Modular, Ion-Propelled, Orbit-Transfer Vehicle

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The design approach is presented for a modular, ion-propelled, orbit transfer vehicle (OTV). The OTV consists of a propulsion module that can be returned to Earth via the Shuttle for refueling and refurbishment, and a reusable power bus that mates to the spacecraft payload and remains in orbit. The technologies required to make the OTV concept both technically and economically feasible are identified. As an example of how the OTV could be applied, the NAVSTAR/Global Positioning System (GPS) Block 3 mission is examined using both conventional (expendable) chemical stages and the ion propulsion OTV. The OTV approach is shown to be particularly attractive, from a cost standpoint, for the specific application to GPS. The high specific impulse provided by ion propulsion is shown to result in a new reduction of $145 to 195 million in overall cost for the GPS Block 3 mission as compared with the cost using the Payload Assist Module (PAM) D-II chemical propulsion stage. This reusable OTV approach is believed to be equally attractive for other missions that require multiple launches.

Nomenclature

A	= altitude
C	= cost
d_f	= drag penalty
e_f	= eclipse penalty
F	= thrust
f_r	= propellant reserve
g	= sea-level gravitational acceleration
I_{sp}	= specific impulse
J_B	= beam current
J_o	= sum of neutral loss and neutralizer propellant flow rate (= 0.184 A)
M	= mass
$\dot{m}_T$	= total propellant flow rate
n_{pl}	= number of spacecraft payloads
n_{rt}	= number of round trips
n_t	= number of thrusters
n_v	= number of vehicles
P_{IN}	= thruster power
P_{TOT}	= total power
R	= ratio of beam-to-total accelerating voltage
r_f	= radiation degradation
T_{RT}	= round-trip time
V	= orbital velocity
V_B	= beam voltage
v_{pt}	= volume of propellant tank
ΔV	= velocity increment
α	= specific mass
γ	= thrust-loss factor (= 0.97)
$\Delta\theta$	= inclination change
η_e	= thruster electrical efficiency
η_t	= total thruster efficiency
η_{tot}	= total efficiency
ρ_{xe}	= density of xenon at 29 MPa (2244 Kg/m^3)
θ	= inclination

Subscripts

1	= initial
2	= final
ACS	= attitude control subsystem
BAT	= battery
CA	= concentrator array
CR	= Shuttle cradle
OTV	= orbit transfer vehicle
PL	= payload
POW	= power subsystem
PM	= propulsion module
PPU	= power processor unit
PP	= propellant
PROP	= propulsion subsystem
PT	= propellant tank
PTCU	= propellant tankage and control unit
RAD	= radiator
REF	= refurbishment
REG	= regulator
ST	= supporting structure
STS	= space transportation system
T	= thruster
TC	= tracking and control
TTC	= tracking, telemetry, and command
VL	= valve

Presented as Paper 86-1394 at the AIAA/ASME/SAE/ASEE 22nd Joint Propulsion Conference, Huntsville, AL, June 16–18, 1986; received July 31, 1986; revision received April 22, 1987.

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Introduction

WITH the advent of the Space Transportation System, development of a low-cost means of transferring spacecraft from low-Earth orbit (LEO) to higher orbits has become a major objective. At present, this transfer task is accomplished using either a solid or liquid chemical propulsion system. However, because of projected advances in both ion propulsion and in concentrator-solar-array technologies, it appears that a high-performance, electric orbit-transfer stage could be developed in the near term. This new stage could be significantly less expensive and operationally more flexible than conventional stages when used for multiple-launch scenarios.

In the material that follows, it is shown that when multiple missions are required, a reusable, ion-propelled, orbit transfer vehicle (OTV) can provide the means to transfer spacecraft to geosynchronous (GEO) and half-geosynchronous orbit (HGEO) at a much lower cost than other propulsion stages. Although several previous studies[1-3] have demonstrated the attractiveness of the electric propulsion OTV concept, none have addressed the practical aspects of developing and utilizing such a vehicle. The present study extends the previous work by focusing on the technologies and operational scenarios required to make the reusable OTV concept both technically and economically feasible.

Modular Orbit Transfer Vehicle

The modular OTV concept proposed herein consists of a reusable power bus (solar arrays; power processing units; batteries; thermal control system; tracking, telemetry and command; and attitude control system) that remains in orbit and a propulsion module (ion thrusters, propellant feed system, and propellant storage tanks) that is detached and returned to Earth for servicing after each mission. This configuration is depicted in Fig. 1. The three-axis-stabilized OTV is propelled to high-altitude orbits and returned to the Shuttle altitude by means of a high-thrust, high-efficiency, xenon-ion-propulsion system, using a slow-ascent, spiral trajectory technique[4] (Fig. 2). The required electrical power would be provided by relatively lightweight, radiation-resistant concentrator solar arrays.

