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Reusable Orbital Transfer Vehicle Applications of an Integrated Solar Upper Stage

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There has been resurgent interest in the development of a reusable orbit transfer vehicle, based in low-Earth-orbit, for satellite placement, repositioning, recovery, and control. An integrated solar upper-stage system is examined for several near-Earth applications, including low-Earth to geosynchronous to low-Earth orbit ferrying. An integrated solar upper stage (a solar thermal power and propulsion system) permits rapid access to high orbits because of its moderate thrust levels (tens to hundreds of pounds) at specific impulses approaching 900 s, intermediate between high-performance chemical propulsion and electrical propulsion systems. An integrated solar upper-stage reusable orbit transfer vehicle can be used in two novel modes: 1) Drag return, in which large, rigid solar concentrators provide an aeroassist capability lowering total delta-V requirements; and 2) a hybridized integrated solar upper stage, coupled with and providing power to a xenon ion thruster, allowing more efficient maneuvers in nontime critical applications.

Introduction

A RECENT study conducted by the Air Force Materiel Command's Office of Aerospace Studies (OAS) concluded that advanced orbit transfer vehicles (OTVs) can significantly reduce the cost of the existing space-launch architecture. This study effort, entitled "Space Propulsion and Power Operational Effectiveness and Cost Comparison Study (OECS),"[1] was performed at the behest of Air Force Space Command and was assisted by the Air Force Phillips Laboratory (AFPL) and the Space and Missile Systems Center (SMC). The analysis demonstrated that acquisition cost for a typical geosynchronous-Earth-orbit (GEO)-based satellite constellation could be reduced by as much as 60%, if advanced upper-stage concepts, e.g., integrated solar upper stage (ISUS) (Fig. 1), were used to step-down or realign satellites from large, expensive launchers to smaller boosters. The potential savings engendered by step-down are enormous: SMC has determined that the current cost of Titan IV/Centaur, the largest expendable booster in the U.S. inventory, is approximately $500 million/launch. If a satellite that was nominally slated for flight aboard Titan IV was remanifested aboard a medium-lift vehicle (such as Atlas IIAR or Delta III), several hundred million dollars in launch costs could be saved.

The OAS examined a selection of advanced stages, including solar thermal, solar electric, nuclear propulsion systems, and bimodal systems (either nuclear or solar), which would be launched as integral components of a satellite vehicle bus and provide additional functionality on-orbit, primarily electrical power and on-orbit maneuvering. OAS determined that solar thermal and solar bimodal systems provided the most robust solution, offering cost reductions across the selected mission model. Solar electric systems provided the greatest cost savings, but could not efficiently reach certain orbits of interest and required long periods of time to perform a given orbit transfer. Nuclear systems of all kinds suffered from the presumed large research, development, test, and evaluation costs required to bring such a system to flight readiness. They were not shown to be cost-effective.

The OAS study did not attempt to look beyond Air Force Space Command's spacelift requirements. Furthermore, OAS compared the cost-effectiveness of advanced OTV concepts only in the context of the current launch infrastructure. No analysis was performed on future architectures involving reusable lower stages or even advanced, lower-cost versions of today's boosters, such as those being developed under the auspices of the Air Force's evolved expendable launch vehicle (EELV) program. These issues, coupled with emerging requirements for nontraditional space missions, led Space Command to request a second study effort.[2] This analysis was performed to determine the effectiveness of OTV concepts in an advanced architecture, consisting of EELV or reusable launch vehicle (RLV) boosters and missions other than spacelift. The OTV concepts examined in the original OAS effort would be analyzed to determine their ability to perform as reusable systems, based on-orbit and intended to carry out numerous missions during their life.

