DOD LOW-THRUST MISSION STUDIES

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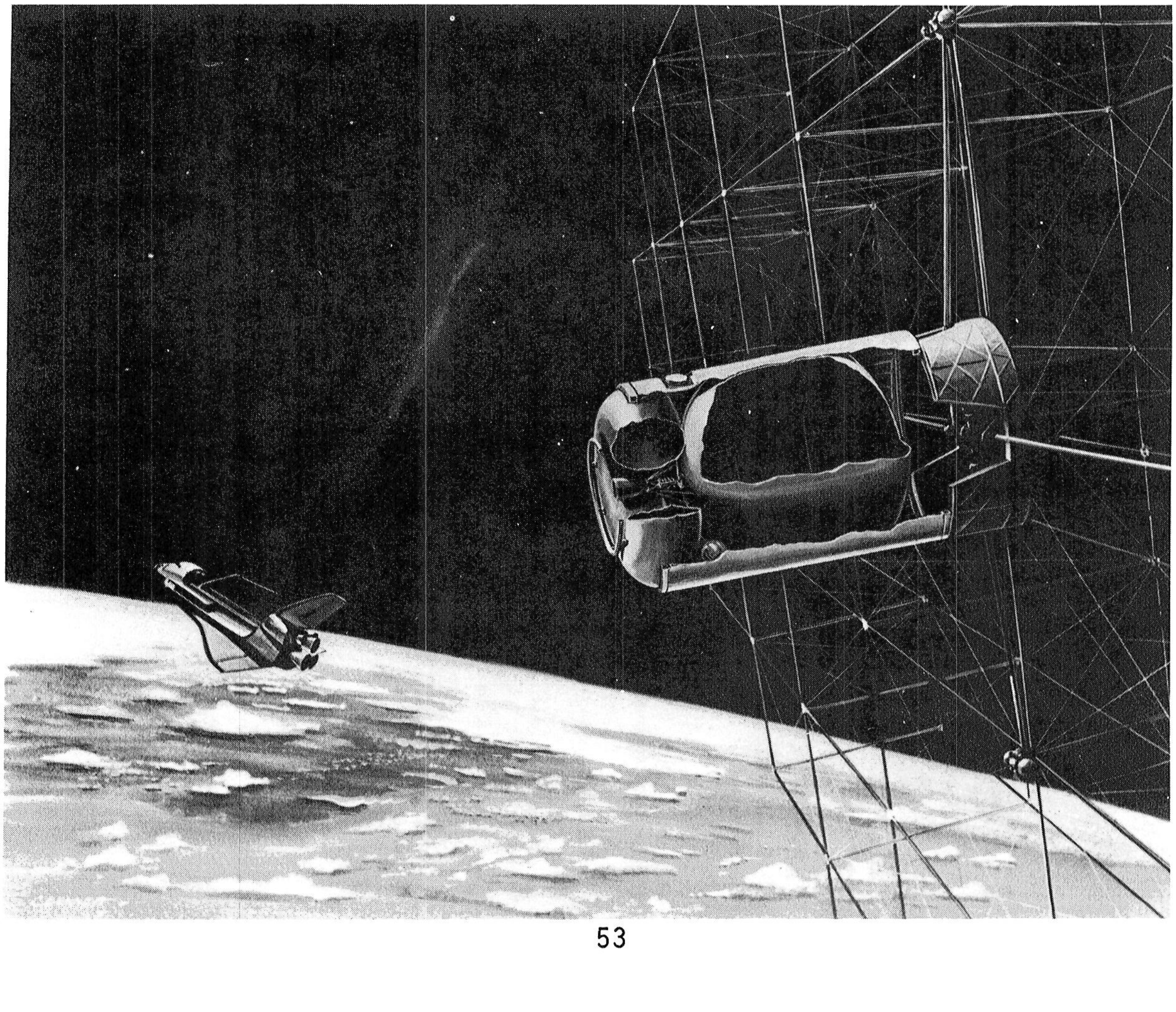
Advanced Low Thrust Propulsion System

The Space Transportation System (STS) will be the principal means of launching USAF spacecraft beginning in the 1980's. Since it is manned and reusable it provides new opportunities for unique approaches for cost effective utilization of its capabilities. The STS also places additional requirements and constraints on advanced spacecraft deployment systems that did not previously exist for expendable launch vehicles. To fully utilize these new capabilities designers must be prepared by having cost-effective technologies available. Martin Marietta Corporation under contract to the Air Force Rocket Propulsion Laboratory (F04611-79-C-0032) performed a study to identify advanced propulsion technology that would provide flexibility, performance, and economic benefits to future Air Force missions.

The figure shown is an artist concept of an advanced low thrust propulsion system delivering a Large Space System from the Shuttle orbit to high earth orbit. This LO_2/LH_2 stage with a torus LO_2 tank and 500 lbf pump fed engine is high on the list of propulsion technology.



Study Ground Rules and Assumptions

The study ground rules and assumptions are presented here. Emphasis was placed on the military requirements for space missions planned from 1985 to the year 2000. NASA missions that complemented the DOD missions were also considered. In most cases all the Non-DOD (NASA, commercial & foreign) missions complement DOD with the exception of planetary missions. Therefore all planetary missions were excluded.

All of the missions were assumed to operate out of the Shuttle with performance and constraints defined in JSC 07700, "Space Shuttle Systems Payload Accommodations". All spacecraft deployment performance requirements are deltas from the standard Shuttle circular orbit of 160 nautical miles. By statement of work advanced STS capability such as the Advanced Military Space Flight Capability or Heavy Lift Launch Vehicle (HLLV) were not evaluated.

Study Ground Rules and Assumptions

Emphasis on Military Requirements 1985 to the Year 2000

Consider NASA Planning That Complements DOD Geocentric

STS Baseline Capability JSC 07700

- ETR; 65,000 lb, 160 n mi Circular at 28.5 deg
- WTR; 32,000 lb, 160 n mi Circular at 98 deg

