	Facilities/Operations
Paul Luz, MSFC	Structural Analysis
Jim McCurry, Lockheed	NLS Derived Concept Definition, Report Editing
Jack Mulqueen, MSFC	Mission Analysis
Wayne Ordway, JSC	Saturn V Derived Concept Definition
Bob Porter, MSFC	Structural Analysis
Kevin Sagis, Lockheed	NLS Derived Concept Definition
Robert Shepard, MSFC	Schedules
Susan Spencer, MSFC	Structural Analysis
Jim Steincamp, MSFC	Reliability
Dale Strider, MSFC	Layouts
Kevin Templin, JSC	Saturn V Derived Concept Definition
Steve Tucker, MSFC	Trans-Lunar Injection Stage Thermal Analysis
Jim Wiser, MSFC	Power Systems
Glade Woods, SSC	Propulsion Test Facilities

TABLE OF CONTENTS

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF FIGURES

Figure		Page

LIST OF TABLES

ACRONYMS AND ABBREVIATIONS

ASRM	Advanced Solid Rocket Motor
CA	forebody axial force coefficient, (n.d.)
CNa	forebody normal force coefficient derrivative with respect to angle of attack, (n.d)
CP	(aerodynamic) center of pressure, (n.d.)
DDT&E	design, development, test, and evaluation
deg	degree(s)
DoD	Department of Defense
EHA	electrohydrostatic actuator
EMA	electromechanical actuator
ET	External Tank
ETO	Earth-to-Orbit
ExPO	Exploration Programs Office
FLO	First Lunar Outpost
FLORG	First Lunar Outpost Requirements and Guidelines
FPR	flight performance reserve
ft	feet
FY	fiscal year
G	acceleration due to Earth's gravity, (m/sec^2)
GLOW	Gross Lift-Off Weight, (Kg)
HLLV	Heavy Lift Launch Vehicle
HTPB	Hydroxyl Terminated Polybutadiene
in	inches
Isp	specific impulse, (sec)
JSC	Johnson Space Center
K	thousand (kilo)
Kg	kilogram
Klbf	thousand pounds force
Klbm	thousand pounds mass
Km	kilometer(s)
KSC	Kennedy Space Center
lbf	pound(s) force
lbm	pound(s) mass

ACRONYMS AND ABBREVIATIONS

IMU	Inertial Measurement Unit
LEO	low Earth orbit
LOX	liquid oxygen
LH_2	liquid hydrogen
LTV	Lunar Transfer Vehicle
LVLH	Local Vertical Local Horizontal (Cartesian Coordinate System)
m	meter(s)
M	million (mega)
MAF	Michoud Assembly Facility
MECO	main engine cut-off
MHz	megahertz
Mlbf	million pounds force
Mlbm	million pounds mass
MLP	Mobile Launch Platform
MLT	Mobile Launch Tower
MPTA	Main Propulsion Test Article
MSFC	Marshall Space Flight Center
N	newton(s)
NASA	National Aeronautics and Space Administration
NLS	National Launch System
nm	nautical mile
PLF	payload fairing
psf	pounds force per square foot
Q	dynamic pressure, (N/m^2)
RCS	Reaction Control System
RPL	rated power level
RP-1	Rocket Propellant 1
s	seconds
S-Band	1550-5200 MHz
SOFI	spray-on foam insulation
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
SSP	Space Shuttle Program
STME	Space Transportation Main Engine

ACRONYMS AND ABBREVIATIONS

t	metric ton
TCS	Thermal Control System
TLI	trans-lunar injection
TPS	Thermal Protection System
TVC	thrust vector control
T/W	ratio of thrust to weight, (n.d.)
VAB	Vehicle Assembly Building
VHM	vehicle health management
ΔV	delta velocity, (m/sec)
X/D	ratio of x-body axis station location to diameter dimension, (n.d.)

1. Introduction

The United States will require a new heavy-lift launch vehicle (HLLV) system to meet the goals of the Space Exploration Initiative (SEI). These goals include lunar and Mars missions, requiring delivery of both crew and cargo, beginning in the year 1999. Current domestic earth-to-orbit (ETO) launch vehicle assets are incapable of providing the required payload capacity to support the SEI program. Additional launch vehicle capabilities are therefore required, that are either derived from past, existing, or planned elements, or are entirely new ("clean sheet") concepts.