The operational scenario is illustrated in Fig. 3. The spacecraft payload would normally be attached to the propulsion module during Shuttle launch. Once in LEO, the propulsion module and payload would be mated (either manually or via the Shuttle Remote Manipulator System) to the power bus, which would already be in orbit. (Since it is reusable, the power bus need only be launched once.) Prior to this operation, the spent propulsion module used during the previous flight of the power bus would have been detached for return to Earth to be refurbished and refueled.

After the payload, propulsion module, and power bus have been mated, the vehicle would begin a slow-ascent spiral trajectory to the required higher mission orbit, thrusting continuously in sunlight and coasting during eclipse periods. Approximately 9% of the transfer time would be spent in eclipse, and drag forces would increase the ΔV requirement by approximately 0.1%. After deploying the payload in its required orbit, the OTV would begin a spiral transfer back to LEO where it would rendezvous with the next payload.

Since the propulsion module would be returned to Earth after each round trip, the life of the OTV would only be limited by the life of the power bus, which, in turn, would only be limited by the design life of the concentrator solar arrays and electronics. The concentrator and electronics would be subjected to hazardous radiation from the Van Allen belts,[5-7] which would have to be traversed twice during each round trip. This harmful exposure would limit the power bus to seven round trips, the design point established for this study.

The ion-propulsion OTV concept proposed herein could result in a large savings in Shuttle launch costs relative to conventional chemical stages. The high specific impulse provided by ion propulsion reduces the propellant mass, and the modular approach eliminates much of the launch mass of a completely new stage each time a spacecraft payload is deployed. The modular approach also allows the ion thrusters to be refurbished or replaced as required (facilitating the high-thrust operation of the thrusters) and eliminates the need for difficult in-orbit transfer of high-pressure xenon gas. The high-thrust operation of the ion thrusters reduces the number of actual thrusters required for any given mission, resulting in lower stage costs. In the material that follows, a more detailed description of the subsystems that would comprise the electric OTV is presented.

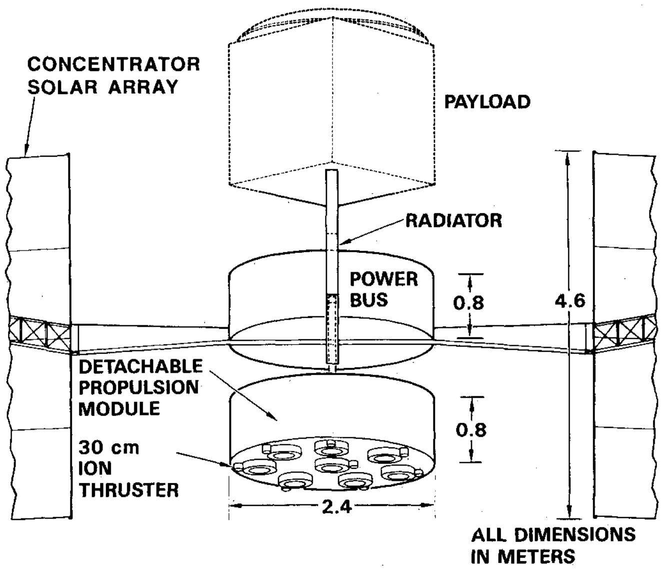

Fig. 1 Modular orbit transfer vehicle.

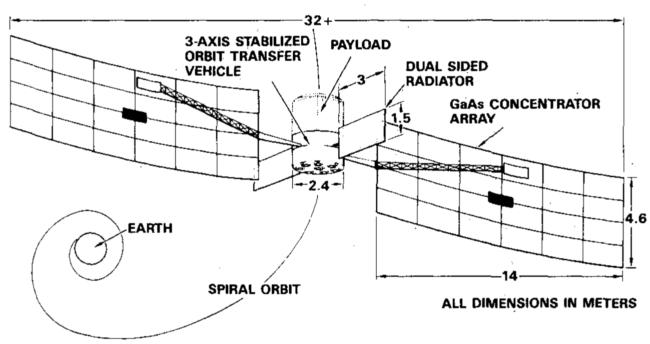

Fig. 2 Ion-propelled orbit transfer vehicle.

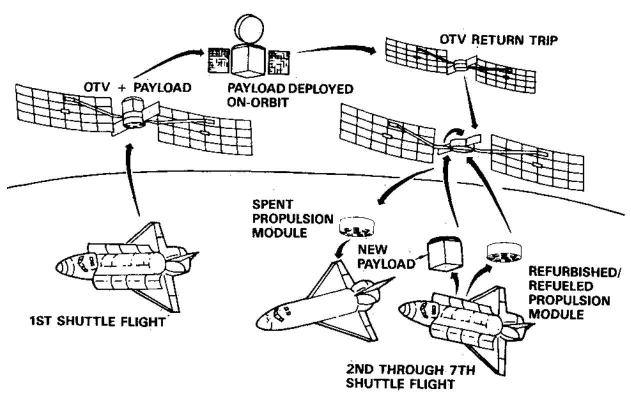

Fig. 3 Reusable OTV operational scenario.