Advanced STS Capability Not Considered

- Heavy Lift Launch Vehicle (HLLV)
- Advanced Military Space Flight Capability

Propulsion Concepts Considered

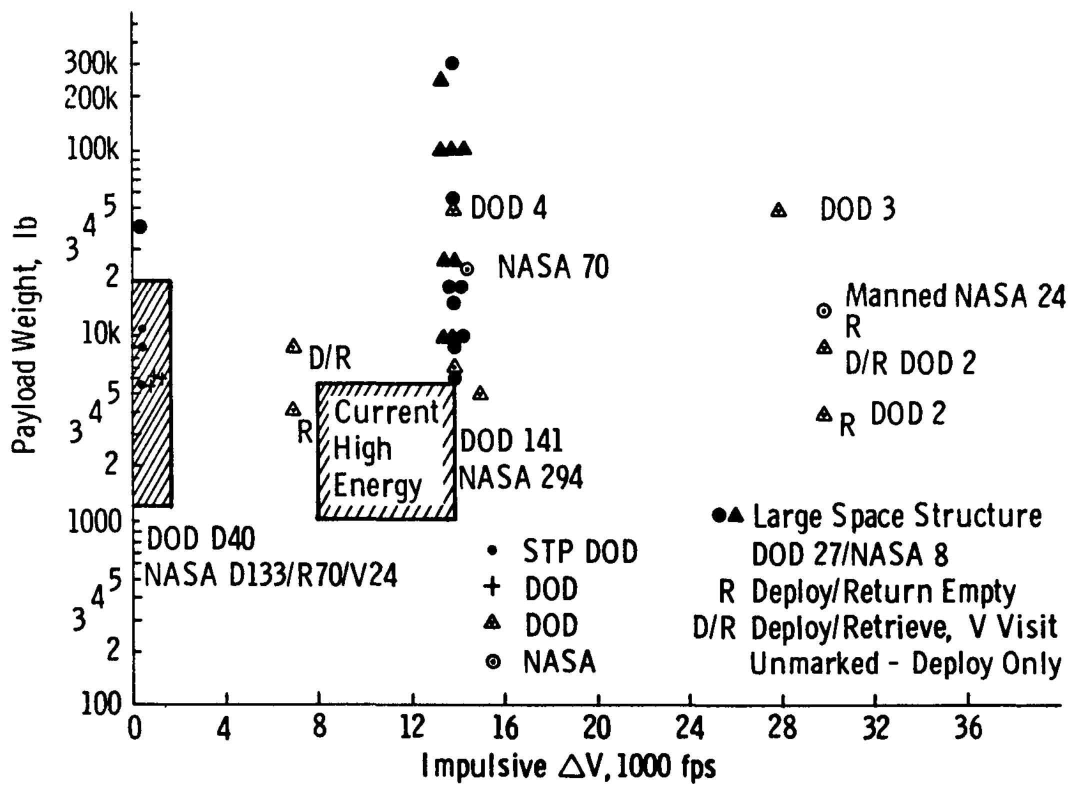
- Liquid Cryogenic and Storable (SOA and ASOA)
- Electric (SOA and ASOA)
- Solid (SOA)
- Combinations

Total Mission Catalog

The results of the Phase I mission characterization are presented here. The mission model contains low energy missions, high energy missions, and future missions which include large space systems. The quantity of missions are indicated in each area and is separated between DOD and NASA which includes commercial and foreign. As can be seen some missions are very large in weight such as the Solar Power Satellite while others require large amounts of delta velocity such as the manned mission to geosynchronous. The low energy NASA missions include deploy (D), retrieve (R), and visit (V).

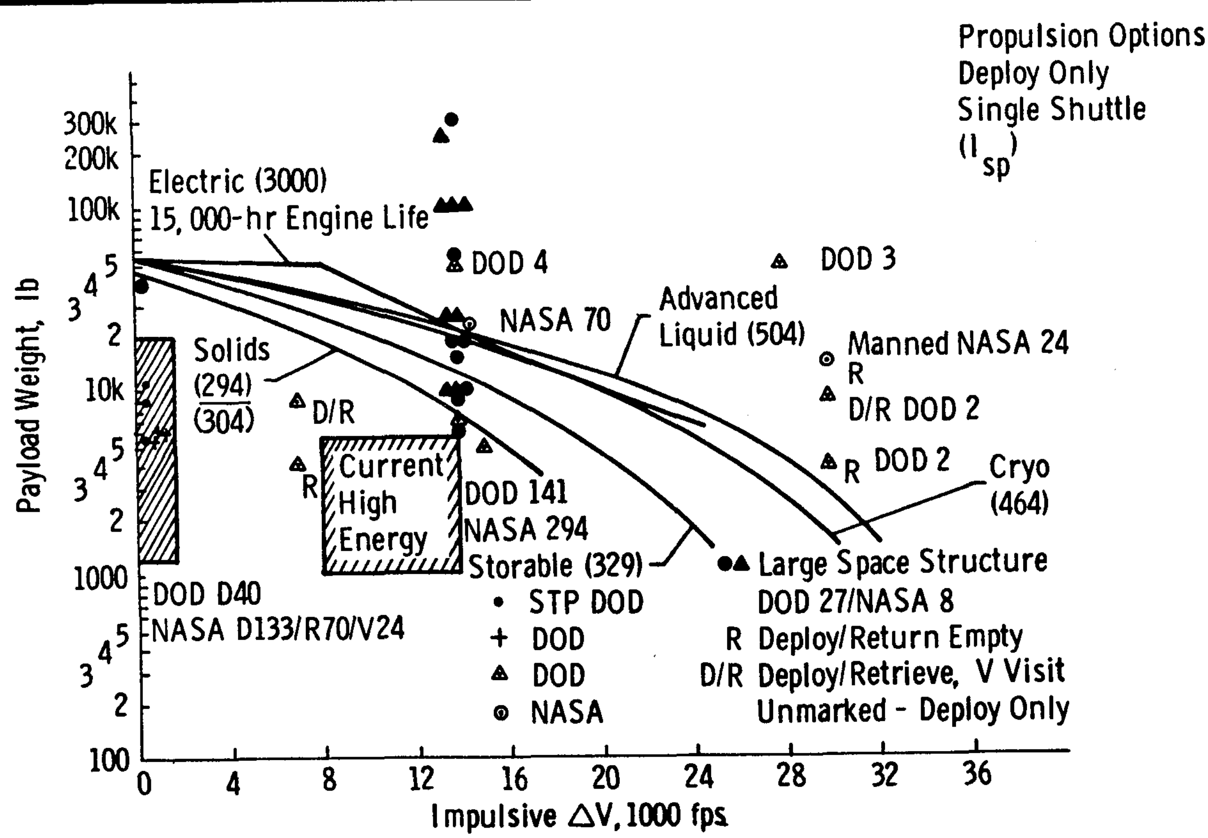
The Large Space Systems (LSS) are indicated by the solid triangles and circles for DOD and NASA respectively.

Total Mission Catalog



To capture these missions different deployment techniques were evaluated to determine single shuttle capability as well as multiple shuttle capability using multiple spacecraft systems. This figure compares the performance capability of the different propulsion systems. The figure includes both state-of-the-art technology and advanced technology such as the advanced liquid with 504 seconds specific impulse representing the upper limit for chemical propulsion ($LF_2/LH2$), excluding the use of metal additives which can increase the performance an additional 40 seconds. As can be seen in the figure there are Large Space Systems that cannot be captured or satisfied by a single shuttle launch.

