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ABSTRACT

This research was performed at the Space Systems Design Lab at the Georgia Institute of Technology, Atlanta, GA, with the charter of identifying economically attractive candidate space transfer vehicle systems for ferrying components of Space Solar Power (SSP) satellites from Low Earth Orbit (LEO) to Geostationary Earth Orbit (GEO). An aggressive price goal of only \$400/kg of payload was established in order to control the cost of transportation for the SSP satellite developer.

A multi-step decision process was employed to down-select from a large number of candidate systems to four. The final four concepts were Nuclear Thermal Rocket (NTR), Solar Thermal Rocket (STR), a rotating tether, and Solar Electric Propulsion (SEP). Additional concepts considered were Dual-Mode (Chemical/SEP) and All-Chemical.

Results show that the most economical concept is one which is highly reusable, has a short turn-around time, a long vehicle life, and small propellant requirements. These characteristics result in a low fleet size and therefore lower debt requirements. These characteristics also lower the Initial Mass in Low Earth Orbit (IMLEO) and therefore lower deployment costs. The goal of \$400/kg, or 2.5¢/kW-hr, for in-space transportation costs is very aggressive and difficult to achieve.

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LIST OF ACRONYMS

AHP	Analytic Hierarchy Process
ASE	Airborne Support Equipment
CABAM	Cost and Business Analysis Module
CAD	Computer Aided Design
CER	Cost Estimating Relationship
CSTS	Commercial Space Transportation Study
DDT&E	Design, Development, Testing, and Evaluation
EMA	Electro-Mechanical Actuator
ETO	Earth-to-Orbit
FRF	Flight Rate Factor
GEO	Geostationary Earth Orbit
HEDS	Human Exploration and Development of Space
HRST	Highly Reusable Space Transportation
IMLEO	Initial Mass in Low Earth Orbit
INSINC	IN-Space INCorporated
IOC	Initial Operating Capability
IRR	Internal Rate of Return
ISS	International Space Station
LCC	Life Cycle Cost
LEO	Low Earth Orbit
LOX/LH2	Liquid Oxygen/Liquid Hydrogen
MER	Mass Estimating Relationship
NAFCOM	NASA Air Force COst Model
NEP	Nuclear Electric Propulsion
NPV	Net Present Value
NTR	Nuclear Thermal Rocket
OTV	Orbital Transfer Vehicle
PMAD	Power Management and Distribution
R/LA	Rocket with Launch Assist
RBCC	Rocket-Based Combined Cycle
RLV	Reusable Launch Vehicle
ROI	Return on Investment
ROM	Rough Order of Magnitude
SEP	Solar Electric Propulsion
SERJ	Supercharged Ejector RamJet
SSDL	Space Systems Design Lab
SSM	Space Segment Model

SSP	Space Solar Power
SSTO	Single-Stage-to-Orbit
STR	Solar Thermal Rocket
TABI	Tailorable Advanced Blanket Insulation
TFU	Theoretical First Unit
TUFI	Toughened Unified Fibrous Insulation
UHTC	Ultra High Temperature Ceramic
WPT	Wireless Power Transmission

INTRODUCTION

Motivation

One solution to potential global warming lies in the study of Space Solar Power (SSP).[1] SSP is a clean energy system that collects solar radiation in orbit and transmits power back to Earth in the form of electromagnetic waves. Such an energy system could provide a non-nuclear alternative to the burning of fossil fuels. Many believe that the burning of such carbon dioxide-producing fuels has produced a global greenhouse effect, which is warming the planet.

The concept of generating solar power in space for Wireless Power Transmission (WPT) to receivers on the ground is not new. Discussed at some length during the past three decades, the idea was first proposed by Dr. Peter Glaser in 1968.[5] Now championed by John Mankins of NASA - Headquarters, this concept would place 30 satellites with many square kilometers of solar collectors in Geostationary Earth Orbit (GEO).[8-11] These satellites would then collect solar power, convert it into microwave energy, and beam it to large rectifying antennas on Earth for distribution into the electric power grid.

The original study of the SSP reference system in the 1970's concluded that the concept was not economically feasible with the technologies of that time.[3,18] Recent advances in technology have prompted the National Aeronautics and Space Administration (NASA) to conduct a "fresh look" re-examination of the technologies, systems concepts, and world energy markets that might make a future SSP system economically viable.[4,8,10,17] A team of engineers from government, industry, and academia continues to study this concept.

This research was performed at the Space Systems Design Lab (SSDL) at the Georgia Institute of Technology, Atlanta, GA under the sponsorship of the NASA - Marshall Space Flight Center, Huntsville, AL, Advanced Projects Office (FD02) grant number NAG8-1713.

Goal

The charter of this research was to identify economically attractive candidate in-space transportation vehicle (or orbital transportation vehicle, OTV) systems for ferrying components of large-scale SSP satellites from Low Earth Orbit (LEO) to Geostationary Earth Orbit (GEO). Earth-to-Orbit (ETO) and in-space transportation remain crucial to the success of any SSP concept due to the large masses required for power generation and transmission. An aggressive price goal of only $400/kg of payload, roughly corresponding to 2.5¢/kW-hr, was established in order to minimize the cost of this transportation service to the SSP satellite developer.

This analysis assumed that a public corporation (OTV Corp.), operating for profit, was subcontracted to provide in-space transportation from LEO (a 300 km circular, equatorial orbit) to GEO. The ground-rule was to deliver to GEO one 1.2 GW SSP satellite per year for 30 years. The total payload over the life cycle of this project (estimated to be about 600,000 MT) far exceeds any previous mission.

Payloads are delivered to LEO in 40 MT "chunks" by an independent ETO transportation system, operating re-usable launch vehicles (RLVs) at an Internal Rate of Return (IRR) of 25%. This service is assumed provided by a separate contractor (RLV, Inc.). A previous Georgia Tech study was conducted to assess ETO transportation options.[14] Results from that study are used here.

Objectives

The goal for SSP is to enable large-scale, commercially-viable solar power in space for terrestrial energy markets. The current global energy level is expected to continue growing at 6.6%/yr in developing countries (Edison Electric Institute forecasts) and 1.9%/yr in the US, perhaps doubling from 12 terawatts to more than 24 terawatts during the first decades of the new century.[8]

In order to capture its share of the world energy market, SSP must provide base-load power at no more than 5¢/kW-hr. Figure 1 shows that of this value, 0.5¢/kW-hr was budgeted to recurring operations and maintenance costs, 1¢/kW-hr to end-to-end wireless power transmission, 1¢/kW-hr to SSP power systems, and 2.5¢/kW-hr to SSP installation. SSP installation includes ETO transportation, in-space transportation, and ground assembly. For an initial 1.2 GW SSP satellite mass, the goal for transportation was set at $800/kg, $400/kg for ETO transportation and $400/kg for in-space transportation.

Approach

The philosophy throughout this project was to evaluate candidate technologies and through these evaluations to learn what is required to meet SSP in-space cost goals. It was not the intent of this study to promote any particular design or technology.

The analysis for this study was achieved through a versatile methodology, which progressed through a series of down-selections from a wide field of many technologies to a small handful of technologies with increasing levels of analytical detail. This analysis included both objective and subjective comparisons of designs. The analysis performed was both top-down and bottom-up. Results captured desirable characteristics of an in-space transportation system and requirements necessary to achieve the aggressive cost goals set by SSP.