In addition to the commonly addressed requirement for more cost-effective satellite deployment, Air Force Space Command identified several missions that an OTV could potentially fulfill, primarily in the arena of space control: Slot denial (rendezvous, docking, and towing of a noncooperative satellite from its intended orbit); payload incapacitation (via sensor degradation, proximity jamming, surgical subsystem impairment, and vehicle pointing disruption); and metric observation (proximity imaging, surveillance of electromagnetic emissions, and satellite ephemeris tracking). It has been postulated that reusable boosters (or even expendable boosters) could be used for these missions; however, the advantage of the OTV is its highly efficient propulsion, permitting a longer reach than what could be achieved by a launch vehicle, essentially restricted in performance of these missions to low-Earth orbit (LEO). A set of vehicles that could contribute to these space control missions would lead to the development of true space superiority,

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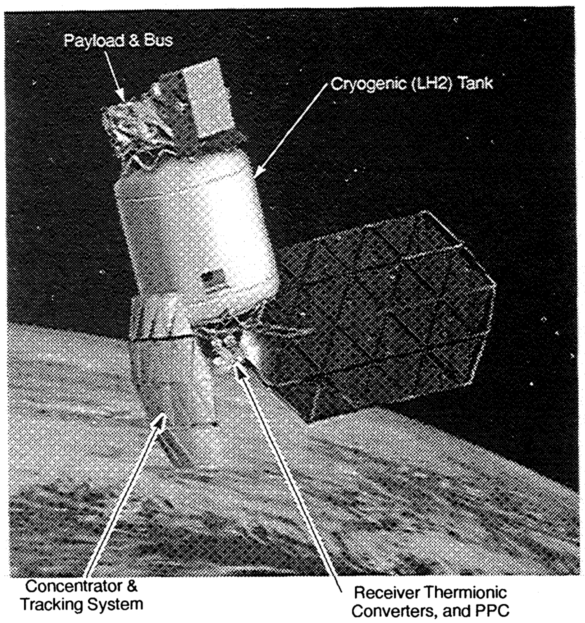

Fig. 1 ISUS.

permitting necessary access to, and control of, all orbits as high as geosynchronous.

While the OAS was unable to adequately examine these additional missions and assess an OTV's capability to perform them, the analysts were able to determine that an evolved expendable or fully reusable launch architecture appeared compatible with reusable versions of solar electric, solar thermal, and solar bimodal OTVs. None of these systems provided significantly greater cost savings as reusable vehicles than as expendable (EOTV) or integrated OTVs (IOTV), but their value as space superiority systems remains to be evaluated. This paper describes the design and mission analysis effort used to produce a reusable orbit transfer vehicle (ROTV) incorporating a solar bimodal power and propulsion system—ISUS.

System Description

The solar bimodal ROTV primary components include the solar bimodal engine system and OTV bus. The engine system includes the concentrator array and tracking system (CATS), receiver/absorber/converter (RAC), and cryogenic storage and propellant feed system (CSPFS). These three subsystems are being developed as a part of the ISUS Phase II effort. The bus will support attitude determination and control, thermal control, computer control and data handling, reaction control subsystem, and backup bus power functions for the ROTV.

RAC

The RAC subsystem basic function is to convert concentrated sunlight into usable heat to 1) drive the propulsion system to produce thrust during orbit transfer and 2) excite thermionic diode emitters to produce electricity. The RAC subsystem assembly consists of two symmetrical blackbody cavities within a refractory metal-coated graphite cylinder insulated with refractory metal foil, contained within a conventional material metal can. At the entrance of each receiver cavity is a secondary, nonimaging concentrator providing a 4:1 concentration ratio. A splash plate is incorporated surrounding the inlet to the secondary concentrator, which accepts spillage from the primary concentrator and provides feedback to the primary fine-pointing loop.

The absorber section of the RAC consists of graphite material with propellant flow passages and radiatively coupled thermionic diodes. The thermionic diodes are radially mounted about the cylindrical receiver with a direct view of the graphite to allow efficient radiative heat transfer. The thruster section contains a single 100:1 area ratio nozzle fed by hydrogen propellant exiting the absorber/graphite. Figure 2 presents a subsystem assembly drawing for an RAC design configuration with 60 kg of graphite for thermal storage and 60 thermionic diodes.[3]

CATS

The CATS consists of dual, off-axis, parabolic concentrators focusing solar energy into the secondary concentrators in the RAC. The sun track function is performed by using sun sensors on the surface of the concentrators that control motorized drives for rotation and by feedback from thermocouples attached to the splash plate at the focal plane.