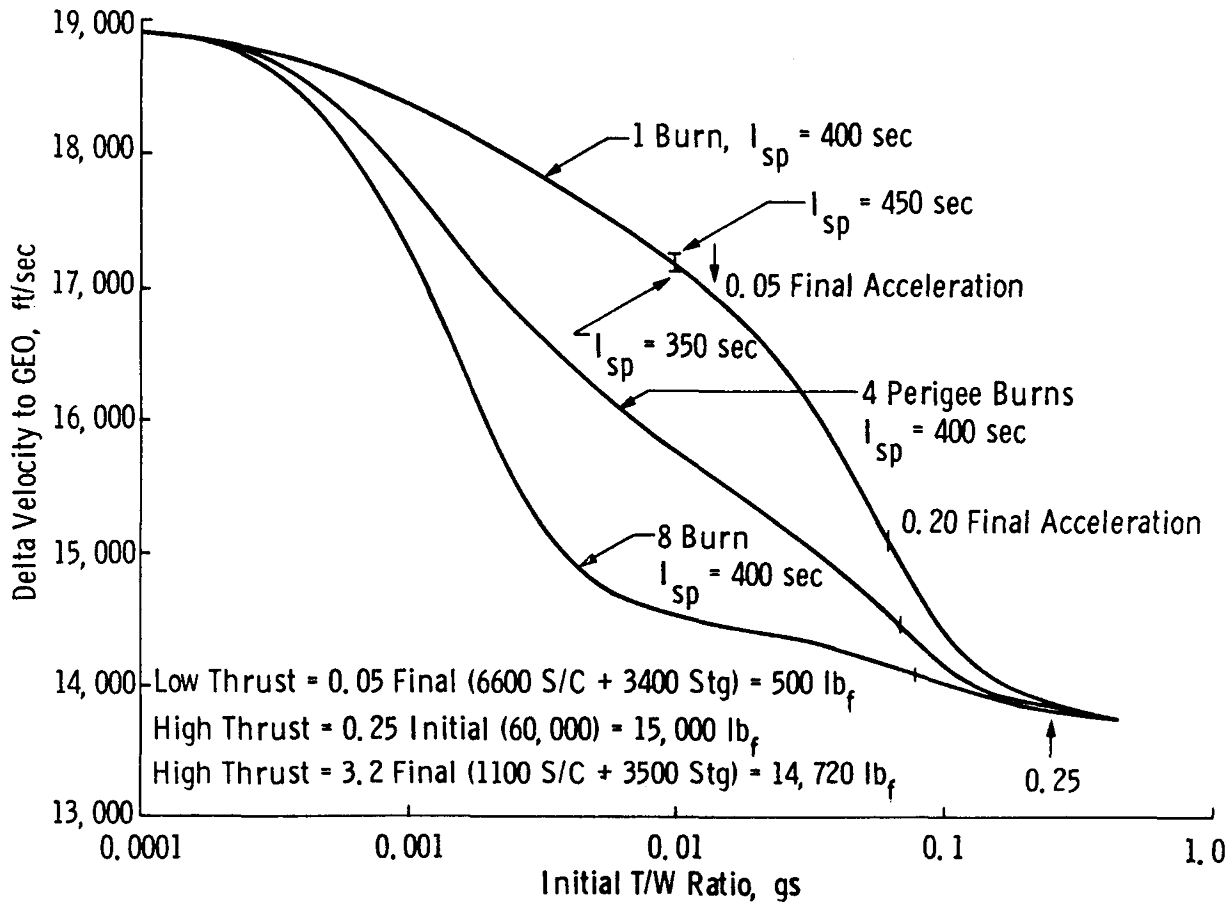
Total Mission Catalog



Acceleration and ISP Effects on Delta Velocity

Large Space Systems deployed in low earth orbit and transferred to higher orbits require low thrust to keep from exceeding their structural capability. The impact of low thrust to weight on delta velocity required is presented here. As thrust decreases to meet the LSS g-level requirements (0.05 gs) the delta velocity required to geosynchronous orbit increases due to burn inefficiencies. One way to increase performance or reduce the delta velocity required is by multiple perigee burns. The three curves are for 1, 4, and 8 perigee burns at an Isp of 400 seconds. If initial thrust to weight is at or above 0.25 g's the effects of low thrust are negligible. Using this initial point and the final burn out g-level of 3.2 for non-LSS spacecraft results in a thrust level of approximately 15,000 lb_f whereas the g-level for LSS spacecraft requires a thrust level of approximately 500 lb_f.

Acceleration and ISP Effects on Delta Velocity



Liquid Chemical Propulsion Vehicles for LSS

Presented here is a summary of the Large Space Systems requirements and the resulting vehicle requirements. With the exception of two DOD missions the spacecraft descriptions were very general with regard to orbiter packaging. The spacecraft were defined as simply one or more shuttle orbiters full. A total of 35 spacecraft were identified of which 27 are DOD. The stages were sized for the stage plus Airborne Support Equipment (ASE) and spacecraft delivery capability to equal 65,000 lbs. Mission durations were defined as a minimum of 8 days to a maximum of 60 days. The minimum value was established as approximately 7 days in shuttle orbit for spacecraft deployment and checkout and approximately 1 day (31 hrs for 8 perigee burns) for transfer to geosynchronous orbit. The 60 days was based on the requirement to assemble stages in low earth orbit to satisfy the impulse required for the larger LSS missions.

Six vehicle configurations were selected to compare the relative economic benefits of storable propellants and cryogenic propellants including an advanced combination, throttleable engine, tripropellant, and a minimum length cryogenic stage with torus LO_2 tank.

A mission capture analysis was performed for each candidate configuration with the results shown here. As indicated the lowest capture results from the advanced propellant candidate. However, the difference is small compared to the three LO_2/LH_2 concepts. The storable and tripropellant capture results are much higher due to the lower performance.

Liquid Chemical Propulsion Vehicles for Large Space Systems

Mission Description

- Spacecraft Weight Range = 6,000 to 300,000 lbm
- g-Level = 0.05 to 1.0
- All S/C Fill Orbiter Bay Except for 2 DOD
  - DOD 8 Missions/27 S/C
  - NASA 8 Missions/8 S/C

Vehicle Requirements

- Low Thrust (500 lbf)
- Spacecraft + Stage + ASE = 65,000 lbm
- 14 ft Dia x 34 ft Length (Max)
- Mission Duration (8 Days to 60 Days)
- 9 Burns Total (Max), ΔV = 14,600 ft/s

Six Concepts Identified

		Shuttle Flights	
	Length, ft	DOD	NASA
- Baseline (N_2O_4/MMH)	15.1	177	56
- Tripropellant ($CLF_5/N_2H_4/LH_2$)	25.4	156	52
- Max Perf (LO_2/LH_2)	22.7	134	47
- Max Perf (LF_2/LH_2)	18.4	132	45
- Throttleable (LO_2/LH_2)	22.3	134	47
- Minimum Length (LO_2/LH_2) - TORUS	17.0	134	47

Mission Capture Ground Rules

The mission capture ground rules used for the study are shown in the accompanying table.