The following techniques, used in this methodology, are discussed later in detail:

1. Brainstorming
2. Rough Order of Magnitude (ROM) Analysis
3. Analytic Hierarchy Process (AHP)

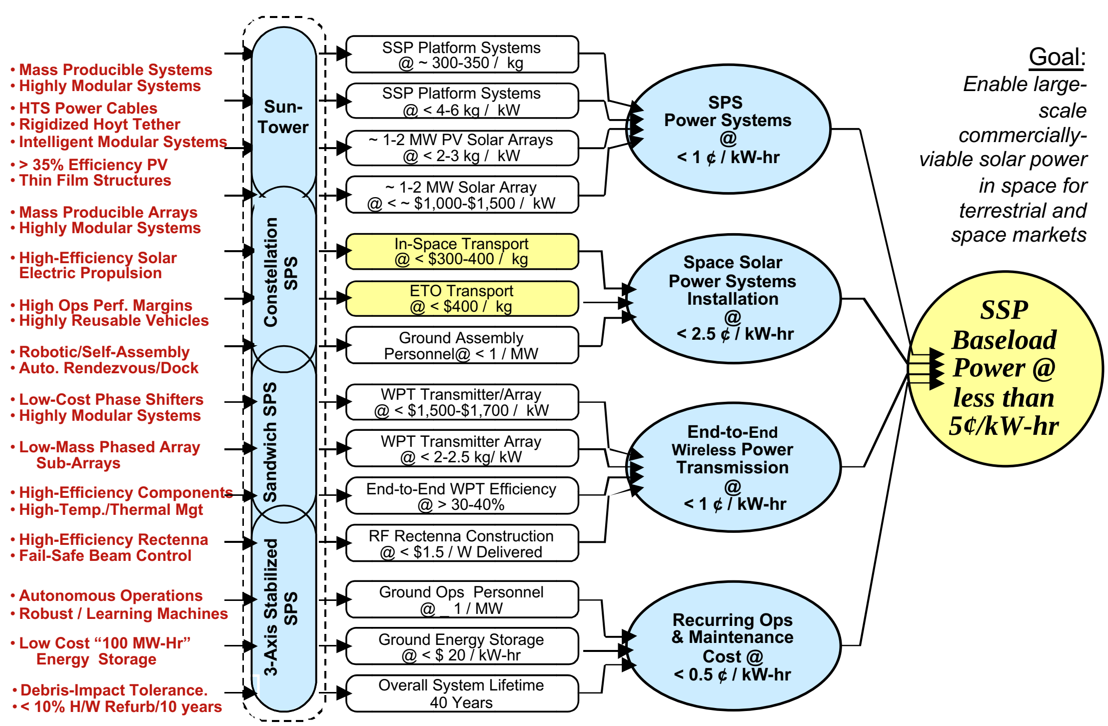

Figure 1 - SSP Economic Goals.

4. Bottom-up Analysis
5. Top-down Analysis

SSP BACKGROUND

Each SSP satellite, designed for collecting 5 GW of solar power, is sized to provide 1.2 GW of that power to the ground receiving stations. Many satellite concepts have been proposed by other researchers on the project with initial masses in low Earth orbit (IMLEOs) ranging from 10,000 MT to nearly 30,000 MT. All of these concepts require advanced technologies in the areas of solar collection and wireless power transmission (WPT). This advanced technology is also assumed to be available to the OTV. Several concepts, including the Abacus/Reflector pictured in Figure 2, are listed below along with their initial on-station masses.

1.	Integrated Symmetrical Concentrator (ISC)	9959 MT
2.	Abacus Suntower	21610 MT
3.	Abacus/Reflector	19529 MT

As mentioned in the introduction, SSP far exceeds any previous mission in mass delivered to orbit. To put this statement in perspective, consider that the completed International Space System (ISS) will weigh about one million pounds (454 MT) and will require 46 construction flights. In comparison, a single SSP satellite will have a mass on the order of 20,000 MT, making each satellite about 44 times larger than ISS! Placing 30 SSP satellites in GEO within 30 years will require about 1200 ETO flights a year, or three flights a day! The space shuttle currently flies about 5 times a year.

Additionally, there were 78 launches worldwide in 1999 and there have been 56 launches to date in 2000 at the time of this writing. Had each flight over the past two years flown at its maximum capacity, the total payload delivered to LEO would have been on the order of 1293 MT. That gives an average number of launches per year of about 76 with an average payload of about 10 MT. Therefore, SSP would have about 16 times higher flight rate and about 63 times higher payload than the current world market (including ISS launches).

The point of this comparison is to emphasize that SSP is an extremely large mission and will constitute the largest construction project ever attempted by man. The magnitude of this project is so large that it forces a paradigm shift in the way transportation costs are minimized. Previous studies showed that launch prices can be reduced through higher flight rates. This is higher flight rates!

ETO BACKGROUND

A recently completed study at Georgia Tech examined various RLV options for delivering a constellation of Space Solar Power satellites of the Suntower configuration to LEO.[14] One of the motivations of the study was to determine whether the aggressive \$400/kg (\$183/lb) launch price goal, established for SSP package delivery, would result in an attractive economic scenario for a future RLV developer. That is, would the potential revenue and traffic to be derived from a large scale SSP project be enough of an economic “carrot” to attract an RLV company into developing a new, low-cost launch vehicle to address this market. Results showed that there is enough economic reward for RLV developers

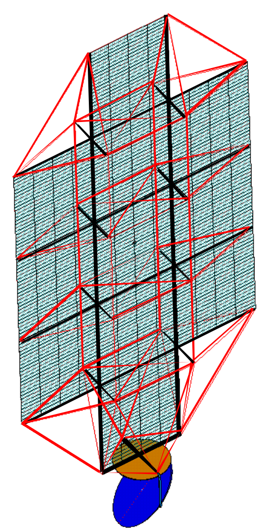

Figure 2 - SSP Abacus/Reflector.

with internal rates of return for the 30 year economic scenario exceeding 22%. However, up-front government assistance to the RLV developer in terms of ground facilities, operations technologies, guaranteed low-interest rate loans, and partial offsets of some vehicle development expenses was necessary to achieve these positive results.

Candidate SSP Vehicle Descriptions

As a point of departure, Georgia Tech started with the three top finishing launch vehicle designs and one additional "wildcard" from NASA's recent Highly Reusable Space Transportation (HRST) study.[7] The HRST study had a goal of achieving direct recurring costs under $400/kg ($200/lb) for payloads in the range of 10 MT to 20 MT and flight rates less than 200 flights/year. To achieve this goal, vehicle concepts had to be highly operable and reliable, require very little maintenance between flights, have sufficient system and subsystem robustness (typically substantial design margins), and contain long life airframe and engine components. HRST-class vehicles typically require no more than $3M - $4M in labor, propellant, and replacement hardware per flight. Airframe service life is on the order of 1000 flights and engine service life is on the order of 500 flights. By comparison, the current Space Shuttle system requires more than $350M in recurring costs per flight, and its service life is around 100 flights for the Orbiter airframe and only a few flights between major overhauls for the main engines.

The four HRST-class vehicles investigated in this SSP study are listed here and described below:

1. *Argus* with Maglifter launch assist
2. *Hyperion*
3. ACRE-92
4. SSTO-R with sled launch assist

The initial payload studied for each of these vehicles was 20 MT. A sensitivity study was performed to assess the economic effects of varying the payload capacity for the high flight rate SSP mission. This study showed that a larger 40 MT payload was more attractive for *Argus* and SSTO-R/LA.