Concentrator mass is a key parameter not only in assessing the performance of a given concentrator technology, but also is a driver in the system performance of a solar bimodal ROTV. The OECS assumed the use of deployable, rigid, solar concentrators with an areal density of 2.5 kg/m^2. NASA Lewis Research Center studies in the Phase 1 design effort of the ISUS program have shown that existing mirror technology using aluminum honeycomb panels should result in concentrators with an areal density of 2.0 kg/m^2.

A good candidate for a reference flight system concentrator design is the splined radial panel concentrator with a projected areal density of 1.0 kg/m^2.[4] Figure 3 shows an on-axis configuration of the splined radial concentrator concept (both deployed and stowed). Harris Corporation has fabricated a 15.24-m (50-ft) diameter off-axis splined radial panel concentrator for an antenna application.[5] Sizing of the concentrators for the solar bimodal ROTVs is based on the splined radial panel design. The concentrator diameters for the designs considered

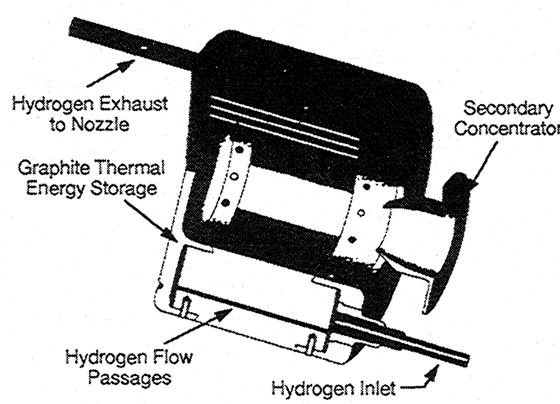

Fig. 2 RAC subsystem assembly.

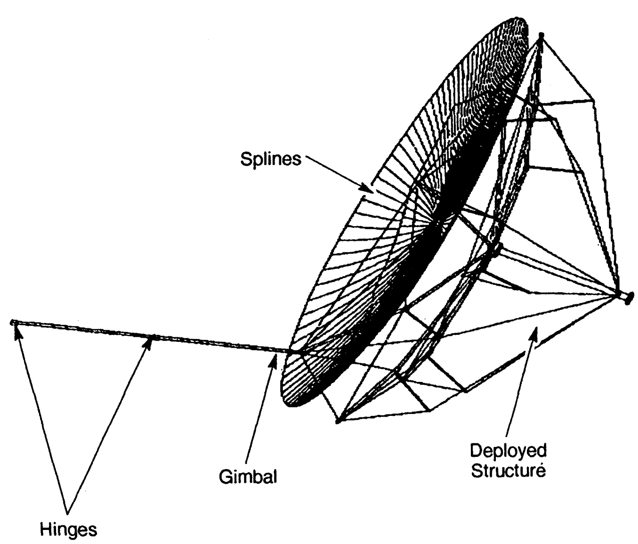

Fig. 3 Splined radial panel concentrator.

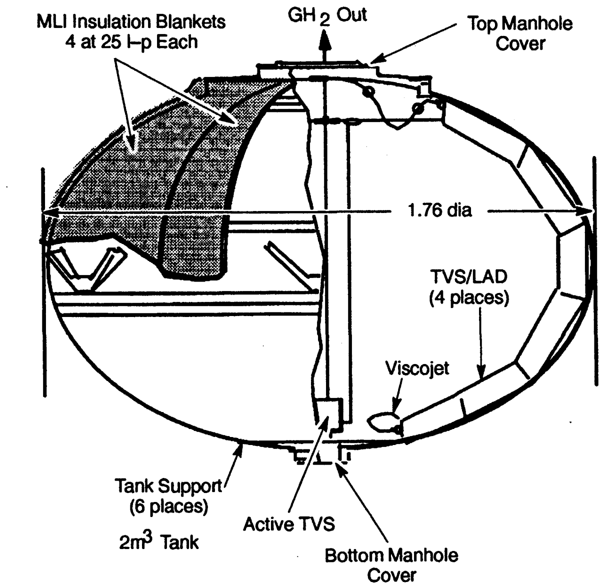

Fig. 4 CSPFS internal delivery components.

range in size from a few meters in diameter up to tens of meters.