Power Bus Subsystem

The power bus subsystem consists of the concentrator solar arrays; thruster power supplies; batteries; tracking, telemetry, and command hardware; control; and thermal radiators. Descriptions of these subsystems are provided below.

Concentrator Solar Array

Studies[8] have shown that flat-panel arrays that use silicon solar cells will not meet the requirements of the OTV because of the large amount of cell degradation that occurs during passage through the Van Allen radiation belts.[5-7] Concentrator-type solar arrays using gallium-based solar cells are believed to be necessary in order to minimize the radiation damage. Using existing techniques, it is believed that a high-performance concentrator array could be developed by the early 1990's, following a focused technology-development effort. The advanced concentrator system is projected to degrade by no more than 5–10% over its life and to have a specific mass of approximately 12 kg/kW. The latter figure includes the mass associated with the solar cells, the concentrator, and the deployment mechanism.

In order to achieve these performance characteristics, the concentrator must employ advanced, high-performance solar cells with a conversion efficiency approaching 30%. In addition, the cells must either be shielded from significant amounts of radiation or be capable of being thermally annealed (similar to the way today's gallium cells can be annealed) to reduce the effects of particulate irradiation. Such cells could be developed within the next decade. The concentrator concept illustrated in Fig. 2 would employ these advanced cells. It would provide shielding for the cells, while simultaneously providing for continuous high-temperature operation of the cells in order to anneal them thermally on a continuous basis. The average degradation per trip for such an array would need to be in the range of 1–2% for the system to be attractive. Obviously, this is a very challenging goal.

The development of the solar concentrator system is critical to the feasibility of a cost-effective electric OTV. The concentrator system will require not only very-high-efficiency solar cells, but also its own thermal control and integral radiator system, as well as development of an optics system that can withstand the numerous traverses of the radiation belts without incurring major damage to reflective surfaces.

Thruster Power Supply

Power processors would be required to condition the electrical power provided by the solar concentrators to the several forms required by the xenon ion thrusters. The power supply necessary to operate an inert-gas thruster is substantially simplified as compared to that required for mercury thrusters because the propellant does not have to be vaporized or heated to maintain it in vapor phase. Simplified power processor technology has been successfully demonstrated for xenon propellant. A breadboard-model power supply has recently been developed that operates with an efficiency of about 92% at the 1.4 kW power level.[9] The flight-packaged specific mass of this unit is estimated to be about 7.8 kg/kW. As compared with the Functional Model Power Processor Unit (FM/PPU) of the J-series mercury ion thruster, the number of modules within the breadboard power supply has been reduced from 12 to 7. The total parts count (including telemetry) in the partially redundant FM/PPU is approximately 4000, while an improved, nonredundant design contains about 2500 parts. The breadboard-model power supply recently developed contains about 600 parts with no redundancy or telemetry. Addition of telemetry would add approximately 200 parts. Of the approximately 600 parts used in the 1.4-kW breadboard model, only about six have not been space qualified.

High-performance power modules for ion thrusters have already been demonstrated at the 10- and 25-kW power levels.[10,11] It is believed that the simplified power-processing technology demonstrated at the 1.4-kW power level can be extrapolated to higher power in a straightforward manner. At the 10-kW power level, an overall efficiency of about 95% and a specific mass of about 4.5 kg/kW is projected for simplified power processors.

Battery

Batteries would be used for safe-hold while the OTV is in eclipse. The maximum eclipse period would be around 35 min and thus the cool-down of the major heat-producing elements of the vehicle would be small. The radiator design would include thermal diodes to cut off the heat leak during these periods to maintain the electronics temperatures near the desired conditions with minimal heater power. In addition, the guidance and control system would be a zero-momentum system and thus would have minimal power system drain during eclipse. The major drain on the battery would be the tracking, telemetry, and command subsystem (TT&C) that must be kept up during all periods of operation.

The battery would require a large cycle life and thus would most likely be a NiH_2 or an advanced NaS battery. Under the conditions stated above, it is estimated that a battery mass of approximately 45 kg would be sufficient. However, if the payload carried by the OTV requires additional power inputs, the battery mass may have to be significantly increased.

Tracking, Telemetry, and Command

The TT&C subsystem requirements are envisioned to be extensive and similar to those required by a modern communications satellite. This system would provide subsystem performance, sensor, status, and attitude information necessary for proper operation of the OTV via an rf downlink and would likewise receive, process, and execute commands received from the ground-based control center. For sizing purposes a requirement for approximately 200–250 commands and a similar number of telemetry signals was assumed, all of which would need monitoring. The mass of such a system was estimated to be approximately 70 kg.