Argus

Argus, shown in Figure 3, is a rocket-based combined-cycle (RBCC), single-stage-to-orbit (SSTO) launch vehicle which utilizes a magnetically-levitated (Maglev) sled and track system to accelerate to Mach 0.8 for a horizontal liftoff.[12] *Argus* uses two LOX/LH2 supercharged ejector ramjet (SERJ) engines for primary motive power and transitions from airbreathing to rocket mode at Mach 6. Like the rest of the vehicles considered, *Argus* is unpiloted and operates autonomously from liftoff to landing. *Argus* employs a lightweight composite airframe in a high fineness ratio, axisymmetric wing-body configuration.

Advanced subsystem and material technologies are used throughout. For example, the wings and other highly loaded structures are made of advanced metal matrix composites such as Titanium-aluminide. Propellant tanks are graphite/epoxy. Subsystems include high power density fuel cells, EMAs, lightweight avionics and power distribution, and built-in test monitoring sensors. Thermal protection is all passive with a combination of Toughened Unified Fibrous Insulation (TUFI) ceramic tiles, Tailorable Advanced Blanket Insulation (TABI) blankets, and ultra-high temperature ceramic (UHTC) nosecap and leading edges.

Figure 3 – *Argus*.

Hyperion

Hyperion is a horizontal takeoff, horizontal landing RBCC SSTO launch vehicle.[13] Shown in Figure 4, it is powered by five LOX/LH2 ejector scramjet engines, but is also equipped with a separate pair of ducted fans for limited subsonic landing operations. *Hyperion* operates in airbreathing scramjet mode up to Mach 10 and requires significant airframe-engine integration. The *Hyperion* forebody is conical on the bottom and elliptical on the top. The aftbody provides an expansion surface for the engine exhaust. Airframe and subsystem technologies are similar to those in *Argus*. Both *Argus* and *Hyperion* were entered into the original HRST vehicle evaluation process by Georgia Tech's Space Systems Design Lab.[7]

Figure 4 – *Hyperion*.

ACRE-92

ACRE-92, Figure 5, is a vertical takeoff, horizontal landing LOX/LH2 all-rocket launch vehicle. It is powered by five new long life, high thrust-to-weight rocket engines (T/W = 92 at sea level). Landing is unpowered. It employs a wing-body configuration similar to that found on the all-rocket SSTO from NASA's Access to Space study. Subsystem and materials technologies are consistent with *Argus*. ACRE-92 was originally entered into the HRST study by Dan Levack of Boeing Rocketdyne.[7]

SSTO-R/LA

The Single Stage to Orbit Rocket with Launch Assist (SSTO-R/LA), shown landing in Figure 6, is a horizontal takeoff, horizontal landing SSTO rocket vehicle. Like *Argus*, it employs a launch assist system to achieve an initial velocity and eliminate the need for heavy takeoff gear. In this case the launch assist system is a rocket-powered sled and track system and the launch speed is only Mach 0.25. Main propulsion for the SSTO-R/LA vehicle is provided by three lightweight LOX/LH2 rocket engines. The vehicle configuration is a medium fineness ratio wing-body. Subsystem and materials technologies are consistent with *Argus*. The SSTO-R/LA was entered into the HRST study by Gordon Woodcock, formerly of Boeing Huntsville.[7]

Study Results and Issues

IRRs for were over 22% for both *Argus* and SSTO-R/LA for the basic GEO SSP model. *Argus* had a slight advantage in cost per kg payload due to lower operations costs, but IRRs of the two vehicles are very close due to the lower up-front costs of SSTO-R/LA. Over the 30 year SSP deployment mission model, best case revenues for the optimized CSTS overlay are close to $240B for RLV, Inc. while total life cycle costs incurred are near $80B (in 1998 dollars).

In our experience, achieving the goal of 20% IRR requires cost per kg payload to be less than 1/2 of the expected price per kilogram. That is, overall life cycle cost per kg of payload

Figure 5 - ACRE-92.

Figure 6 - SSTO-R/LA.

delivered should be less than $200/kg for this model with direct recurring costs less than 1/2 of all costs. Typical values used for price-to-cost ratio are 2 to 4. For *Argus*, the best case costs were $147/kg. For SSTO-R/LA, the best case costs were $160/kg. Achieving this low cost goal was the result of a combination of several factors.

1. Having sufficient total flights in the model to amortize vehicle DDT&E and fleet costs
2. Achieving very low operations cost with new ways of doing ops and high annual flights
3. Augmenting SSP revenues with emerging LEO-bound commercial market traffic to increase profits
4. Reducing financing costs for initial capital (low interest loans and smaller, cheaper vehicles)
5. Government assistance to reduce up-front costs (facilities/launch assist and DDT&E offsets)

The results from this SSP ETO study supported the conclusion that, the GEO-based SSP scenario produced an attractive economic scenario for a potential RLV developer, even if the revenues are limited to only $400/kg. With proper support from the government, the sustained, high traffic mission model from the SSP creates a steady revenue source that enables RLV, Inc. to recoup startup costs and still provide an adequate return on investment.

IN-SPACE TASK METHODOLOGY

Brainstorming

In contrast to the ETO study, there was not a pre-established fleet of low cost concepts to evaluate. Therefore, the first step in the analysis was to consider as wide a range of candidate technologies as possible. A brainstorming session, enlisting the help of the entire SSDL membership, was organized to determine potential candidates. The simple ground-rules followed were first that all suggestions were accepted and second, that no idea was rejected or criticized by the group. Each participant was asked to bring a written list of several methods of transportation ranging from the near-term to the far-out. In a round-robin fashion, each participant was asked, in turn, to present one technology or pass. This continued until all ideas were exhausted.

The brainstorming ideas were recorded, consolidated, and grouped by technology. The major technology groups included nuclear, solar, chemical, tethers, and others. Once the brainstorming session was over, the concepts and technologies were discussed and 14 of the most promising ideas were selected for further examination.

ROM Analysis

The second step in the methodology was to perform a fast, rough order of magnitude (ROM) analysis on a large number of concepts. The results from this analysis, estimates of the launch prices, were used for further decision-making. The following 14 candidates, evaluated as the most attractive options to emerge from the brainstorming session, were analyzed in this fashion:

1. Chemical OTV (partially re-usable solid)
2. Chemical OTV (re-usable cryogen)
3. Cycler
4. Laser propulsion
5. Mass driver in LEO
6. NEP OTV - nuclear electric propulsion
7. NTR OTV - nuclear thermal rocket
8. Revolutionary propellants

9. Self ferry of SSP components
10. SEP OTV (one-way)
11. SEP OTV (two-way)
12. STR OTV - solar thermal rocket
13. Tether (single ΔV)
14. Tether (dual ΔV)

Sub-teams of two or three were formed from the group and given several of the above options to evaluate. These sub-teams were tasked with researching, determining the gross characteristics, and supplying inputs for the ROM analysis for each of their chartered options. This ROM analysis, performed on a spreadsheet by the team-lead, used the following top-level inputs provided by the sub-teams to generate estimated launch prices for all the options:

1. Transit Time (Days)
2. Refuel & Service Time (Days Per Flight)
3. Number Of Payload Chunks Per OTV flight
4. Tug Life (Flights)
5. Outbound Delta-V (M/S)
6. Inbound Delta-V (M/S)
7. Lambda (Structural Mass Fraction)
8. Isp (Sec)
9. Depot Mass (Kg)
10. Non-Propellant Mass/Inert Mass (0-1)
11. Average Tug Flight Ops ($/Flight)
12. OTV Complexity Factor (0-1)
13. OTV DDT&E ($)
14. Depot DDT&E ($)

The results of the ROM analysis began to establish the desirable characteristics necessary for low launch costs and served as a down-selection for the next step in the methodology. The top eight candidates were chosen to continue to the next selection. This was the first attempt at providing an objective comparison of the candidates and the feasibility of reaching the $400/kg cost goal.