CSPFS

The CSPFS consists of the propellant storage tank, thermodynamic vent system (TVS), propellant acquisition device, and propellant feed lines. The CSPFS design provides storage of cryogenic hydrogen during the orbit transfer phase without venting. Between perigee and/or apogee burns, the tank undergoes a lock-up condition in which heat is absorbed by the propellant through the multilayer insulation (MLI) and tank support structure. During firing of the RAC thrusters, the TVS removes heat from the propellant remaining in the tank and provides hydrogen propellant to the RAC. Figure 4 shows the internal delivery components of the LH_2 tank design.

Mission Analysis

The mission performed by an ROTV differs from an IOTV in that the propulsion/power bus does not stay with its payload. In this sense, the ROTV acts as a space tug moving payloads in the Earth orbital environment. The ROTV is expected to enable an expanded set of space operation functions that include satellite deployment, repositioning, graveyarding, and rescue. Space control functions are also enabled that include surveillance, metric observation, and counterspace.

Concept of Operations

The basic concept of operations (CONOPS) envisioned for the solar bimodal ROTV are shown in Fig. 5. The solar bimodal ROTV will rendezvous with a propellant module and payload in LEO and then proceed to the final payload orbit in thermal transfer mode. The payload is then deployed and the main propellant tank jettisoned. This up-leg portion of the ROTV transfer will use hydrogen as the propellant and is baselined for rapid 30- to 60-day transfer times. Two methods considered for the return-leg of the ROTV transfer are a combined thermal propulsion/aeroassisted (drag) return and xenon ion electric propulsion. Once the ROTV is returned to LEO it is placed in a passive/standby mode to await the next propellant module and payload.

Electrical power produced by the RAC is limited to the support of bus functions for the aeroassisted return-mode ROTV. An electric propulsion return-mode ROTV would require that the thermionic diodes produce electrical power for the electric

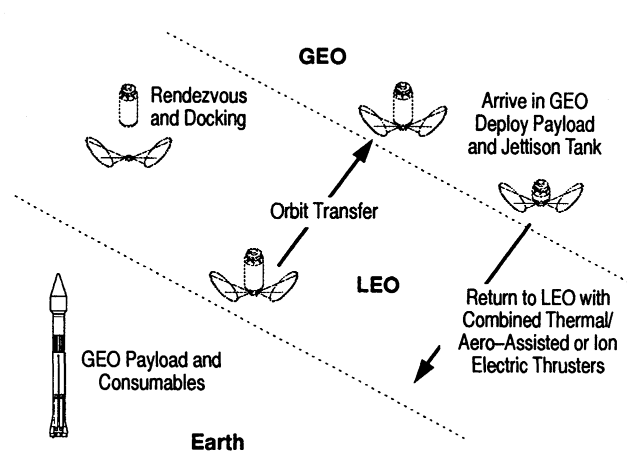

Fig. 5 Solar bimodal ROTV CONOPS.

thrusters plus bus functions. For the electric propulsion return mode the thrusters are operated in sunlight only, minimizing the thermal energy storage (TES) requirement.

Thermal Propulsion Transfer

Transfer from LEO to a final orbit will be performed using a multiple perigee and apogee burn approach. This approach uses several successive burns at perigee to raise apogee to a final apogee altitude and, also, several successive apogee burns to raise perigee to the final perigee altitude.