Beginning in January, 1992, a conceptual design study was undertaken to define candidate launch vehicle configurations for the First Lunar Outpost (FLO) mission. A joint NASA Marshall Space Flight Center (MSFC), Kennedy Space Center (KSC), Stennis Space Center (SSC), and Johnson Space Center (JSC) team was formed to analyze the options. The FLO payload mass requirements have ranged from 76 t (167 Klbm) to 93 t (205 Klbm) after trans-lunar injection (TLI). While the current payload requirement is 93 t post-TLI, some analysis results contained in this report represent a previous 76 t requirement. The FLO requirement that each piloted or cargo mission be performed with a single launch, in order to reduce on-orbit operations, has the single largest effect on defining the candidate launch vehicle concepts. The single-launch scenario equates to a launch vehicle payload mass requirement twice that of the Apollo Saturn V. An additional constraint assessed for the lunar launch vehicle was the desire to utilize existing KSC facilities and ground support equipment (GSE) to the maximum extent possible. One resulting derived requirement is a 119-125 meter (390-410 foot) limit to the launch vehicle length, in order to vertically clear the Vehicle Assembly Building (VAB) high bay entrance while on the mobile launch platform (MLP) and crawler.

The HLLV, as defined by this study, is comprised of a core, boosters, second stage (if required), trans-lunar injection (TLI) stage, and a payload shroud. Requirements for a Mars mission were also defined so that evolutionary paths could be identified for the candidate lunar vehicles and associated infrastructure. Due to the current uncertainty in the Mars payload definition, 250 t is viewed as a minimum Mars payload requirement to be delivered into low Earth orbit (LEO). Both a reference National Launch System (NLS) derived and Saturn V derived HLLV have been baselined for further analysis. This report presents a status of the assessment of those configurations. As the requirements and the implications of the assessment results are better understood, modifications to the candidate HLLV concepts will be identified and analyzed.

The FLO activity is an on-going requirements development process that will progress through numerous iterations before a final selection of the preferred technical approach. Current FLO concepts, as documented in this report, provide a framework for developing and testing requirements. The concepts herein should be treated as a "first cut" that will be refined considerably as analysis proceeds. The concepts are not final, and other candidate concepts have not been ruled out. Additional concepts, approaches, and issues will be identified and assessed.

2. FLO Requirements and Reference Missions

2.1 Requirements

Figure 2.1-1 summarizes the First Lunar Outpost (FLO) requirements that affect the HLLV design. The requirements are included in the First Lunar Outpost Requirements and Guidelines (FLORG) document that was issued by the Exploration Programs Office (ExPO). The study requirements include providing the capability to place a cargo, i.e., habitat and science experiments, onto the lunar surface in a single launch, and sending a four-man crew to the moon and back to Earth, in a single launch. The cargo mass to be placed onto the lunar surface is 31.4 t (69 Klbm) without a manager's reserve. Assuming a 10 percent manager's reserve, this translates to a 93 t (205 Klbm) payload requirement post-TLI for the launch vehicle. The piloted mission requirement without a manager's reserve, but including 5 t (11 Klbm) of usable cargo, currently totals 32.7 t (72 Klbm) to the lunar surface. The same launch vehicle is to be used for both piloted and cargo missions.

Requirements

ExPO

1. The Earth to Moon Transportation System (HLLV, TLI Stage, Lander) Shall Provide The Capability To Emplace 27.5 t (Including 10% Manager's Reserve) On The Lunar Surface In A Single Flight. (Current Assessment Is 34t Of Cargo With Margin Resulting in 93t To TLI)

2. A Single HLLV Shall Be Utilized For Each Flight To The Moon.

3. The HLLV Shall Provide The Capability For Designed Growth To 250 t To 220 nm.

4. Flight Elements Shall Provide The Capability To Access Any Lunar Latitude Or Longitude.

5. The HLLV Shall Provide The Capability For Launch As Early As 1999.

7. The Capability Shall Be Provided To Support Four (4) Flights Per Year.

Figure 2.1-1 First Lunar Outpost Mission Requirements

2.2 Launch Vehicle Reference Missions

The purpose of the launch vehicle reference mission for both the piloted and cargo single-launch lunar scenarios is to place the TLI stage and payload into a 185 Km (100 nm) circular orbit from any launch azimuth between 72 and 108 degrees, assuming the use of Launch Complex 39 at KSC. A slightly different ascent mission profile was developed for the Saturn V-derived and NLS-derived vehicle options as a result of vehicle specific characteristics. Sections 4.1.1.1 and 4.2.1.1 discuss those respective profiles. The orbital mission profile is the same for the two vehicle options. Figure 2.2-1 illustrates the mission profile during the orbital phase in low Earth orbit.