AHP Selection

The next step was to convene the group for a down-selection to the top four candidates. The final four concepts would go on for a more detailed bottom-up analysis. The Analytic Hierarchy Process (AHP) was used to facilitate this decision. AHP is a systematic, analytically-based, multilevel process. Developed by Dr. Thomas Saaty at the Wharton School of the University of Pennsylvania, AHP is often used in annually prioritizing technology investments in support of NASA's space transportation programs.[16] In the AHP, a multi-criteria decision is structured in the form of a hierarchy of evaluation criteria. Then the alternatives are prioritized through a series of pair-wise comparisons. Each evaluator is asked for his or her judgment as to the relative strength of the alternatives against the evaluation criteria. Scores are assigned according to the Saaty scale, shown in Table 1, and recorded in a matrix:

The eight concepts (alternatives) compared

Table 1 - Saaty Scale.

Numerical Scale	Explanation
1	The two alternatives are of equal strength
3	Experience and judgment indicate that alternative A is moderately stronger than alternative B
5	Experience and judgment indicate that alternative A is significantly stronger than alternative B
7	Experience and judgment indicate that alternative A is very significantly stronger than alternative B
9	Alternative A is considered totally dominate over alternative B

using the AHP were:

1. Laser Propulsion
2. Nuclear Electric Propulsion
3. Nuclear Thermal Propulsion (NTR)
4. Solar Electric Propulsion (SEP), expendable
5. Solar Electric Propulsion, re-usable
6. Solar Thermal Propulsion (STR)
7. Tether #2 (momentum)
8. Tether #3 (electro-dynamic)

Five groups were established to independently rank the candidates in one of five criteria: low cost, low technical risk, low political risk, low exposure (or low up-front debt), and synergy with HEDS programs. Low cost was determined to be the most important criteria and was given a weighting of 60%. The remaining criteria were each weighted 10%. The prioritization matrices were then synthesized, normalizing each column and summing across each row. The resulting vectors were then assembled in a synthesis matrix and the process was repeated. The synthesis matrix is shown in Figure 7 along with the resulting prioritization vector, which provided an overall ranking of the concepts.

The highest scoring alternative in each criteria is circled along with the overall highest scorer. Four sub-teams were again appointed, each being tasked with providing a bottom-up analysis of one of the final concepts. The two tether options were determined to be similar enough to constitute the same concept and were combined. The decision whether or not to include an apogee kick stage was left to the design team as a trade study. Therefore, the four final concepts, representing a diverse set of technologies, chosen for study were:

1. Solar Thermal Propulsion (STR)
2. Solar Electric Propulsion (SEP), 1-way
3. Momentum Transfer Tether
4. Nuclear Thermal Propulsion (NTR)

Bottom-up Analysis

A bottom-up analysis, performed by each of the sub-teams, consisted of the following disciplinary analysis, described in detail below:

1. Weights and Sizing
2. Trajectory Analysis
3. ETO Cost Analysis
4. In-space Cost Analysis

Weights and Sizing

The weights and sizing analysis was performed using photographic scaling of the OTV and a set of Mass Estimating Relationships (MERs) that had a NASA Langley heritage, augmented with relationships from the Space Segment Model (SSM), developed by SAIC. This analysis was performed on an Excel spreadsheet. Using the results of the trajectory analysis, the OTV was scaled up or down until the available mass ratio and the required mass ratio matched. The baseline MERs were adjusted downward by linear scaling factors to reflect the selection of advanced material technologies, construction techniques, and

	Low Cost	Low Technical Risk	Low Political Risk	Low Exposure	High HEDS Synergy		
	60%	10%	10%	10%	10%	Score	Ranking
NTR	0.11212	0.11730	0.01878	0.22900	0.03989	0.10777	5
NEP	0.08755	0.13929	0.01809	0.15550	0.03989	0.08780	6
Laser	0.02656	0.02260	0.05630	0.05286	0.17918	0.04703	8
SEP (1-way)	0.15731	0.24283	0.22091	0.19610	0.08889	0.16926	2
Tether #2	0.12680	0.06860	0.11043	0.03329	0.22501	0.11981	4
Tether #3	0.17749	0.09015	0.11818	0.03329	0.22501	0.15316	3
STR	0.26706	0.21656	0.22866	0.19610	0.12156	0.23653	1
SEP (2-way)	0.04511	0.10267	0.22866	0.10386	0.08056	0.07864	7

Figure 7 - AHP Synthesis Matrix.

lightweight subsystems consistent with the program initial operating capability (IOC) of 2020. The output of the weights and sizing analysis was a 28-point weight (mass) breakdown structure.

Trajectory Analysis

Several types of trajectories were analyzed from high to low thrust. All transfers were assumed to start in a 300 km circular, equatorial LEO and finish in a circular GEO, thus not requiring any plane changes. The high thrust trajectory was the simplest to calculate, requiring only two Hohmann transfers and a total ΔV of 7634 m/s. The multiple burn transfer required for the STR concept requires an adjustment for gravity losses. A first-order approximation was used, increasing the Hohmann ΔV by 5% to 8015 m/s.

SEPSPOT, a time optimal or nearly time optimal trajectory program developed at NASA Lewis Research Center, was used for analyzing the low-thrust transfers. The simulation assumed no degradation effects due to distance from the sun, the Van Allen radiation belt, or shadowing.

A useful parameter for comparing low thrust SEP trajectories is the initial mass to power ratio (m_0/p_0). The amount of propellant required mainly depends on the specific impulse. Figure 8 plots the resulting optimal mass ratios (initial mass to final mass ratios) for various values of specific impulse. The time of flights required to transfer from LEO to GEO, on the other hand, depends on both m_0/p_0 and specific impulse. Figure 9 illustrates this relationship. For a m_0/p_0 of 200 kg/kW, the one-way required ideal ΔV was 4.6 km/s. This provides a first-order approximation of a 20% increase in low thrust ΔV over the Hohmann transfer.

ETO Cost Analysis

As a way of introducing the in-space cost analysis, a brief review of the ETO cost analysis methodology from the previous study[14], is presented here.

ETO Cost analysis was done using the Georgia Tech in-house code, CABAM.[6] CABAM (Cost and Business Analysis Module) was developed in response to the need to have a tool that provides a financial assessment of conceptual launch vehicle designs. This tool incorporates not only the cost attributes associated with a project, but also identifies the potential revenue streams and projects a number of evaluation metrics including net present value (NPV), internal rate of return (IRR), and return on investment (ROI). IRR is defined as the discount rate for a certain project that results in a \$0 NPV. Neglecting risk, higher IRRs are better.