Mission Capture Ground Rules

No DOD and NASA Mixing

For Grouping, the Payload Must Fly in the Same Year

Launch Site Must Be the Same

Available Shuttle Length 60 - 4 ft = 56 ft

Maximum Diameter = 14 ft

Payload Adapter Length 2 ft

Payload Adapter Weight 10% of Payload (Maximum of 1000 lb)

Grouped Payloads Require Diameter Spacing of 1 ft

Single Shuttle Flights

Reusable - Expend Only When Required for Delivery

Stage Dry Weight Contingency 10%

Flight Performance Reserve 2% (ACPS 10%)

ASE 3, 000 to 5, 000 lb Based on Diameter (Existing Stages Use Actuals)

LCC Analysis

To quantify the benefits of advanced technology Life Cycle Cost (LCC) was developed for each propulsion candidate based on the mission capture results. The approach to costing the propulsion candidates was to review the previous storable and cryogenic Space Tug studies and determine the major cost elements. In addition cost differences were reviewed to determine how cost would be affected by the different propulsion stage candidates. Applicable Cost Estimating Relationships (CER) were then obtained and the concepts costed based on the mission capture analysis. The costs are presented in 1980 dollars with a 95% learning curve applied.

LCC Analysis

Approach

Review Storable and Cryogenic Tug Studies

- Determine Major Cost Elements and Cost Differences
- Obtain Applicable CERs
- Cost Concepts Accordingly

Ground Rules

FY 80 $

Refurbishment Cost 30% of Unit (Reuse)

95% Learning Curve

10% Contingency Factor on All Configurations

Reliability Loss (Sensitivity)

- LCC for Resupply Includes Two Delta Missions Lost

LCC Cost Areas

The major cost elements included are: RDT&E, investment or production, operations, and shuttle launch cost. The sub-elements include avionics, structures, thermal, propulsion (tanks, engine, propellant feed, pressurization, attitude control propulsion system, and propellant), Airborne Support Equipment (ASE), systems engineering, and project management.

The costs not included are technology development, spares and logistics (which are small) facilities, and Ground Support Equipment (GSE). For facilities and GSE it was assumed that existing systems would be used or any changes would be similar for each concept. An advanced propellant loading facility was found to be small (0.1%) compared to the total LCC.

LCC Cost Areas

The Following Elements Are Included in Our Cost Analysis:

Major Elements
- RDT&E
- Investment
- Operations
- Refurbishment (Reuse)
- Shuttle Launch Cost

Subelements
- Avionics
- Structures
- Thermal
- Propulsion
  Tanks
  Engine
  Propellant Feed
  Pressurization
  ACPS
  Propellant
- ASE
- Systems Engineering
- Project Management
- Reliability (Sensitivity)

Cost Elements Not Included:

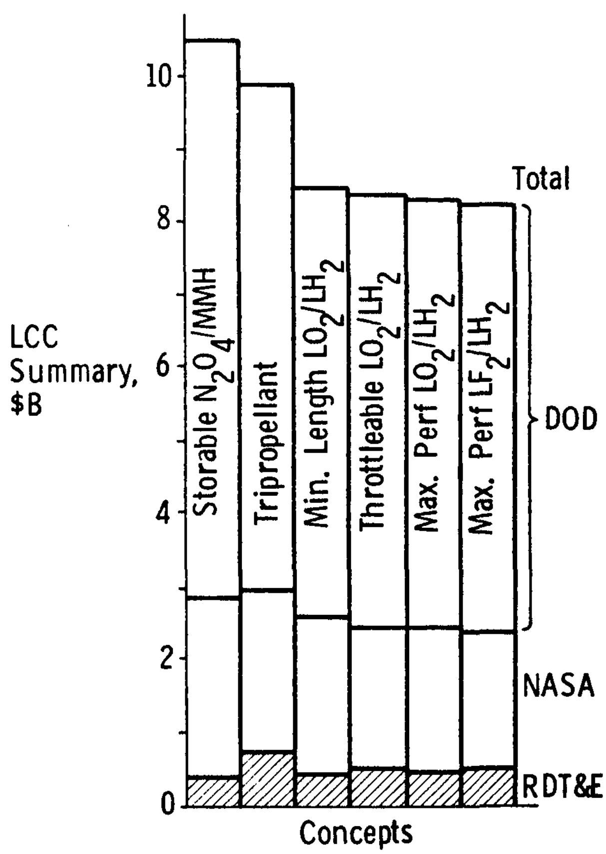
- Technology Development
- Spares
- Logistics
- Facilities (Use Existing/Changes Similar for Each Concept)
- Adv Propellant Loading Equipment (RDT&E ~ \$3.9M Unit ~ \$2.8M)
- GSE (Assumed Similar for Each Concept)

LSS Conclusions for Liquid Chemical Vehicles

Based on the mission capture analyses and LCC conclusions were reached regarding advanced technology for the LSS category of missions.

For the six stage configurations evaluated the LCC results are presented here. The results indicate that the cryogenic stage configurations are significantly lower cost than the storable and tripropellant. There are also no LCC advantages for a throttleable engine; however, interaction with the large space system due to dynamic effects may prove to be beneficial. It can also be seen that there are no LCC advantages for advanced propellants and no LCC advantage or penalty for the short torus LO_2 tank stage. This in part is due to the LSS mission definitions which in all but two DOD cases the spacecraft filled the Orbiter independent of the stage. However, from other studies performed by Martin Marietta as well as other mission categories in this study the importance of length is recognized. It is also important that the conclusions for DOD and NASA missions are the same.