The solar bimodal ROTV was assumed to operate in an on-sun mode in this analysis. The TES medium in the RAC is charged to a peak temperature of 2500 K before each burn. The hydrogen propellant is then heated by the TES and incoming flux from the CATS. The propellant flow rate for apogee burns is lower than that for perigee burns because the magnitude of gravity losses is reduced. This allows for a modest increase to the mission average specific impulse (*Isp*) of the solar bimodal ROTV at the expense of a slight increase in delta-V.

Thermal propulsion is used to return the ROTV from a final circular orbit, e.g., GEO, to a highly elliptical transfer orbit, e.g., geosynchronous transfer orbit (GTO), to begin aeroassisted return. Hydrogen propellant is assumed for this short return-leg using a separate tank than the propellant for the up-leg of the mission. Depending on the dwell time required of the ROTV in the final orbit, a space-storable propellant such as ammonia (NH_3) or methane (CH_4) could be used. The thermal propulsion system is also used to provide control authority to the ROTV during the aeroassisted apogee reduction.

Aero-Assisted (Drag) Return

This return mode uses the solar bimodal system to return to an elliptical transfer orbit, and then lowers perigee to increase the drag force on the ROTV and return to a circular LEO orbit. Several successive perigee passes are used to reduce the apogee altitude. The solar concentrators are the primary drag surface area for the aeroassisted maneuvers at altitudes ranging from 80 to 120 n mile with drag forces on the order of 10–20 N. Figure 6 shows that the return transfer time for this aeroassisted maneuver was selected based on aerodynamic heating of the rigid solar concentrators. The aerodynamic heating of the concentrator surface was selected to be 1.5 times the heating experienced during normal launch and deployment of the concentrators. This selection was based on consideration of the length, altitude, and number of perigee passages and margin for current flight concentrator designs.

The aeroassisted drag return maneuvers occur in transitional and free molecular flow regimes. Propulsive control authority for the drag return portion of the transfer can be provided by either the solar bimodal engine or the reaction control system

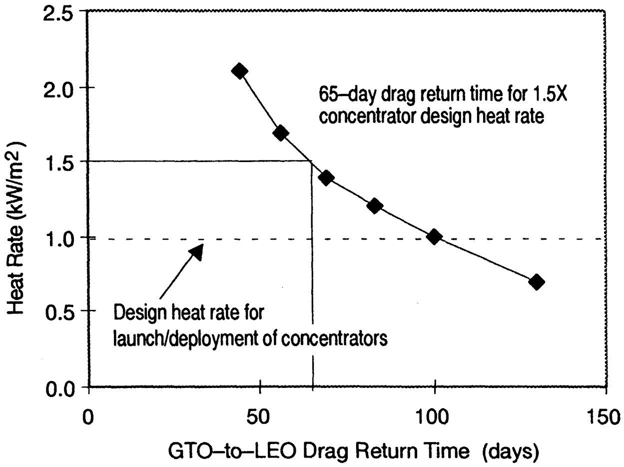

Fig. 6 Basis for selecting drag return time.

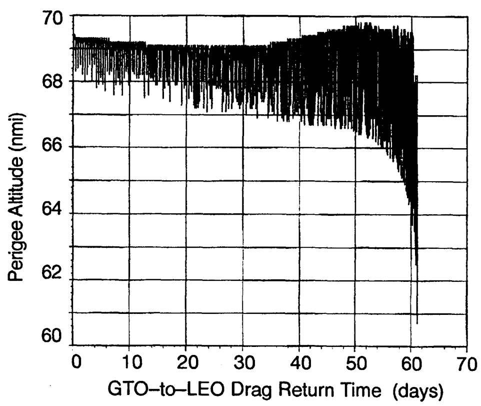

Fig. 7 Perigee altitude for aeroassisted return.