CABAM uses data from the NASA Commercial Space Transportation Study (CSTS) and user entered competition models to

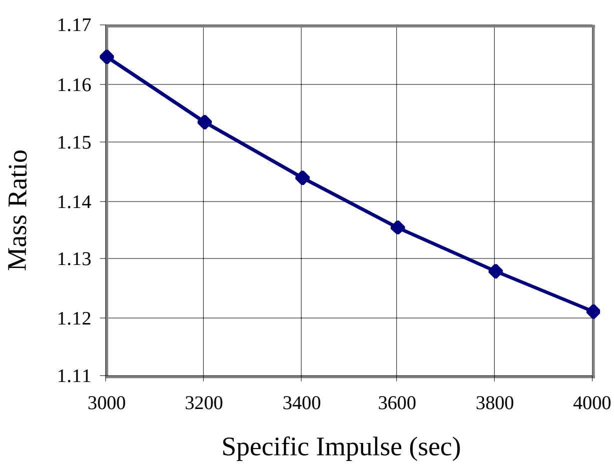

Figure 8 - Low Thrust Mass Ratio.

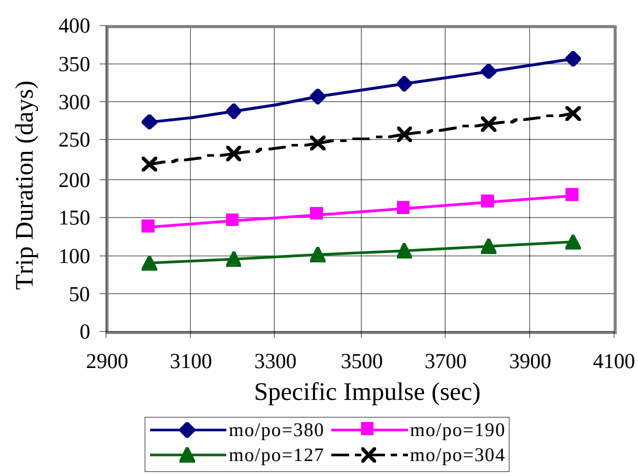

Figure 9 - LEO to GEO Trip Duration.

approximate the price elastic behavior of potential markets. The percent of market capture, therefore, depends on the price. The 'medium' market growth model from the CSTS study was used for the baseline, but the nuclear waste disposal market was not included.

CABAM is NASA- Air Force Cost Model (NAFCOM) based and uses Cost Estimating Relationships (CERs), which depend on subsystem masses. CABAM determines LCC and IRR based on estimating the project cash flow through calculations of both the cost and revenue streams.

After the designs were converged, CABAM was used to estimate vehicle DDT&E and theoretical first unit (TFU) costs for each vehicle. For a given fleet size, overall fleet procurement costs were estimated using a 75% learning curve for units produced beyond the first. Key assumptions made for this study included: government offset of 20% of the airframe development and 100% of the engine development costs (but none of the production), a constant source of revenue from SSP for payload delivered, government-backed loan interest rates of 7.5%, and a 3:1 debt-to-equity ratio model for raising necessary capital.

In-space Cost Analysis

Literature reviews, cost-by-analogy, and cost estimating relationships (CERs) were all used to cost tether, NTR, STR, and SEP in-space transportation concepts. Information for the costing process was also gathered from both the Space Segment Model (SSM), developed by SAIC to examine SSP architectures, and NAFCOM. Spreadsheet models were developed to determine the total program cost of various concepts based upon input mission and mass statements.

Assumptions were made in the costing process for many of the concepts. Chemical transportation systems used NAFCOM CERs for each of the following line items: LH2 Tank LOX Tank, Other Structures, Propulsion, and Subsystems. NTR costing consisted of specified chemical transportation CERs along with previous cost estimates of NTR programs in the 1960s in the United States. The STR cost process used elements from both the SEP and chemical system costing process while the tether costing relied on literature reviews for estimates of production tethers.

Propellant costs (FY$2000) were assumed as follows: $2.3/kg for LH2, $0.6/kg for LOX, and $288/kg for xenon or krypton propellant. The propellant costs shown here are justified on the grounds of better future extraction of these propellants in 2020 due to technological development. In addition, economics of scale are inherent for the large industrial production for SSP compared with modern demand.

Economics for the in-space concepts was estimated using a new Georgia Tech in-house code, named INSINC (IN-Space, INCorporated). INSINC builds a vehicle development program around projected SSP infrastructure demand. The financial qualities of that program were determined from user defined programmatic variables. INSINC is robust enough to handle different vehicle concepts, development schemes, financing plans, and pricing structures. The model can also scale up the required number of in-space vehicles depending upon the payload to be delivered to any final SSP destination orbit. The commercial provider of in-space transportation services modeled in INSINC was assumed to be using the same ETO launch service provider as the SSP power company. In order to account for the cost of launching the in-space transportation system INSINC requires the payload capability and ETO launch price from CABAM.

The economic and financial portions of the INSINC model obtain inputs from the market, schedule and economic, and vehicle definition sections of the model. Financial metrics like internal rate of return (IRR) and net present value (NPV) are determined through calculation of specific program costs coupled with user-defined pricing. Thus there is no elastic market for demand specified in the model. The SSP infrastructure company is assumed to pay the in-space transportation company a set price ($/kg) for its services.

The economic variables that need to be defined for each analysis include:

- dollar year
- inflation rate (3%)
- tax rate (30%)
- discount rate (around 25%)
- average annual interest rate (used for calculation of the interest that needs to be paid on deferred liability or debt, around 10%)

The financing variables include those that determine both the frequency and amount of equity (i.e. stock) offered as well as the per-year fixed and per-flight variable SG&A (Selling, General, and Administrative) expense. The scheduling variables include user determination of IOC, program termination, and years for vehicle development. Before any flights can occur, INSINC (based upon user input) segments development into appropriate years before IOC.

For each vehicle the following fleet definition variables are needed:

- system dry mass w/o payload
- total propellant mass
- payload capability of module
- payload inefficiency factor
- overall reliability
- trip time to delivery orbit (days)
- in space turn-around-time (days)
- average annual salary per man ($/yr)
- manpower per launch
- labor cost per year ($M/yr)
- OTV flights per year
- expended hardware/launch
- hardware refurbishments ($/kg reusable)
- propellant costs ($/kg)
- DDT&E cost
- TFU cost
- learning effects
- government contribution percentages

Insurance in the model refers only to vehicle liability insurance per flight based upon the expected probability of failure multiplied by the TFU cost of the vehicle's airframe and engine.

A separate mission and costs section determines the spread of flights dependent upon the payload to be delivered per year. The payload capability and reusability data of the in-space transportation vehicle determines the actual trips per year, number of vehicles for such trips, the number of refurbished vehicles, the total dry mass required, the total propellant mass required, the total expended hardware mass, and the new propellant mass required. These are aggregated to determine the total number of ETO flights per year. This data is then used to determine non-recurring costs (vehicle and facilities development and government contribution), recurring costs (site fee, insurance, labor cost, propellant cost, hardware plus propellant refurbishment), ETO launch costs, and revenues (for an input price).

Equity calculations are then determined along with associated depreciation schedules. Depreciation is defined using U.S. government standards based upon a 5-year depreciation of fixed assets. A separate debt calculation is made with the assumption that negative cash flows in any given year (after accounting for revenue and equity infusion) are paid off using either long or short-term bonds (20, 15, 10, 5, or 1 year varieties). For this financial analysis, the free cash flow was defined as: earnings before interest and taxes minus capital expenditures (airframe and engine acquisition) plus depreciation

All the above information is aggregated to obtain the discounted cash flows and associated summary metrics like NPV (for NPV, based upon user defined discount rates).