Control

The OTV control subsystem is envisioned to be a zero-momentum system that uses inputs from various sensors to maintain the vehicle pointed along the orbit vector, the concentrator panels normal to the plane of the ecliptic, and the radiator panels in the plane of the ecliptic. Such a system would require the minimum amount of hardware; for the present study a mass of approximately 35 kg was assumed. Directional control would be achieved by gimballing the ion thrusters. It would also include a sun-tracking system to enable pointing of the solar concentrators toward the sun at all times. Since the OTV attitude control would be accomplished by means of engine gimballing, all control functions must occur during engine operation. Therefore, during the eclipse periods, the vehicle would be allowed to drift uncontrolled. However, the longest period of eclipse would be around 35 min so that significant vehicle attitude errors would occur only early in the mission when the vehicle is in a relatively low-altitude orbit. The errors caused by this approach would require some excess propellant to correct, but they would not be of sufficient magnitude to require a more costly and complex momentum-based attitude-control system. The control algorithms required for the envisioned system would most certainly be one of the major development items necessary for such an OTV.

Thermal Radiator

The radiator subsystem is required to remove the excess heat from the power electronics units of the ion thrusters. Depending on the actual conversion efficiency of the power supplies, the amount of heat to be rejected would be about 4 kW. The concept used in the present study is that of a thermal bus, wherein the power supplies are mounted to a base plate that is maintained at a near-constant temperature by means of variable-conductance heat pipes. These heat pipes conduct the excess heat to the radiator where it is rejected from the vehicle. This thermal bus is presently under development and should be available in a time frame commensurate with development of an electric OTV.

The design of the thermal bus and radiator system would include a thermal switch, or diode, which would prevent excessive cooling of the thermal bus when the vehicle is in eclipse and the ion-thruster power supplies are not operating. The radiator itself would require development for this application with a goal to reject approximately 90 W/kg of radiator mass and 275–325 W/m^2 of area when used as a dual-sided radiator. When installed on the OTV the radiator would be oriented to be edge-on to the sun at all times. (The exception occurs near the solstice periods when the OTV orbit is at its maximum deviation from the plane of the ecliptic.) Radiator degradation due to space radiation should be held to a level of approximately 2% per trip.

Propulsion Module Subsystem

The detachable propulsion module houses the xenon-ion-propulsion subsystem consisting of the thrusters, propellant tanks, and propellant control units. The basic xenon-ion-propulsion technology required for a reusable OTV has already been demonstrated at the laboratory-model level for north-south stationkeeping of communication satellites. A high-performance, xenon-ion-propulsion subsystem has recently been developed[9] and evaluated in a cyclic wear-mechanism test. This 1.4-kW unit defines the state of the art in inert-gas-ion-propulsion technology; it combines the heritage of the 30-cm J-series mercury ion thruster with recent advances in discharge chamber design and performance along with substantial simplification and performance improvements in the power processor technology.

Thruster

The analysis performed under the present study assumed a 30-cm-diam thruster equipped with a three-grid ion extraction assembly. This technology has been demonstrated at the 1.5 A beam current level (1.4-kW power level) using laboratory-model hardware. Operation of a modified 30-cm-diam J-series thruster on xenon at power levels up to 10.7 kW has recently been demonstrated at NASA's Lewis Research Center[12] using a two-grid ion extraction assembly. Therefore, the primary technology that must be developed for the high-thrust-density OTV application is three-grid ion extraction assemblies operational at power levels of up to about 9 kW. Three-grid ion-extraction technology has been demonstrated in short-duration laboratory operation on mercury propellant at power levels up to about 7 kW and net-to-total voltage ratios as low as $R = 0.4$.[13]

Thruster lifetime at high power levels is not verified. However, short-term erosion-rate measurements[9,14] performed at the 1.4-kW power level (1.5 A beam current) have resulted in a projected thruster lifetime (based on ion erosion of the screen electrode to 50% of its initial thickness) of about 25,000 h, or nearly 40,000 A-h. With total trip times on the order of 145 days, the 40,000 A-h figure translates to an upper limit on beam current of about 11.5 A. The optimized OTV performance requires only about 8 A of beam current, and so there is a comfortable margin in anticipated lifetime. It should be pointed out that if internal erosion of the thruster were an issue, a substantial reduction in ion sputtering (a factor of two) could probably be realized by the addition of a small amount of nitrogen to the xenon.[13,15]

Propellant Tankage and Control Units

The propellant-handling system is similar to the technology developed for conventional attitude-control and stationkeeping applications with chemical propulsion.[9] Xenon would be stored in several tanks as a high-pressure gas (up to about 29 MPa at beginning of life) with a density about twice that of water. The tanks would be heated to maintain the gas above its critical temperature of 16.6°C. Once the valves are opened the rate at which xenon flows into the thrusters would be controlled in an open-loop manner using a redundant, pressure-regulated system and flow impedances. This simplified gas-control approach has been demonstrated[9] using a modified pressure regulator flight-qualified for the Mars Viking program. The majority of the system components are available as flight-qualified commercial hardware.