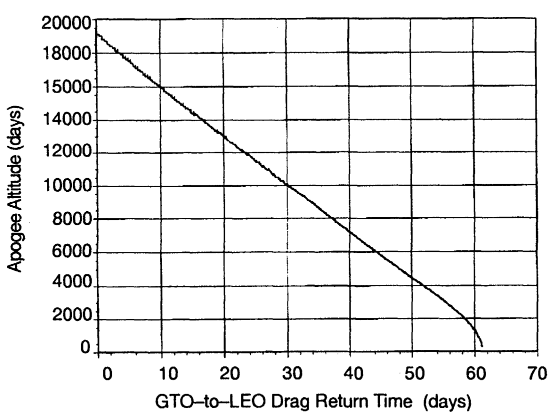

Fig. 8 Apogee altitude for aeroassisted return.

(RCS). Graphs of perigee and apogee altitude as a function of return transfer time from GTO to LEO are presented in Figs. 7 and 8. These plots are based on a return transfer analysis without propulsive control authority and indicate that the propulsive requirements are minimal because the reduction in perigee altitude is limited until the end of the transfer.

The aeroassisted braking maneuver has been successively used by the Magellan Spacecraft at Venus and is planned for the upcoming Mars Global Surveyor.[6,7]

Ion Electric Propulsion Return

The electric propulsion return mode uses a set of xenon ion electric thrusters to return the ROTV to LEO. This hybridized approach combines aspects of both thermal and electric propulsion systems to accomplish the return-1g. The electric thrusters are powered by the solar hybrid engine operating in a power mode and are operated only in the sunlight portion of the return transfer. The xenon ion electric thrusters were assumed to have a specific thrust of 0.04 N/kWe and *Isp* of 3200 s. These values are based on technology parameters from both the OECS and ROTV Spacecraft Concepts Study (RSCS).

The propellant module for this CONOPS includes a hydrogen tank and a xenon tank. The hydrogen tank is left in the vicinity of the final payload orbit and the xenon tank is jettisoned in LEO.

Analysis Results

The solar bimodal ROTV was evaluated as part of the RSCS. ROTV concepts evaluated in the RSCS include solar bimodal, solar thermal, and solar electric (xenon ion and H_2 arcjet) systems. This study focused on the cost-effectiveness and step-down capability of satellite deployment missions.

Medium and large ROTV configurations were developed for each technology. The solar bimodal ROTV configuration was initially evaluated for both return modes discussed in the CONOPS section. Table 1 presents performance for the solar bimodal ROTV using both modes for a LEO-to-GEO-to-LEO transfer. The time of flight (TOF) listed in the table indicates the total round-trip transfer time. The up-leg portion of the trip time for these cases was 60 days for the medium and 30 days for the large configurations. The initial mass in LEO (IMLEO) values listed in Table 1 include the ROTV, propellant, and tankage needed to place the respective payloads in GEO and to return the ROTV to LEO. The mission average specific impulse for the solar bimodal engine was about 790 s for the up-leg portion of the transfer and 835 s for the return-leg portion (GEO to GTO) of the thermal propulsion/aeroassisted return option.

The solar bimodal ROTV with the drag return mode was the configuration selected for more detailed evaluation in the RSCS. Although the electric propulsion return mode offers a slight decrease in IMLEO (1–2%), it adds complexity to the solar bimodal ROTV by requiring a 20- to 40-kWe power system and multiple electric thruster sets that would require change-out between trips. The round-trip transfer times for the electric propulsion return mode are also increased by about 50% over the thermal/drag return mode, which may imply the need for a larger ROTV fleet size to meet a given availability requirement.

The 5000-kg payload delivered to GEO (listed in Table 1) is comparable to the 5220-kg capability of the existing Titan IV (SRMU)/Centaur capability. The payload and propellant module needed to be launched and docked with the solar bimodal ROTV requires about 56–63% of the Titan IV [solid rocket motor upgrade (SRMU)] IMLEO capability. While the satellite and propellant module do not permit a Titan IV (SRMU) to Atlas IIAS step-down, several new launch vehicles currently under consideration could capture this step-down.