Top-Down Economic Analysis

A quick, top-down economic analysis was also used to evaluate concepts. In this approach, the life cycle cost (LCC) of each concept was constructed from its components: DDT&E, production, ETO transportation, and operations costs. Each of these costs was derived from the mass statements of the concepts, as described above. For example, the DDT&E was determined simply as the product of an average

Table 2 - ETO Launch Price Data.

Annual Flight Rate	15% IRR Launch Price	25% IRR Launch Price
45	1258.00	2239.00
90	735.75	1262.00
180	470.75	778.75
360	305.25	490.25
450	271.25	432.25
540	246.50	391.00
630	224.00	353.25

Table 3 - Curve-fit Parameters.

Parameter	15% IRR	25% IRR
L (low value)	107.41	170.99
H (high value)	269.88	430.21
a (exponent)	0.85	0.90

cost per kilogram and the dry weight of the vehicle.

The ETO transportation cost was the cost born by OTV Corp. to place the OTV in LEO. It was assumed that the same launch price, determined by the total number of flights, would be charged to OTV Corp. and SSP, Inc. The ETO launch price is a function of both the annual flight rate and the IRR for RLV Inc. A price trade for *Argus* in the previous study determined price values at flight rates ranging from 45 to 630 flights per year for both 25% IRR and 15% IRR. This data, shown in Table 2 below, was curve-fit to get expressions for launch price at the two IRRs.

The form of the curve-fit equations used is shown in Equation (1). The parameter values are listed below in Table 3. Figure 10 shows the data points and the resulting curve fits.

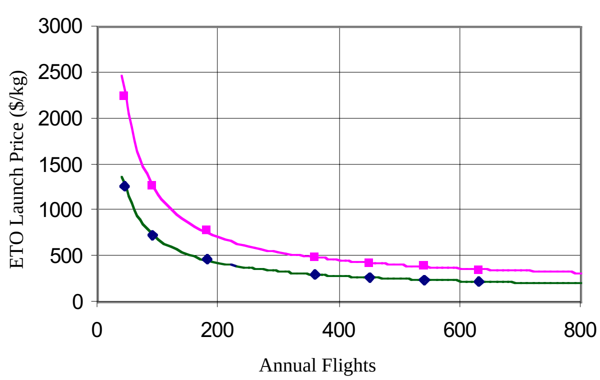

Figure 10 - ETO Launch Price.

$$\text{Launch Price} = \text{L} + (\text{H - L})\left(\frac{\#\ \text{flights}}{450}\right)^{-a} \quad (1)$$

An important factor in determining the total number of ETO flights is the Flight Rate Factor (FRF). This factor accounts for the additional flights needed to launch the SSP satellite above what would be expected for just the mass of the satellite. This includes the effects from the extra mass of the OTV and its propellant as well as the lost mass (i.e. payload capacity not used) due to volume constraints in packaging.

An example of the FRF is discussed for the *Swarm* solar electric propulsion concept (described later). The *Swarm* concept was chosen to investigate in detail how an Abacus/Reflector SSP satellite might be packaged in an ETO payload bay. NASA-MSFC personnel packaged the payloads using CAD models of SSP components and airborne support equipment (ASE) within a volume constraint of a 7 m diameter by 10 m cylinder and a mass constraint of 24 MT. The OTV was assumed to take up 2 m of the 12 m length of the payload bay and 14 of the total 40 MT ETO payload capability. The ASE was assumed to have a mass of 2 MT. The results for several of the major SSP subsystems is shown in Table 4.

This example shows that this factor is typically in the 2-3 range. A simplified expression for estimating the FRF, not accounting for volume constraints, Equation (2), can be used for concepts with less definition:

$$FRF = \frac{m_{payload}F + m_{propellant}F + m_{OTV\,dry}}{m_{payload}F} \quad (2)$$

Here F is the annual flights per OTV. Since this value does not account for volume constraints, it represents a lower bound on the FRF. Dividing by a packaging factor, 85% for

Table 4 - Abacus/Reflector Packaging.

Major SSP Abacus Subsystem	Subsystem Mass (MT)	SSP mass-based Launches (Mass)	Packaging-based Launches	Flight Rate Factor
Primary structure	3563	89	-	-
Solar array and PMAD	8034	201	589	2.93
Secondary structure	129	3	-	-
Transmitter modules	7030	178	375	2.11
Backing structure	281	7	60	8.57
Reflector	961	24	-	-
Other systems	1035	26	-	-
Total	21133	528	>1024 (~1200)	>1.94 (~2.27)

example, provides a better estimate. Figure 11 shows the combined effect of FRF and the ETO IRR on the ETO launch price.

The production cost was determined by multiplying an average unit cost by the dry weight of the OTV. The average unit cost for the vehicle was determined by applying average unit costs to each of the subsystems. Table 5 below shows the values used in this study. These values were arrived at through discussions with engineers from SAIC. The results for the vehicles studied (ranging from $1500 to $3600 per kilogram) were typical for vehicles produced in large quantities (lot sizes larger than 300 units).

The operations cost, completing the LCC, was set at a flat $1M per flight. The in-space launch price was determined next, dividing the LCC by the total payload, 20000 MT, and then applying the price-to-cost ratio. For this study, the price-to-cost ratio was set at 2. Previous experience shows that values for this could be as high as 4 for a project with 25% IRR and large up-front costs.

Another important metric, derived from the SSP goals, is the total transportation cost (the sum of ETO and in-space costs) expressed in cents per kilowatt-hour. This metric was calculated by multiplying the mass of an SSP satellite by the sum of the two launch prices and dividing by the total power produced by the satellite over its life (assuming continuous operation at 1.2 GW for 40 years).

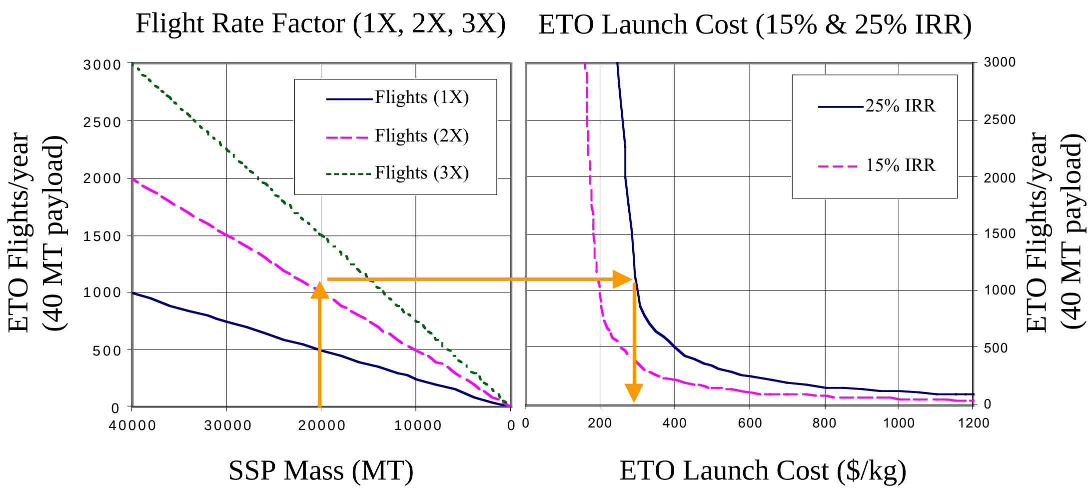

Figure 11 - Flight Rate Effects on Price.