LSS Conclusions for Liquid Chemical Vehicles



- Low Thrust and High Performance Required
- LCC of Cryogenic Stages Lower Than Storable Combinations
- LCC of All Cryogenic Stages Nearly the Same
- No LCC Advantage for Tripropellant or Advanced Propellants
- No LCC Advantage for Throttleable Engine
- No LCC Advantage or Penalty for Short Stage (Note LSS Mission Definitions)
- DOD Results Unchanged by NASA

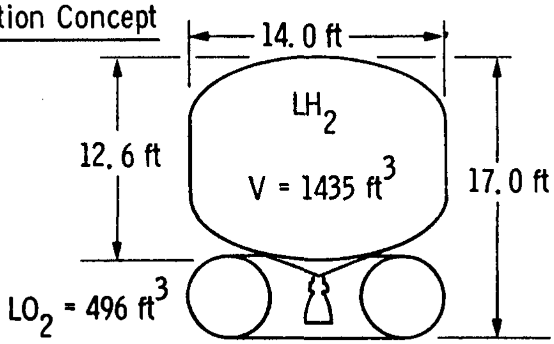
Chemical Propulsion Technology Requirements

As a result of this study the recommended chemical propulsion technology is low thrust/high performance pump fed engines combined with torus propellant tank technology. Neither of these technologies exist in a mature form and are required to meet the Large Space System requirements of the near future. The thrust level is approximately 500 lb_f and the key technology areas include small pumps, high chamber pressure, engine cooling, engine life in excess of 5 hours, and large gimbal capability. Torus tanks have not been constructed in 14 ft diameters and the propellant acquisition feed and thermal management has not been evaluated and demonstrated in these sizes or with cryogenic propellants. Summarized here are the configuration concept and key propulsion technologies. The engine performance has been updated to an Isp of 466 based on a point design provided by Aerojet Liquid Rocket Company under subcontract to Martin Marietta Corporation on the AFRPL study effort. The engine utilizes a staged combustion dual preburner engine cycle with a chamber pressure of 1000 psia.

The torus LO_2 tank was selected over other configurations based on an assessment of other tank arrangements including parallel tanks, tandem tanks, and common domes. The LO_2/LH_2 combination was also compared to LO_2/LCH_4 and found to provide nearly 1/3 more performance and for our mission model resulted in LO_2/LH_2 being the lowest life cycle cost candidate.

Chemical Propulsion Technology Requirements

Configuration Concept



Notes: Propellant LO_2/LH_2, MR = 6.0
Engine: Constant Thrust
ϵ = 400:1, Pc = 1000 psi, 96% Eff, ISP = 466 sec
Burns = 9, ΔV = 14,600 ft/s
7 Day Shuttle Orbit
Transfer Time = 31 Hours
2% Flight Performance Reserve
10% ACPS Propellant Margin
Self-Pressurization with Helium Tank for Start
Stage Weight = 44,940 lbs
Mass Fraction = 0.856
Payload Delivery = 17,060 lbs

Key Propulsion Technology

Engine Performance Demonstration
- LO_2/LH_2 Pump Fed
- Thrust = 500 lbf
- Cycles = 10
- Gimbal = ±10 deg
- Life = 5.4 hrs

Small Pumps
- Mixture Ratio Control

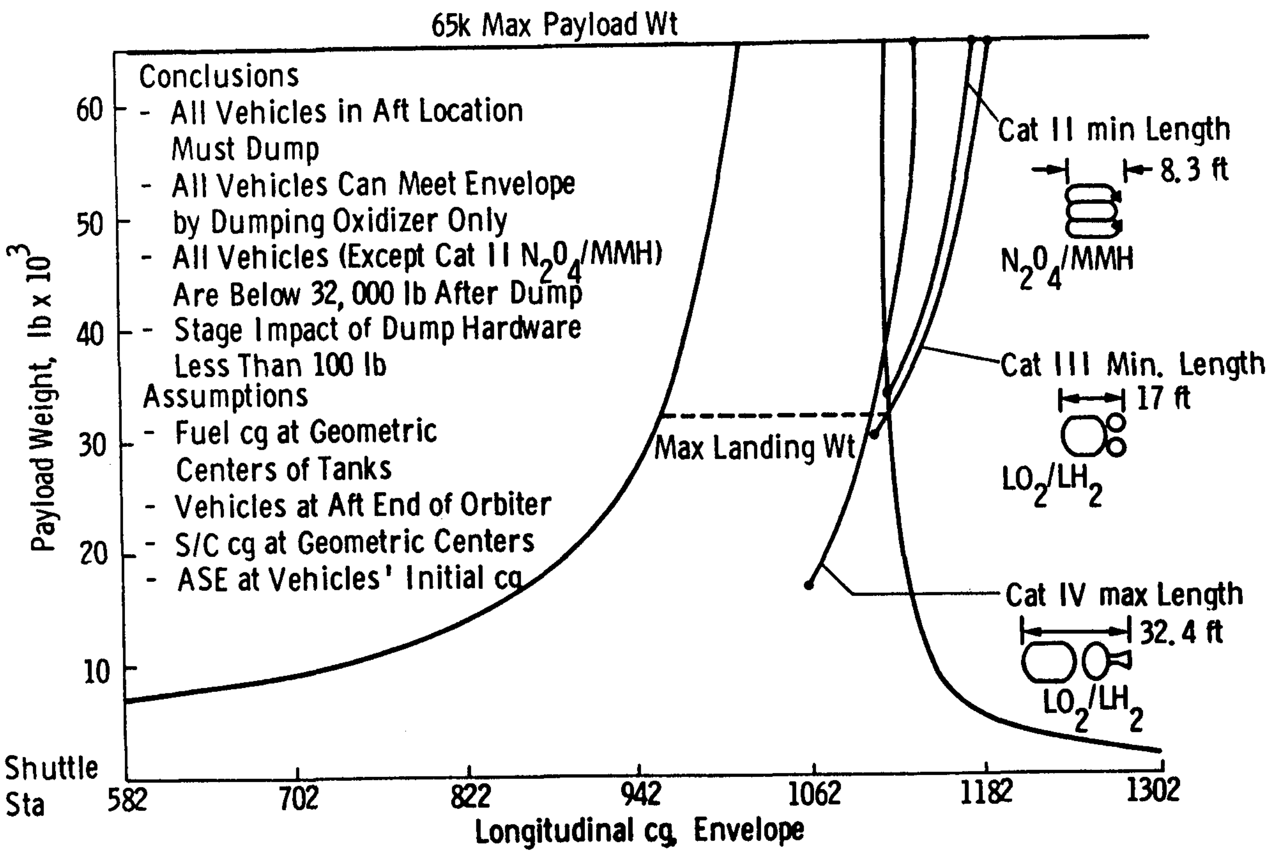
High Chamber Pressure
- Pc = 1000 psi
- ϵ = 400:1

Large Torus Tank
- 14 ft Diameter
- Weight and Manufacturing
- Propellant Management

Orbiter Payload cg Envelope During Abort

It should be noted that the large orbit transfer vehicles will require propellant dump as shown here. The most critical abort mode because of the time available is the ascent abort. This mode assumes one engine out on the Orbiter which must immediately return to the launch site since it cannot achieve orbit. The dump philosophy is to dump during powered flight above 150,000 ft. This period was selected because it provides the highest beneficial g forces, eliminates possible Orbiter ingestion, minimizes dump thrust impact on Orbiter control, minimizes the effect of center of gravity change on the Orbiter, and the propellant orientation relative to the dump outlet is the same for on-orbit dump. All vehicles must dump oxidizer to stay within the Orbiter center of gravity constraints. For this reason, parallel redundancy is required in the oxidizer system. Fuel could also be dumped; however, this imposes additional requirements on the dump pressurization system as well as requiring another set of large dump lines impacting both the stages and Orbiter. Fuel can be dumped on-orbit when time is available. For the cryogenic stages LH_2 disposal is by boil-off rather than drain; therefore, a horizontal vent is required.