The 2500-kg payload delivered to GEO (listed in Table 1) is directly comparable to the 2500-kg capability of the existing Titan III. The payload and propellant module for the solar

Table 1 Solar bimodal ROTV–GEO performance

ROTV configuration	TOF, days	Payload, kg	IMLEO, kg
Thermal up/thermal/aeroassisted return			
Medium	101	5,000	13,200
Large	61	5,000	15,300
Medium	101	2,500	8,100
Large	61	2,500	10,100
Thermal up/electric propulsion return			
Medium	149	5,000	13,100
Large	109	5,000	15,100

Current Baseline	GPS		DSP	Milstar	DSCS
	DELTA		ATLAS	TITAN IV	ATLAS
	GPS (from 28)	GPS (from 55)	DSP	Milstar	DSCS
Xe Ion (75 W/kg)	LLV	LLV	LLV	TITAN NUS	LLV
Xe Ion (120 W/kg)	LLV	LLV	LLV	ATLAS	LLV
H_2 Arcjet (75 W/kg)	ATLAS			TITAN NUS	
H_2 Arcjet (120 W/kg)	ATLAS	LLV		TITAN NUS	
H_2 Resistojet (75 W/kg)	ATLAS		TITAN NUS		TITAN NUS
H_2 Resistojet (120 W/kg)	ATLAS	LLV	TITAN NUS		TITAN NUS
Solar Thermal (865 Isp)	ATLAS			TITAN NUS	
Solar Thermal (1000 Isp)		LLV		TITAN NUS	
Solar, Bimodal		LLV		TITAN NUS	DELTA
Nuclear, Xe Ion		LLV		TITAN NUS	
Nuclear, Bimodal	ATLAS	ATLAS	TITAN NUS	TITAN NUS	TITAN NUS
OECS Solar Thermal		LLV	LLV	TITAN NUS	LLV
OECS Bimodal		LLV		TITAN NUS	LLV
OECS Xe Ion		LLV	LLV	TITAN NUS	LLV
OECS H_2 Arcjet		LLV		TITAN NUS	LLV

Fig. 9 ROTV step-down/step-up results.

bimodal ROTV require 84–100% of an Atlas IIAS IMLEO capability, permitting step-down.

Figure 9 presents results from the RSCS for Global Positioning Satellite (GPS), Defense Support Program (DSP), Milstar, and Defense Satellite Communication System (DSCS) satellite categories. The GPS satellites were staged from starting inclinations of 28 and 55 deg. The step-down/step-up capability of ROTV technologies and selected integrated (IOTV) and expendable (EOTV) technologies are detailed. The ROTV technologies include Xe ion (solar and nuclear), H_2 arcjet, H_2 resistojet, solar bimodal, solar thermal, and nuclear bimodal concepts. With respect to the ROTV configurations, the ROTV tug is assumed to be in place and, payload, resupplied components, and consumables are launched for each given mission. The resupplied inerts include electric thrusters and ascent/descent tankage for the electric propulsion systems and ascent/descent tankage for the thermal propulsion systems.

The IOTV configurations include solar bimodal, arcjet, and ion propulsion systems and the EOTV system is solar thermal. These systems require launch of the IOTV/EOTV and payload for each given mission. The OECS evaluated these concepts, which are also evaluated with the RSCS mission set for the sake of comparison.

Figure 9 shows that the IOTV/EOTV systems permit step-down more consistently than the ROTV systems. The ROTV systems allow step-down for only a portion of the mission set and some ROTVs (specifically arcjet, resistojet, and nuclear bimodal systems) require step-up for parts of the mission set. The xenon ion ROTV is the only system that allows step-down for the entire mission set, but it requires very long trip times on the order of 300 days for ascent and 90 days for descent. This is not a surprising result considering that the propulsive requirement (delta-V) is doubled for the ROTV systems as compared to the IOTV/EOTV systems.

Differences in the technology and operational assumptions exist between the RSCS and OECS systems. The electric propulsion-based systems evaluated in the OECS assumed a specific power of about 60 W/kg, whereas specific powers of 75 and 120 W/kg were assumed for the concentrator arrays systems in the RSCS. Also, cell degradation was limited to only 1% per passage through the radiation belts, whereas it approached nearly 50% for some OECS cases.