RESULTS

Phase I In-space Concepts

Nuclear Thermal Rocket

The nuclear thermal rocket (NTR), Figure 12, is a highly reusable (100 flight) OTV that employs a propulsion system characterized by both high thrust (engine T/W of 5) and high specific impulse (Isp of 954.4 sec). These characteristics, combined with a long life airframe, dramatically reduce required fleet size for the high flight rate mission model. With a turnaround time estimated to be on the order of 2-3 days, each OTV can complete 100 flights within a year, resulting in a required fleet size of only two vehicles.

The NTR uses a reactor core of graphite-moderated uranium to heat the hydrogen propellant to a very high temperature before exhausting the hot gas through a high area ratio (200:1) convergent-divergent nozzle. The propellant is stored in a lightweight cryogenic fuel tank and pumped through the reactor with high operability-margin turbopumps. Lightweight metal matrix composite primary structures reduce the IMLEO. When sized for a 120 MT payload (three ETO "chunks"), gross mass is around 233 MT and dry mass is around 23 MT. Some vehicle characteristics are listed here along with economic metrics in Table 6:

- Dry mass = 23.3 MT
- H_2 mass = 90.2 MT
- IMLEO= 233.5 MT
- Structural mass fraction = 0.205
- Propellant fraction = 0.386
- Mass ratio = 1.63
- Flight rate factor = 1.75
- Average unit production cost = $3346/kg

Table 5 - Production Costs.

OTV Subsystem	Average Unit Cost
Main Propulsion	$6000/kg
Solar Arrays	$1000/kW
Cryogenic Propellant Tanks	$500/kg
Avionics	$3000/kg
Primary Structure	$1000/kg

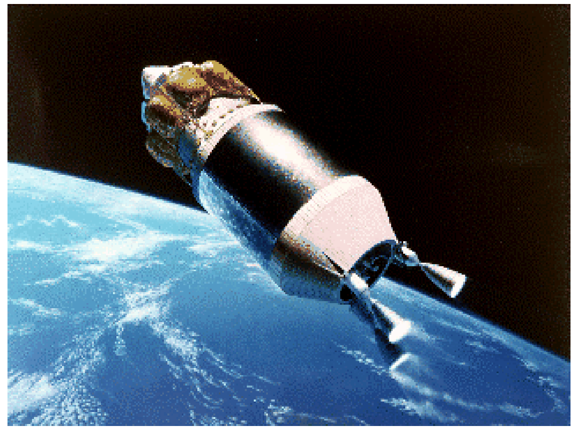

Figure 12 - Nuclear Thermal Rocket.

Using a nuclear system to put in place a non-nuclear alternative energy source doesn't make much sense from a political point-of-view. However, the NTR has several desirable characteristics that make it an attractive option, namely high thrust and high Isp. This concept, therefore, provided valuable insight into the characteristics and requirements necessary for any concept to meet the aggressive price goals set by this project.

Table 6 - NTR Metrics.

Transportation Cost	15% IRR	25% IRR
Per Unit Mass ($/kg)	1078	1363
Per Unit Energy (¢/kW-hr)	5.12	6.48

Solar Thermal Rocket

The solar thermal rocket (STR) is also a highly reusable (100 flight) OTV that employs a propulsion system characterized by higher thrust engines and a longer life airframe relative to solar electric propulsion (SEP). These characteristics help the STR to reduce the fleet size. However, turnaround time was estimated to be on the order of 60 days, allowing only 6 flights per tug per year, and requiring a fleet size on the order of 160 vehicles.

The STR, with a specific impulse of 766 sec, at first glance appears comparable to the NTR since the mechanism for propulsion is the same. Thrust is generated when very hot hydrogen is expanded through a convergent-divergent nozzle. However, the energy source is different and this limits the STR in power. As a result, the STR must store its collected solar energy by heating a graphite/rhenium block over the period of an orbit prior to each engine firing. This results in an orbit transfer, shown in Figure 13, consisting of 130 perigee burns to raise the apogee followed by 30 apogee burns to circularize the orbit. This transfer orbit accounts for the significantly slower transfer time and the larger fleet size. Additionally, a 5% penalty was added to the ideal ΔV to account for gravity losses. When sized for a 20 MT payload, gross mass is around 80 MT and dry mass is around 18.5 MT. Some vehicle characteristics are listed here along with economic metrics in Table 7:

- Dry mass = 18.5 MT
- H2 mass = 41.5 MT
- IMLEO = 80.0 MT
- Structural mass fraction = 0.308
- Propellant fraction = 0.519
- Mass ratio = 2.08
- Flight rate factor = 3.08
- Average unit production cost = $1500/kg

Table 7 - STR Metrics.

Transportation Cost	15% IRR	25% IRR
Per Unit Mass ($/kg)	2360	2836
Per Unit Energy (¢/kW-hr)	11.22	13.48

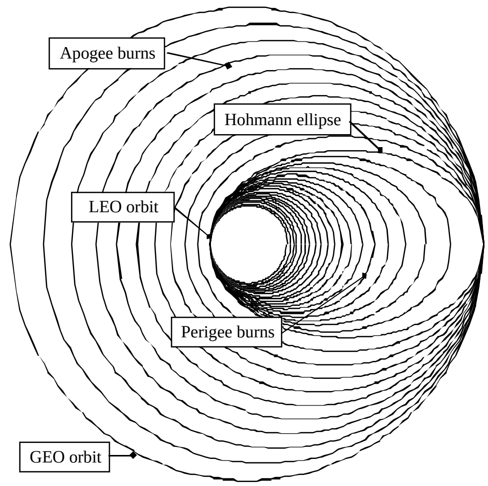

Figure 13 - STR Trajectory.

Momentum Exchange Tether

The spinning tether provides nearly propellantless deployment through momentum exchange. A single stage system was studied with circularization at GEO accomplished by an expendable upper stage. The tether was chosen because of this promise of a propellantless transfer, but did not live up to its promise since a significant source of energy is needed to re-boost the tether's orbit after every "toss" and to spin-up and spin-down the tether to the proper rotation rates. Additionally, NASA personnel noted that a single tether may not be feasible for transferring large payloads directly from LEO to GEO, since the centrifugal force on the tether at these high rotation rates would exceed the tensile strength of the tether. A two-stage tether system is then required, involving two facilities permanently in orbit, one in LEO and the other in Medium Earth Orbit (MEO).[2] A two-stage tether increases the complexity of the system due to the rendezvous requirements of multiple "snatches" and "tosses". A more detailed analysis of tether dynamics is required to properly treat this concept.

A single tether system, 60 km in length, was estimated to have a turnaround on the order of 7 days, due to the time required for re-boost, spin-up, and spin-down. When sized for a 68 MT payload, gross mass is around 387 MT and dry mass is around 318 MT. Most of the dry weight is ballast mass needed for proper momentum transfer.