Orbiter Payload cg Envelope During Abort for ASDS Vehicles



Electric Propulsion Vehicles for LSS

The electric propulsion analysis included five stage concepts utilizing various power options as shown here. The power options include nuclear and solar with consideration of power on the stage or spacecraft. Many of the large spacecraft require large amounts of power which can potentially be utilized by the electric propulsion system.

The stage definitions, mission capture analyses, and Life Cycle Cost (LCC) generation were prepared for comparison of the five concepts. The concepts include a baseline mercury ion with a 50 KW solar power supply, three large inert gas (Xenon and Argon) thruster systems (considered as next generation), and a magnetoplasmadynamic (MPD) system utilizing a 200 KW nuclear power source. The stage concepts were compared on the basis of how well they can deliver the required spacecraft for the LSS missions in terms of stages required, shuttle flights, and LCC.

The baseline 50 KW SEPs concept using 30 cm mercury ion thrusters was sized to meet a thrust to drag ratio of 10 for an assumed 600 ft LSS in the minimum drag orientation. This resulted in the selection of 8 BIMOD units to maximize thrust and packaging availability in the orbiter.

The number of stages and shuttle flights to capture the missions are also summarized here. The MPD has a slight increase in shuttle flights since the stage is carried up separate from the spacecraft. However, the number of stages required are approximately half that of the other concepts due to the higher performance of the MPD.

Electric Propulsion Vehicles for Large Space Systems

Mission Description
- Spacecraft Weight Range = 6,000 to 300,000 lbm
- g-Level = 0.05 to 1.0
- All S/C Fill Orbiter Bay Except for 2 DOD
  - DOD 8 Missions/27 S/C
  - NASA 8 Missions/8 S/C

Vehicle Requirements
- Solar Power Vehicles Require 1 OMS Kit
- Nuclear Power Vehicles Require 2 OMS Kits
- Spacecraft + Stage + ASE = 50,000 lbm (Except MPD)
- ΔV = 19,000 ft/s to GEO

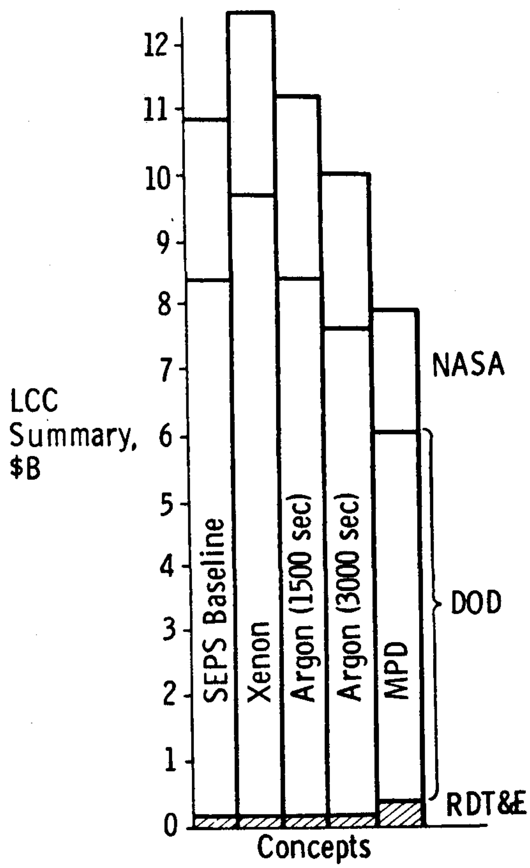
Five Concepts Identified	ISP, sec	Power, kW	Length, ft	Stages Req	Shuttle Flights
- Baseline SEPS (30 cm Hg ION)	3020	50 (Solar)	15.2	93	113
- Xenon ION Thruster	1500	50 (Solar)	15.0	106	120
- Argon ION Thruster	1500	60 (Solar)	15.0	106	120
- Argon ION Thruster	3000	76 (Solar)	15.0	98	114
- MPD	2400	200 (Nuclear)	45.0	53	122

LSS Conclusions for Electric Propulsion Vehicles

From the life cycle cost analysis the most economical electric propulsion stage is the MPD. When compared to the baseline SEPS the MPD stage is approximately 27% lower cost due to fewer stages and shorter transfer time. The Argon large inert gas thruster stage is approximately 10% lower cost. This is true for the DOD mission model and NASA mission model individually as well as the total. It is significant because it shows that the conclusions for DOD are unchanged by NASA. This is effectively a sensitivity analysis on the mission model since the DOD and NASA models differ in size, weight, frequency, and orbits.

Comparing the large inert gas thrusters to mercury ion show a slight cost advantage which in part is due to the reduction in thruster quantity. The development of these thrusters should not be on the basis of economic benefit, but on the basis of environmental impact (inert gas versus mercury) and spacecraft contamination. The use of Xenon propellant for orbit transfer is not justified due to its high cost and limited availability.

LSS Conclusions for Electric Propulsion Vehicles



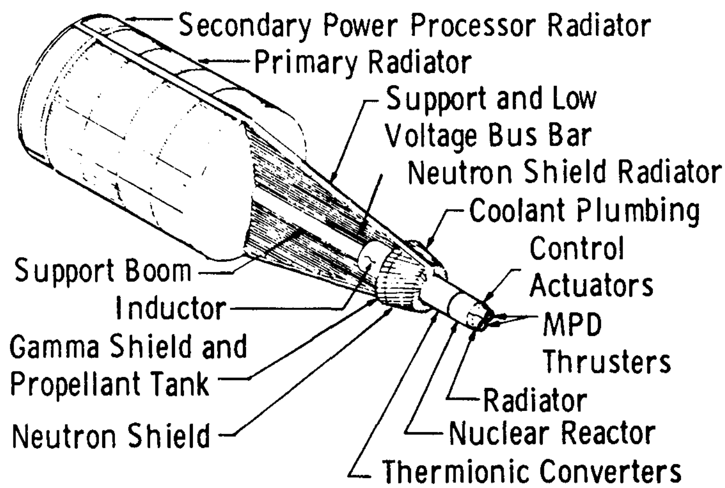
- Xenon Superior Performance Not Justified for High Traffic Orbit Transfer
  - High Cost
  - Limited Availability
- Argon Prime Candidate for Orbit Transfer Application
  - Reduced Environmental Impact
  - Reduced Spacecraft Contamination
- MPD Most Economical Electric Propulsion Concept Evaluated
- DOD Results Unchanged by NASA

Electric Propulsion Technology Requirements - MPD

The MPD key technology areas and stage configuration are summarized here. The thruster is the primary technology that should be pursued. Thrust level should be maximized by improving efficiency at the expense of weight. Related subsystems include power switching, energy storage, propellant management and thermal control, propellant flow control and isolation, and packaging of the system in Shuttle with the power supply.