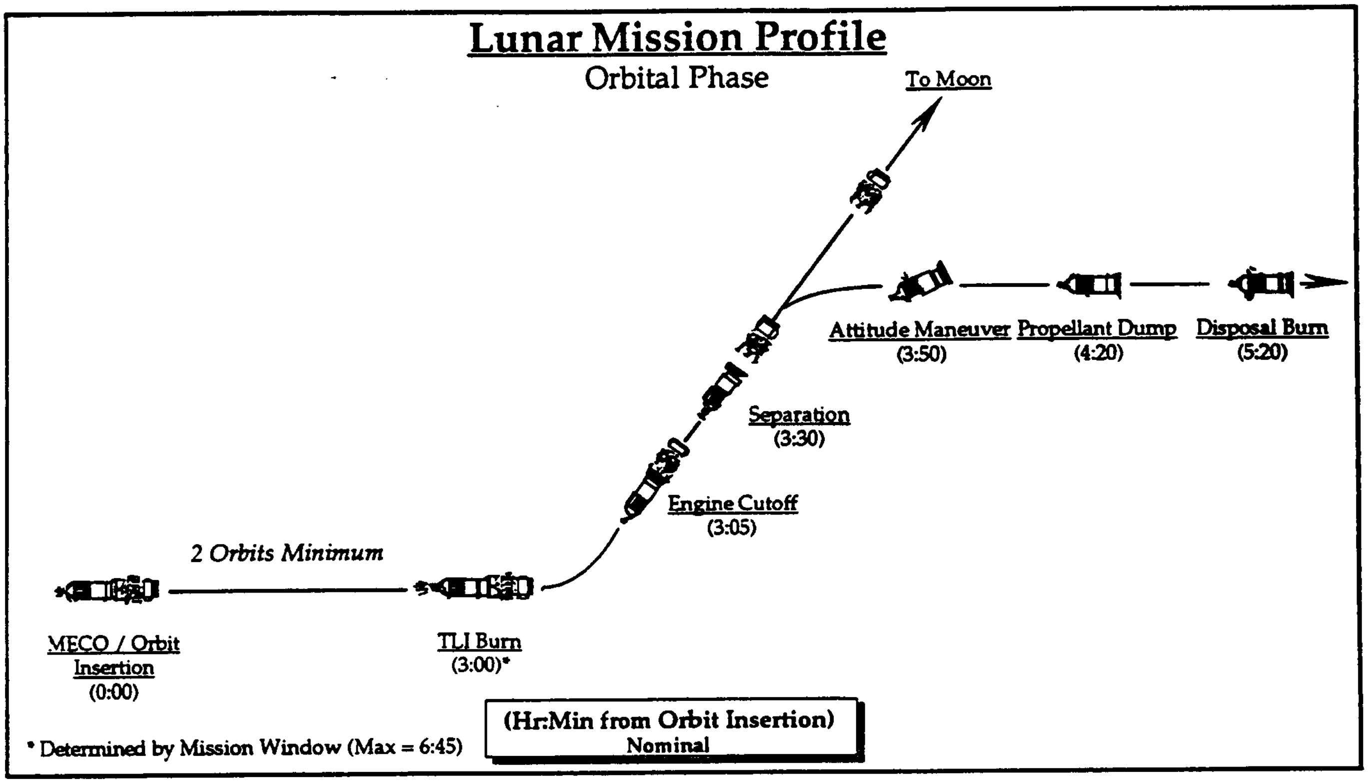


Figure 2.2-1 Single Launch Mission Profile During Orbital Phase

3. Groundrules and Assumptions

3.1 Groundrules

Figure 3.1-1 summarizes the groundrules that were used to assess the sizing and performance of the candidate Saturn V-derived and NLS-derived vehicle configurations.

Groundrules and Assumptions

- Payload Size: 93 t (204.6 Klbm) after Trans-Lunar Injection
  - 10m (33 ft) Diameter x 18m (60 ft) Length
- Maximum Acceleration During Ascent: 4 Gs
  - Use Step Throttling / Engine Shutdown for acceleration limiting
- Maximum Dynamic Pressure (Max Q): 43.1 N/m^2 (900 psf)
- Minimum Thrust-to-Weight at Liftoff: 1.2
- Jettison Shroud / Nosecap at Altitude of 121.6 Km (400,000 ft)
- No Engine-Out on Core / Boosters / Upperstage
- Earth Orbit (Circular): 184.8 Km (100 nm) Pre-TLI Burn Check-out Orbit
- Launch Azimuth Capability of 72 deg to 108 deg
- 60 Day Launch Centers: Minimum
- Primary Propulsion Options Include: F-1A, J-2S, SSME, and STME
- 10 Percent Dry Mass Contingency
- Ascent Flight Performance Reserve: 1 Percent of the Total Ascent ΔV
- Primary Avionics Located on TLI Stage
- On-Pad Hold-Down During Booster/Core Engine Start
- Minimize Impacts to Existing or Planned KSC Facilities

Figure 3.1-1 Study Groundrules and Assumptions

The candidate HLLV configurations were sized to meet or exceed the minimum payload requirement of 93 t post-TLI. A constraint was imposed on the physical sizing of the launch vehicles based on the groundrule of minimizing any resulting impacts to existing Kennedy Space Center ground processing facilities. As a result, the Vehicle Assembly Building (VAB) highbay door vertical clearance constraint was used to limit the total length of the launch vehicle, when considering that the vehicle would be mounted on a mobile launch platform (MLP) and carried into the VAB by a Saturn V/Space Shuttle Program style crawler.