High-pressure gas storage and handling technology is already in use on communication satellites and the operation of xenon-ion thrusters under constant-flow conditions has been demonstrated. Therefore, the OTV application appears to be straightforward. Although the yearly amount of xenon that would be consumed during four round trips of the OTV is about equal to the total production rate in the U.S., suppliers have indicated that production can be increased to meet the demand. In spite of its rarity, the quantity of xenon that would be needed on a yearly basis is only about 1 part in 10^8 of that contained in the Earth's atmosphere.

Structural Subsystem

In the next section the major structural components (supporting structure, the mechanisms used to deploy the solar arrays, and the thermal radiators) are discussed.

Supporting Structure

For the purposes of this study it was assumed that the vehicle structure would require approximately 19% of the total OTV mass. Although determination of the exact structural mass would necessitate a much more detailed study than the present one this estimate is generally thought to hold true for space vehicles of this type. A cradle would be needed to carry the OTV in the Shuttle bay. Although the cradle would have a mass of about 1680 kg, it was assumed that the cradle mass for the electric OTV would not be significantly different from that required for a conventional chemical system. Thus, cradle mass is not a driving factor in the present study.

Deployment Mechanisms

The OTV vehicle, as envisioned herein, calls for the deployment of two major elements: the concentrator solar arrays and the heat pipe radiators. The concentrator system is conceptually made up of a series of hinged mirror elements that unfold and lock into place to form the concentrator optics. The radiator deployment depends upon use of flexible heat pipe elements currently under development. An initial review of the packaging requirements for the proposed OTV indicates that it would be possible to launch the entire OTV in a single Shuttle launch.

Baseline Performance Evaluation

The following section presents an example of how the ion-propulsion OTV could be applied to the GPS Block 3 mission planned for the early 1990's, resulting in significant cost savings relative to a conventional PAM D-II stage approach. The basis for the comparison is the total cost to deploy 28 spacecraft at a rate of four per year for seven years.

The ion OTV was sized to deliver the GPS Block 3 spacecraft with a projected mass of 1135 kg from a 28.5 deg Shuttle orbit (200 km) to a 55 deg half-geosynchronous orbit (20,000 km). The outbound transfer time was limited to 90 days to be able to replace a failed satellite within a reasonable length of time. Also, the use of technologies that could be developed by the early 1990's with moderate extrapolation of the state of the art was emphasized to ensure a smooth development program in terms of final design and test. This, of course, would not preclude use of the concentrator array that could be developed by the early 1990's if a focused technology-development effort were initiated. The mission parameters are summarized in Table 1.

Table 1 Summary of GPS mission parameters

Parameter	Value
Initial altitude A_1	200 km
Final altitude A_2	20,000 km
Initial inclination θ_1	28.5 deg
Final inclination θ_2	55 deg
Drag penalty d_f	0.1%
Eclipse penalty e_f	9%
Mission ΔV	5525 m/s

Using these constraints and the additional OTV inputs listed in Table 2, the mission-analysis model summarized in the Appendix was used to calculate the mass, transfer time, and power requirements of the OTV, and the total mission cost. The optimized system is summarized in Table 3. This system has an outbound trip time of 90 days, a return trip time (HGEO to LEO) of 53 days, a dry mass of 2170 kg, and a total mass of 3211 kg, requires 8 ion thrusters operating at a thrust of 400 mN and a specific impulse of 3600 s, and has a total power requirement of 76 kW, which is furnished by two 65 m^2 GaAs concentrator arrays. Since the power bus is constrained by Van Allen radiation effects to a maximum of seven round trips, a total of four OTV's must be built. On the first Shuttle launch the entire 3211-kg OTV must be delivered to LEO. However, because of the modular design, on Shuttle trips 2–7 only the 1470-kg propulsion module is delivered. This results in a significant reduction in launch costs on the subsequent Shuttle trips

The recurring cost of delivering 28 GPS spacecraft to HGEO using the OTV is compared in Table 4 with the equivalent cost using the expendable PAM D-II stage (that includes a separate apogee stage for orbit circularization). The results indicate that use of the OTV can result in a total recurring cost savings of $245 million with 98% of the savings in the Shuttle launch costs. This savings, however, would be partially offset by a nonrecurring OTV research and development cost of $50–100 million for a net savings of $145–195 million. This assumes that a post-1988 Shuttle launch fee of $87 million in fiscal year 1986 dollars applies, that the OTV is a mass-critical payload (i.e., the Shuttle launch charge is determined by the OTV mass, not length), and that the cost to track and control the OTV's over the seven-year mission is approximately $10 million. This also assumes that four OTV's are purchased and that the recurring cost of each OTV would be approximately equal to the recurring cost of a typical com-