Electric Propulsion Technology Requirements-MPD

Configuration Concept



Length = 45 ft (Including 8 ft Dia Tank)

Key Propulsion Technology

Thruster Demonstration
- Increase Thrust
- ISP Range 1500 to 3000 sec
- Maximize Efficiency at Expense of Weight
- Life Required ~15, 000 hr

Related Subsystems
- Power Switching
- Energy Storage
- Propellant Management and Thermal Control
- Propellant Control and Isolation
- Packaging of Complete System
- Nuclear Power

Notes:

Power 200 kW Nuclear Power at 36 lb/kW
Efficiency = 31% MPD, Processing - 90%
Thrust - 1.067 lb_f with Extra Thruster for Redundancy, ISP = 2400 sec
ΔV = 19, 000 ft/s, g-Level = 10^{-5}
Orbiter Capability = 40, 000 lbs at 425 n mi

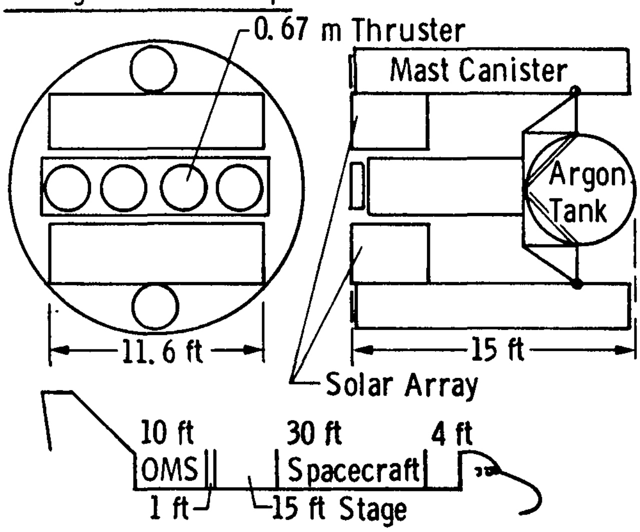
Flight Performance Reserve 2%
Transfer Time = 651 Days
Thruster Life Required = 15, 625 hr
Stage Weight = 35, 220 lbs
Mass Fraction = 0. 71
Payload Delivery = 78, 500 lbs

Electric Propulsion Technology Requirements - ION

The technology requirements for the Argon large inert gas thrusters and configuration concept are summarized here. This system has the potential to reduce contamination and environmental effects that exist with mercury. The key technology areas include propellant management, thermal control and isolation, and thruster duration testing in the larger size and its effects on the discharge chamber and cathodes both main and neutralizer.

Electric Propulsion Technology Requirements—ION

Configuration Concept



Key Propulsion Technology

Propellant Management and Thermal Control

Propellant Control and Isolation

Thruster Demonstration

- Size 67 cm
- Duration
  - Discharge Chamber
  - Cathodes, Main & Neutralizer

Notes:

Solar Power = 76 kW
Thruster Power = 17.15 kW, Eff = 42%
Thrust = 0.1088 lb_f Each, ISP = 3000 sec
Power Processing Eff = 90%
ΔV = 19,000 ft/s, g-Level = 10^{-5}
Orbiter Capability 50,000 lbs at 335 n mi

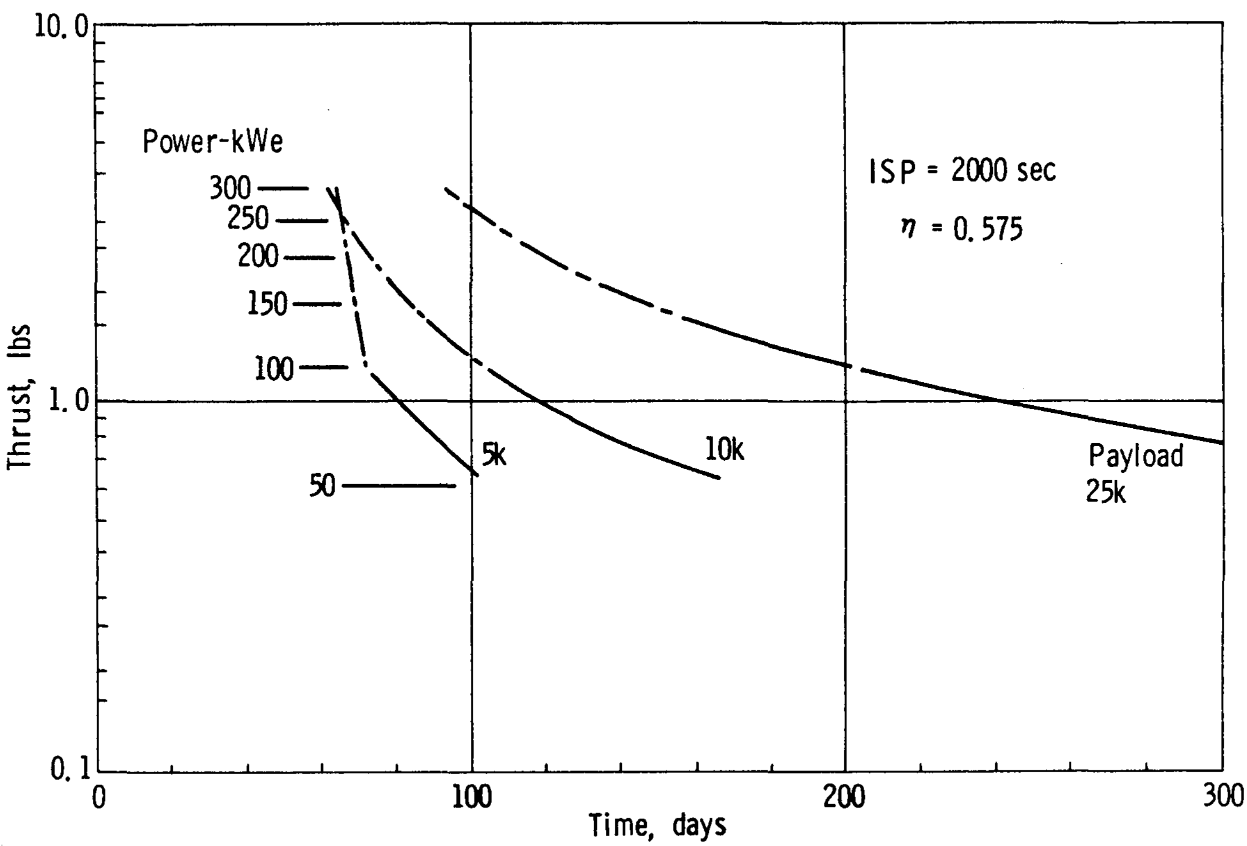
Flight Performance Reserve 2%
Transfer Time = 819 Days (Assuming 30% Solar Array Degradation Plus 5% Shadowing)
Engine Life = 14,600 hrs
Stage Weight = 12,670 lbs
Mass Fraction = 0.60
Payload Delivery = 28,600 lbs

Electric Propulsion Transfer Time from LEO to GEO

For DOD as well as NASA there are priority spacecraft and missions that must be delivered in the shortest possible time. Studies were conducted on the effect of specific impulse and efficiency on transfer time. The study results showed that a minimum transfer time of approximately 60 days to geosynchronous orbit is required to achieve any meaningful delivery capability such as 5000 lb. Shown here is an example of the study results for an Isp = 2000 sec and efficiency of 57.5%.