Additionally, assumptions for the masses of the concentrator arrays for the solar bimodal and thermal systems differed from OECS to RSCS. The rigidized concentrators for the solar bimodal systems have lower specific masses and the inflatable concentrators have higher specific masses in RSCS as compared to OECS. However, the rigid concentrators system are heavier than the inflatable concentrators in both studies.

The solar bimodal ROTVs were designed particularly for propulsion, whereas the solar bimodal IOTVs were designed for both the propulsion and power aspects of the missions examined. Therefore, the solar bimodal ROTV designs produced mission average *Isp* that were 30–40 s higher than the IOTV configurations.

Using the combined thermal/drag return option significantly lowered the return delta-V requirement for the solar bimodal ROTV as compared to the other ROTVs evaluated.

The electric propulsion-based systems had a 50-kWe power system limit, which resulted in ascent trip times of about 300 days for a LEO-to-GEO mission and round-trip times in excess of a year for a given LEO-to-GEO mission. The solar bimodal and thermal ROTVs had trip times of 30 and 60 days and the nuclear bimodal ROTV had LEO-to-GEO transfer times on the order of hours.

Conclusions

The ISUS configured as an ROTV provides a good solution for meeting the Air Force Space Command's spacelift and space control requirements. The solar bimodal and solar thermal systems are the only ROTV configurations examined in the RSCS that provided both rapid response and a cost-effective method for meeting the U.S. Air Force Space Command's spacelift requirements. Space control missions such as denial, impairment, and metric observation that require an on-demand ROTV capability were not specifically addressed by the RSCS. However, the solar bimodal and solar thermal ROTV systems configured for this study could effectively enable such space control missions.

The RSCS did show that solar bimodal in an ISUS or IOTV configuration and solar thermal in an EOTV configuration were more cost-effective approaches for enabling rapid satellite deployment than similar ROTV systems. Xe ion electric propulsion systems are seen to provide the greatest cost savings but require very long transfer times and have difficulty reaching highly elliptical mission orbits.

The ISUS system uses rigid concentrators that offers the unique advantage of an aeroassisted return that lowers the overall delta-V requirement for the ROTV. Using inflatable concentrators for this tactic is questionable. This maneuver uses an elliptical starting orbit, which would require very long transfer times and high delta-Vs for electric propulsion ROTVs.

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References

[1]Feuchter, C. A., and Giczy, A. V., "Space Propulsion and Power Operational Effectiveness and Cost Comparison Study (OECS)," Office of Aerospace Studies, TR-91-1, Phillips Lab., Kirtland AFB, NM, Oct. 1996.

[2]Feuchter, C. A., Grobman J., and Hurtado, J., "ROTV Spacecraft Concepts Study (RSCS) Final Briefing," Office of Aerospace Studies, Phillips Lab., Kirtland AFB, NM, 1996.

[3]Jacox, M. G., Kennedy, F. G., Malloy, J., Merk, C., and Miller, T. M., "ISUS Space Demonstration System Definition Study Final Report," Phillips Lab., Kertland AFB, NM, PL-TR-96-1006, Jan. 1996.

[4]Schumacher, K. M., and Adams, M. A., "Solar Concentrators for Advanced Solar Dynamic Power Systems in Space Final Report," NASA Contract NAS3-25413, 1989.

[5]Borell, G. J., and Campbell, J. S., "ISUS Solar Concentrator Array Development," AIAA Paper 96-3045, July 1996.

[6]Rault, D. F., "Aerodynamic Characteristics of Magellan Spacecraft in Venus Upper Atmosphere," AIAA Paper 93-0723, Jan. 1993.

[7]Rault, D. F., Cestero, F. J., and Shane, R. W., "Spacecraft Aerodynamics During Aerobraking Maneuver in Planetary Atmospheres," AIAA Paper 96-1890, June 1996.