Table 2 Summary of baseline GPS OTV sizing parameters

Subsystem/component	Parameter	Value
Propulsion subsystem		
Thrusters	Mass M_T	14.8 kg each
Thruster valves	Mass M_{VL}	1.0 kg each
Regulation system	Mass M_{REG}	3.9 kg
Propellant tanks	Mass M_{PT}	11.6 kg each
	Volume v_{PT}	0.04 m^3 each
	Propellant reserve f_r	4%
Power subsystem		
Concentrator arrays	Specific mass α_{CA}	12.3 kg/kW
	Radiation degradation r_f	10% after 7 round trips
Power processing units	Specific mass α_{PPU}	4.5 kg/kW
	Efficiency η_{PPU}	95%
Battery	Mass M_{BAT}	45 kg
Thermal radiators	Specific mass α_{RAD}	11.4 kg/kW
Tracking, telemetry, & command subsystem	Mass M_{TTC}	68 kg
Attitude control subsystem	Mass M_{ACS}	36 kg
Supporting structure	Mass M_{ST}	10% of total OTV mass
Shuttle cradle	Mass M_{CR}	1678 kg
Payload	Mass M_{PL}	1135 kg

Table 3 Mass summary and general characteristics of the optimized OTV for GPS mission

Subsystem/component	Mass, kg	Characteristics
Propulsion module		Dimensions: 2.4 m diam 0.8 m height
Propulsion subsystem ion thrusters	119	Number: 8 Thrust: 400 mN Specific impulse: 3600 s Efficiency: 79% Input power: 9.0 kW
Propellant tankage and control unit	163	Number of 0.04 m^3 propellant tanks: 13
Supporting structure	147	
Propulsion module dry mass	429	
Xenon propellant	1041	Propellant reserve: 4%
Propulsion module total mass	1470	
Power bus		
Power subsystem		Dimensions: 2.4 m diam 0.8 m height
GaAs concentrator arrays	1034	Number: 2 (65 m^2 panels) Total power: 76 kW
Power processing units	341	Efficiency: 95%
Battery	45	
Thermal radiators	43	Number: 2 (4.5 m^2 panels) Heat rejected: 3.8 kW
Tracking, telemetry, and command subsystem	68	
Attitude control subsystem	36	
Supporting structure	174	
Power bus total mass	1741	
OTV dry mass	2170	
OTV total mass	3211	Trip time Outbound: 90 days Return: 53 days

Table 4 Cost to deploy 28 GPS satellites using electric and chemical stages

OTV		
Propulsion modules plus power bus @ $40–50 million per vehicle (C_{OTV})	4 × $45 million (avg) =	$180 million
Xenon propellant @ $570/kg ($C_{PP}$)	24 × 104/kg × $570/kg =	$14 million
Ion thruster refurbishment/replacement @ $75,000 per thruster ($C_{REF}$)	24 × 8 thrusters × $75,000 =	$14 million
OTV orbital support (C_{TC})		$10 million
Shuttle launch @ $87 million post = 1988 rate (C_{STS})		
1st Shuttle load: propulsion module plus power bus	4 × 3211 kg = 12,844 kg	
2nd–7th Shuttle load propulsion module alone	24 × 1470 kg = 35,280 kg	
Cradle	1,678 kg	
	Total Shuttle load: 49,802 kg	
	$87 million × (49,802 kg/29,484 kg) =	$148 million
	Total recurring cost	$366 million
	Nonrecurring research & development cost	$50–100 million
	Total OTV cost	$416–466 million
PAM D-II		
PAM D-II stage	28 × $7 million =	$196 million
Apogee stage	28 × $1 million =	$28 million
Shuttle launch		
PAM D-II stage	28 × 3720 kg = 104,160 kg	
Apogee stage	28 × 907 kg = 25,396 kg	
Cradle	1,678 kg	
	Total Shuttle load: 131,234 kg	
	$87 million × (131,234 kg/29,484 kg) =	$387 million
	Total PAM D-II cost	$611 million
	Recurring cost differential	$245 million
	Total cost differential	$145–195 million

munications satellite, or about \$40–50 million. Thus, if each OTV could be designed to survive 10 round trips through the radiation belts with minimal degradation, such that only three vehicles would be needed to meet mission requirements, an additional savings of \$40–50 million would be possible. The cost model is described in the Appendix.

It should be noted that the 26.5 deg plane change is very costly in terms of trip time and OTV mass. Elimination of the plane change by launching the Shuttle directly into a 55 deg orbit can reduce the trip time by 28% and decrease the OTV mass by 8%. Of course, the Shuttle launch cost would increase by 5% as well. It should also be noted that a geosynchronous mission ($\Delta V = 6000$ m/s) requires a velocity change of only 9% (475 m/s) more than the GPS half-geosynchronous mission ($\Delta V = 5525$ m/s). Therefore, geosynchronous spacecraft could also benefit from the development of a high-performance electric OTV.