A significant cost factor in the LCC is the added spacecraft transfer time due to the low thrust of the electric propulsion system. To account for this it is necessary to both inflate and discount the dollar value of the spacecraft program. We followed DOD Directive 7041.3 on Economic Analysis in performing this task and found that this factor alone can be as high as 1/3 of the LCC.

Electric Propulsion LEO to GEO Transfer Time in Days



MPD Isp Optimization for 150,000 lb Spacecraft

The transfer time is determined by the system weight (stage + spacecraft) and thrust level. The higher the specific impulse the lower the stage weight or propellant weight. However, thrust decreases with increasing specific impulse by the following equation:

$$F = \frac{2n P}{g I_{sp}}$$

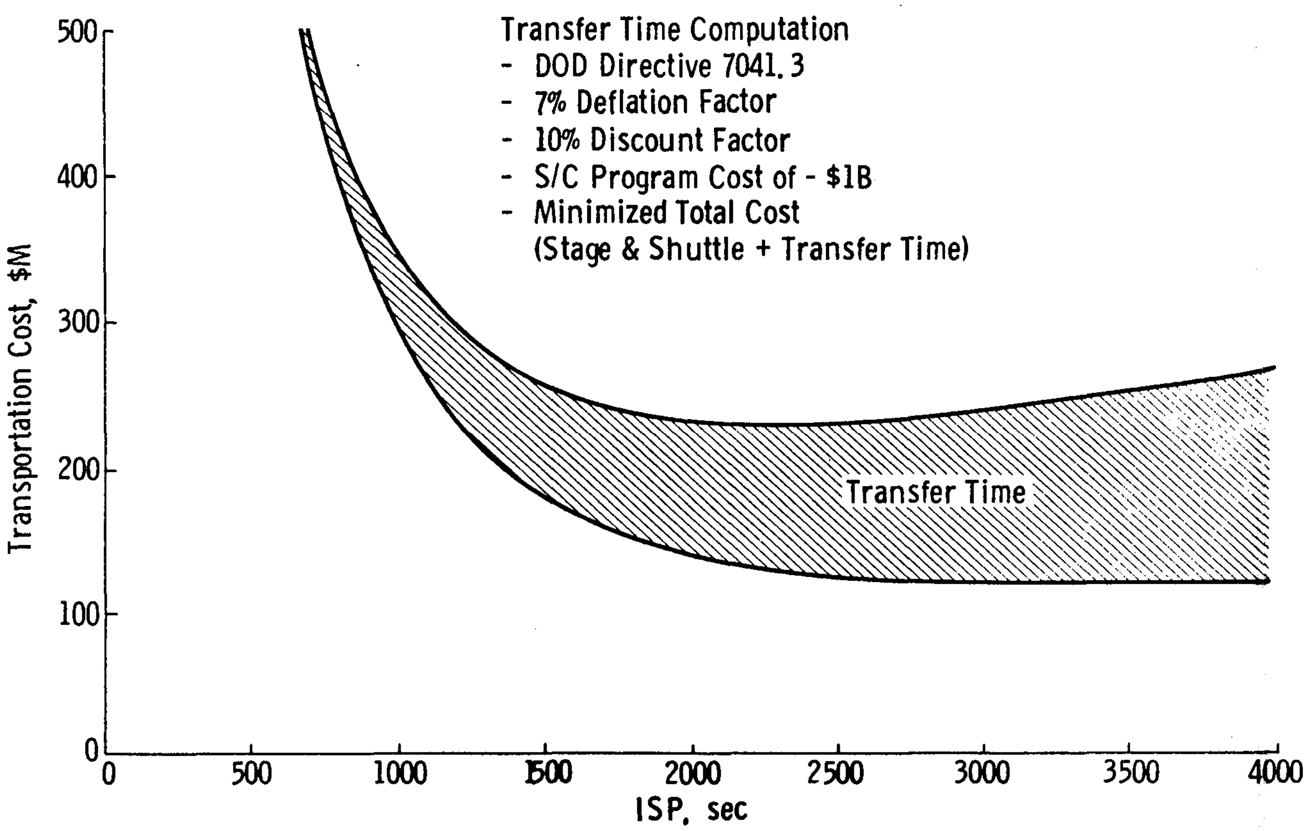
where:

- F = thrust
- P = electric power
- n = efficiency (converting electric power to thrust)
- I_{sp} = specific impulse
- g = acceleration due to gravity

This decrease in thrust increases transfer time and the effect on life cycle cost.

Because of this effect specific impulse optimization studies were performed on both the large inert gas thruster (Argon) stage concept and the MPD concept. These results show the optimum specific impulse to be in the 1500 to 3000 second range as indicated here for MPD. In this range the specific impulse is high enough to reduce shuttle flights yet low enough to prevent the transfer time from negating the economic benefits of electric propulsion.

MPD ISP Optimization for 150,000 lb Spacecraft



Electric vs Chemical Propulsion

Comparing electric propulsion to chemical propulsion has shown that economic advantages can be obtained when electric propulsion is utilized on very large delivery weight systems. To better define the advantage of electric propulsion and the spacecraft weight at which the advantage begins an analysis was performed as a function of spacecraft weight independent of any mission model. The results of this analysis are shown here and represent the transportation cost for the spacecraft and stage, stage unit cost, and transfer time effect. The RDT&E for the stage is not included and the cost is for a single spacecraft at the weight indicated being delivered to GEO. The stages used for comparison are electric MPD and cryogenic LO_2/LH_2. The results show the electric propulsion stage having significant economic advantage for spacecraft greater than 60,000 lbs. If the cost of transfer time is removed the advantage occurs at a lower spacecraft weight of approximately 15,000 lbs. This saving comes primarily from reduced stages and shuttle flights due to the higher specific impulse yielding a lower weight and volume for high impulse requirements.

Electric Versus Chemical Propulsion