Separate bulkheads between the LOX and fuel tanks are used to simplify design, manufacturing, and operational complexity (except with the Saturn V S-II stage derivative). All tank endcaps are elliptical with a semi-major/semi-minor axis ratio of $\sqrt{2}$. All tanks have been designed with an ullage volume of 3 percent of the propellant volume. The flight performance reserve (FPR) is quantified to be one percent of the total ascent delta velocity and is bookkept in the last ascent core stage. The maximum thrust acceleration and dynamic pressure that the vehicle will be allowed to experience during ascent is 4.0 Gs and 43.1 $K\,N/m^2$ (900 psf), respectively. The payload shroud for the cargo mission will be jettisoned at 121.6 Km (400,000 ft) geodetic altitude. The launch escape system for the manned mission will also be jettisoned at 121.6 Km (400,000 ft) geodetic altitude.

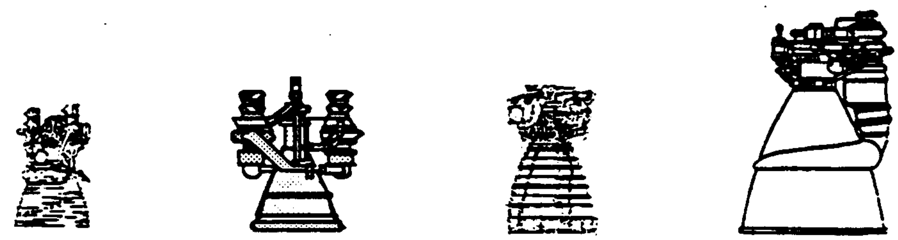
3.2 Assumptions

The HLLV design activity seeks to identify and assess candidate HLLV configurations that could satisfy the FLO requirements discussed in Section 2.1. In order to bound the solution set of possible HLLV configurations to a manageable number that could be assessed in a pre-Phase A environment, a series of design assumptions were made. The following sections highlight those assumptions.

3.2.1 Propulsion Options

The demonstrated capability and reliability of the Saturn V propulsion systems were among the most significant attributes which led to consideration of a vehicle using Saturn V technology. The F-1A and J-2S engine concepts were baselined for use on the Saturn Derived design option. These are evolutionary concepts of the F-1 and J-2 engines incorporating modifications for improvements in performance, reliability, and manufacturing. Space Transportation Main Engines (STMEs) and an upper stage version of the Space Shuttle Main Engine (SSME) were baselined for the NLS Derived design option, along with the additional use of F-1As. The four engine types are not reusable, but are assumed to have a designed-in capability for multiple firings in order to support flight readiness testing. The additional development effort and schedule risks associated with the new Saturn derived engines are deemed to be minimal, since both engines have already gone through partial development and testing by Rocketdyne. Figure 3.2.1-1 summarizes the performance characteristics of the four respective engines.

Liquid Engine Comparison



	J-2S	STME	SSME	F-1A
Thrust (lbf)				
VAC	265,000	650,000	470,000	2,020,500
SL	—	551,430	375,000	1,800,000
ISP (s)				
VAC	436.0	428.5	452.9	304.2
SL	—	364	361.4	271.0
Mixture Ratio	5.5:1	6:1	6:1	2.27:1
Chamber Pressure (psia)	1,200	2,250	3,006	1,161
Expansion Ratio	40:1	45:1	77.5:1	16:1
Length (ft)	11	13	14	18.4
Nozzle Exit Dia (ft)	6.7	8	7.5	12
Mass (lbm)	3,800	9,974	6,990	19,000 (est)
Throttle	None	Step to 75%	65%-109%	Option: Step to 75%
Other				Tests: 1,800 Klbf - 1700 sec Step Throttle - 24,000 sec

Figure 3.2.1-1 Performance Specifications of Candidate Engines

F-1A

The F-1A uses liquid oxygen as the oxidizer and Rocket Propellant 1 (RP-1), or high grade kerosene, as the fuel, turbopump lubricant, and hydraulic working fluid for the thrust vector control and engine valve components. A gas generator drives the turbine which is direct-coupled to the turbopump. Several improvements to the baseline F-1 design result in an F-1A engine capable of a larger 100 percent rated power level (RPL), 8,006,760 N (1.8 Mlbf), and the capability to step throttle to a minimum value of 75 percent RPL, 6,005,070 N (1.35 Mlbf).

J-2S

The J-2S engine is an uprated version of the J-2 engine, which was used on the Saturn V