Sensitivity Analysis

The OTV design presented in the present paper is based on near-term technologies that could be developed with little risk and moderate extrapolation on the state of the art. The major exception is the concentrator, which will require a more focused development effort. In conducting this study, it was assumed that a 12 kg/kW concentrator that does not degrade more than 10% after seven round trips through the radiation belts could be developed by the early 1990's. However, since the concentrator drives the performance of the OTV design, analyses were performed to evaluate sensitivity of the GPS OTV configuration and its performance to expected variations in the concentrator design parameters. Two worst cases were examined: 1) a concentrator with a specific mass of 14 kg/kW and 2) a concentrator with a total degradation of 15% after seven round trips through the radiation belts.

Increasing the specific mass of the concentrator from 12 to 14 kg/kW increased the total mass of the OTV by 260 kg, the outbound trip time by six days, and the total recurring cost by \$10 million. Increasing the radiation degradation factor from 10 to 15% increased the OTV mass by 100 kg, the outbound trip time by two days, and the total recurring cost by \$4 million. Clearly for both cases the proposed OTV concept would still result in a large cost savings relative to conventional chemical stages with little increase in outbound trip time. In fact, sensitivity analyses confirm this conclusion for the other system parameters as well.

Table 5 shows outbound trip time and cost sensitivities calculated for the range over which the input parameters can be expected to vary. In all cases, the sensitivities are relatively small in comparison to the overall cost savings; the OTV configuration is relatively insensitive to projected variations in the baseline design parameters. It should be noted that the sensitivities to net-to-total accelerating voltage ratio R are negative. This means that the requirement on ion-optics performance could be relaxed (by going to higher values of R), resulting in shorter transfer times and lower deployment costs. However, increasing the net-to-total accelerating voltage ratio would require more power and a larger array, so that the tradeoff may be one of diminishing returns.

It should also be noted that failure of one or more thrusters in transit would not compromise completion of the OTV mission. Failure of one thruster would increase the outbound trip time by only 13 days, and failure of two thrusters would increase it by 20 days. Both increases would surely be acceptable under the special circumstances.

Conclusions

A design approach for a modular, ion-propelled, orbit transfer vehicle that can provide the means to transfer spacecraft to geosynchronous and half-geosynchronous orbit has been presented. Using the GPS Block 3 mission as an example, it is shown that such an approach could be significantly less costly and operationally more flexible than expendable chemical stages because of the large number of GPS spacecraft that must be launched. Successful development of the OTV will

Table 5 Sensitivity of GPS trip times and deployment costs to input parameters

Parameter	Nominal value	Trip time sensitivity	Cost sensitivity
Drag penalty	0.1%	≈ 0	≈ 0
Eclipse penalty	9%	0.9 days/%	≈ 0
Thruster mass	14.8 kg	0.2 days/kg	\$1.7 million/kg
Number of thrusters[a]	8	≈ 0	\$4 million/thruster
R factor[b]	40%	− 0.1 days/%	− \$0.5 million/%
Propellant tankage and control unit mass	163 kg	0.03 days/kg	\$0.1 million/kg
Propellant reserve	4%	1.3 days/%	\$1.8 million/%
Concentrator array specific mass	12.3 kg/kW	3 days-kW/kg	\$5 million-kW/kg
Concentrator array radiation degradation	round trips 10% of total	1.4 days/%	\$4.2 million/%
Power processing unit specific mass	4.5 kg/kW	0.1 days-kW/kg	\$0.2 million-kW/kg
Power processing unit efficiency	95%	− 0.9 days/%	− \$1.6 million/%
Battery mass	45 kg	0.07 days/kg	\$0.2 million/kg
Thermal radiator specific mass	11.4 kg/kW	≈ 0	≈ 0
Tracking telemetry and command subsystem mass	68 kg	0.05 days/kg	\$0.1 million/kg
Attitude control subsystem mass	36 kg	0.03 days/kg	≈ 0
Structural mass	10% of total OTV mass	1.4 days/%	4.2 million/%
Cradle mass	1678 kg	—	\$0.08 million/kg
Payload mass	1135 kg	0.03 days/kg	\$0.03 million/kg
OTV cost	\$45 million	—	\$4 million/\$1 million
Propellant cost	\$570/kg	—	\$0.03 million-kg/\$
Thruster refurbishment cost	\$75,000	—	\$0.2 million/\$1,000
Orbital support cost	\$10 million	—	\$1 million/\$1 million
Shuttle launch cost	\$148 million	—	\$1.9 million/\$1 million

[a]Constant total power. [b]Reoptimized I_{sp}.

depend on many factors, the most important being near-term development of lightweight and radiation-resistant GaAs concentrator solar arrays. Other areas include qualification of the xenon ion thruster at the high thrust levels that have been demonstrated and development of appropriate orbit-trajectory analysis and control algorithms. All of the relevant technologies are developed or are on the "drawing boards" and could be developed by the early 1990's with proper focusing of research and development efforts.

Appendix: Thruster Performance Model and OTV Sizing Algorithm

Table A1 presents the thruster performance model used in the OTV analysis. The model was derived using thruster performance measurements obtained at Hughes[9] and ion-optics performance results obtained at NASA Lewis Research Center.[12] The thruster model is based on a 25-cm-diam ring-cusp discharge chamber that operates at an ion-production cost of 130 eV/ion and a neutralizer-coupling voltage of about 20 V. This level of performance has been demonstrated at the 1.4-kW level using a three-electrode ion-extraction assembly. Similar performance has been demonstrated at power levels up to 10.7 kW using a 30-cm-diam J-series ion thruster modified for operation on xenon.

Sizing Algorithm

The general characteristics of the OTV were obtained in terms of mission parameters by use of the simple analytical procedure summarized below:

1) For a given I_{sp} and R factor, calculate F and η_{tot} using the thruster model described above.

2) Calculate the total power required from the equation

$$P_{TOT} = \frac{gI_{sp}\, n_t F}{2\, \eta_{tot}}$$

3) Calculate the equivalent field-free, low-thrust velocity increment for orbit transfer (corrected for drag) from the Edelbaum equation:[4]

$$\Delta V = (1 + d_f)\left[V_1^2 + V_2^2 - 2V_1 V_2 \cos\left(\frac{\pi \Delta\theta}{2}\right)\right]^{1/2}$$

where V_1 and V_2 are the initial and final velocities of the orbits.

4) Estimate M_{OTV}, the total mass of the OTV.

5) Calculate the mass of propellant required for the round trip using the rocket equation

$$M_{PP} = M_{PP1} + M_{PP2}$$

where $M_{PP1} = (1 + f_r)(M_{OTV} + M_{PL})\,(1 - e^{-\Delta V/gI_{sp}})$ is the propellant required for the outbound trip, and $M_{PP2} = (1 + f_r)(M_{OTV} - M_{PP1})\,(1 - e^{-\Delta V/gI_{sp}})$ is the propellant required for the return trip.

6) Calculate the round-trip time from

$$T_{RT} = \frac{gI_{sp}\, M_{PP}\, (1 + e_f)}{n_t F}$$

7) Calculate the mass of the propulsion subsystem from

$$M_{PROP} = M_{PTCU} + n_t\, M_T$$

where $M_{PTCU} = M_{REG} + n_t\, M_{VL} + n_{pt}\, M_{PT}$ is the mass of the propellant tankage and control unit, and $n_{pt} = M_{PP}/\rho_{xe} v_{pt}$.

8) Calculate the mass of the power subsystem from

$$M_{POW} = M_{CA} + M_{PPU} + M_{BAT}$$

where $M_{CA} = P_{TOT}\, \alpha_{CA}/(1 - r_f)$ is the mass of the concentrator arrays, $M_{PPU} = P_{TOT}\, \alpha_{PPU}$ is the mass of the power processing units, and $r_f = 0.03 + 0.01\, n_{rt}$ is the solar-array power degradation factor (over the life of the OTV).

9) Calculate M_{RAD} from the equation

$$M_{RAD} = P_{TOT}\, \alpha_{RAD}\, (1 - \eta_{PPU})$$

10) Calculate M_{ST} from the equation

$$M_{ST} = 0.1\, M_{OTV}$$

11) Calculate M_{OTV} from

$$M_{OTV} = M_{PP} + M_{PROP} + M_{POW} + M_{RAD} + M_{TTC} + M_{ACS} + M_{ST}$$

12) Finally, iterate steps 5–11 until M_{OTV} converges.

Cost Model

A simplified cost analysis was performed using the data generated by the OTV sizing algorithm. The total recurring cost of purchasing n_v OTV's and deploying n_{pl} spacecraft payloads was obtained from the equation

$$C = n_v\, C_{OTV} + (n_{pl} - n_v) M_{PP}\, C_{PP} + (n_{pl} - n_v) n_t\, C_{REF} + C_{TC} + C_{STS}$$

where

$$C_{STS} = \frac{\$87 \text{ million } [n_v M_{OTV} + (n_{pl} - n_v)\, M_{PM} + n_{pl}\, M_{CR}]}{29{,}484 \text{ kg}}$$

Acknowledgments

The authors would like to express their appreciation to Capt. Jess Sponable of the Air Force Space Division and to Mr. Jay Penn of the Aerospace Corporation for providing the technical information necessary to determine the OTV's applicability to the GPS mission.

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