Swarm (Solar Electric Propulsion)

The solar electric propulsion (SEP) concept studied was a very low cost one-way, expendable OTV pre-packaged with SSP payload on the ground. The concept presented here is a Georgia Tech design named *Swarm* (Figures 14 and 15). Powered by 18 Hall effect thrusters, *Swarm* has a high specific impulse (Isp of 2500

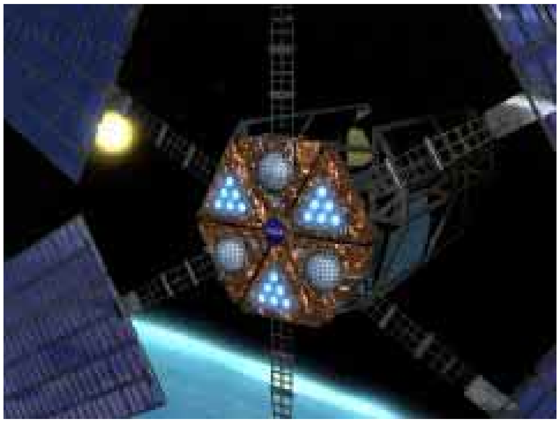

Figure 14 – Swarm.

sec) resulting in low propellant costs per flight. The low thrust (16 N), spiraling trajectory, however, requires a 116 day outbound transit at a m_0/p_0 of 200 kg/kW.

Swarm was selected for use in an effort to determine specific packaging requirements of an Abacus/Reflector SSP satellite. The SEP concept was chosen for this effort because of the high synergy with SSP systems and because the expendable concept reduces in-space operations costs. SSP component subsystems were packaged by NASA-MSFC into the *Argus* payload bay (7 m diameter by 12 m cylinder). Figure 16 shows the packaging of the solar arrays and Power Management and Distribution (PMAD). This effort showed that many of the common technologies with the SSP satellite. Because of the high degree of synergy, SSP systems integration personnel considered salvaging the OTV thrusters and extra propellant for SSP station-keeping. Engineers at NASA – Glenn Research Center determined that 100 N of continuous thrust was needed to counter-act perturbations due to solar pressure. At 16 N of thrust per OTV and approximately 800 OTVs required to place a single SSP satellite in GEO, these thrusters were not a good match for this purpose, due to excess thrust available, and that idea was rejected. Some vehicle characteristics are listed here along with several economic metrics in Table 8:

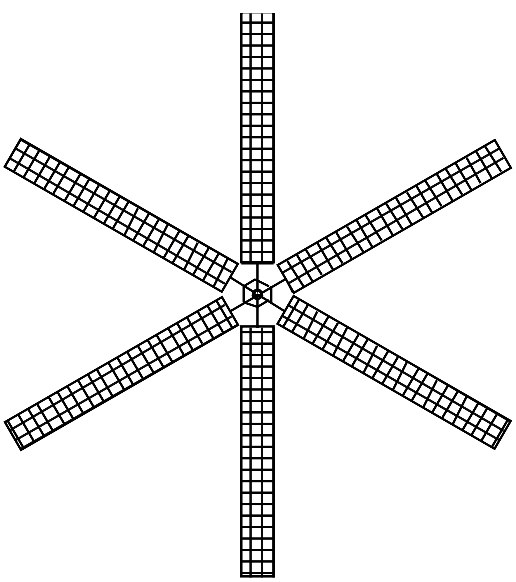

Figure 15 - Solar Electric Propulsion, Swarm.

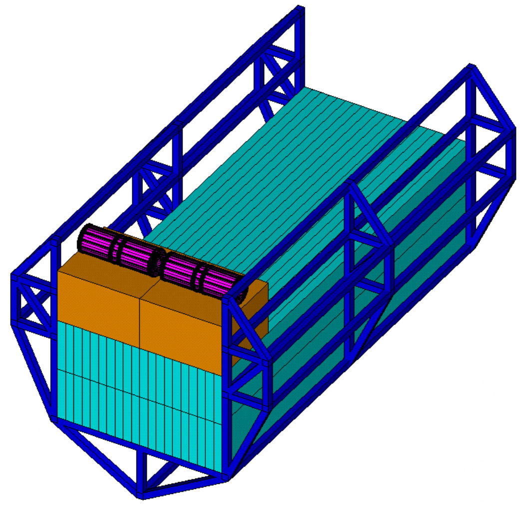

Figure 16 - SEP Payload Packaging.

- Dry mass = 7.6 MT
- Krypton mass = 12.8 MT
- IMLEO= 40.0 MT
- Payload mass = 19.6 MT
- Structural mass fraction = 0.373
- Propellant fraction = 0.319
- Mass ratio = 1.47
- Flight rate factor = 2.22
- Average unit production cost = $1869/kg

Table 8 - *Swarm* Metrics.

Transportation Cost	15% IRR	25% IRR
Per Unit Mass ($/kg)	2693	3048
Per Unit Energy (¢/kW-hr)	12.80	14.49

SEP systems had the most detail in their cost process. Looking strictly at a per unit basis, the cost estimation process revealed that SEP systems normally have higher non-recurring costs and lower recurring costs (as a percentage of overall system costs) than NTR systems. This is due to the fact that an actual SEP transportation unit consists of many concentrators, thrusters, arrays, etc. This construction of the SEP unit through combinations of such "sub-units" caused learning curve effects increasingly to become an important contributor to SEP's lower recurring cost. Concurrently, there are increased systems integration costs in the development phase of such systems.

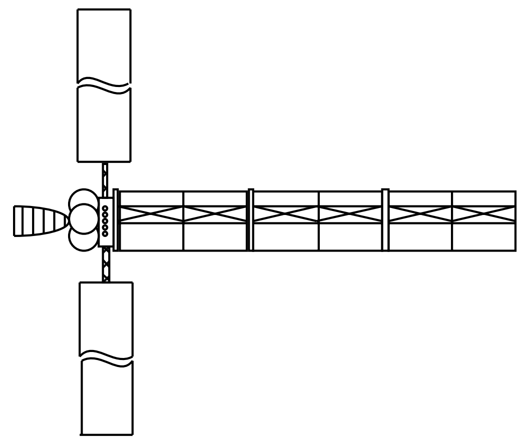

Figure 16 - Dual-Mode Propulsion.

Phase II In-space Concepts

None of the final four concepts examined in the first phase of the study meet the SSP goals with baseline cost, programmatic, and performance assumptions. Therefore a second phase of study was pursued to determine the required assumptions necessary to achieve the goals. Two new concepts were examined: Dual-Mode and All-Chemical. In addition, the *Swarm* concept was revisited and refined.

Dual-Mode (Chemical and SEP) Propulsion

This concept, proposed by John Mankins at NASA - Headquarters, is a dual-mode (chemical/SEP) propulsion system. The OTV uses high thrust, LOX/LH2, liquid rocket engines for the outbound transfer to GEO. The return leg is completed using high Isp, Hall effect electric propulsion. This concept sought to combine the advantages of a high thrust, quick turn-around system with a fuel-efficient high Isp system to provide an economical re-usable transportation system. The turnaround time for this concept was estimated to be on the order of 16 days, with 14 days required for the low-thrust return spiral.

For this concept, and the all-chemical concept discussed next, the propellant was assumed to be available on-orbit at a pre-existing orbiting propellant processing and storage facility. The propellant was assumed to be available, in this fashion, at a price equivalent to only placing the mass of the propellant in orbit at the same ETO launch price used by SSP and OTV Corp. Additionally, the effect of a lower price, \$50/kg, was investigated. This price was assumed to be the result of a currently undefined and hypothetical, extremely low-cost, method of placing water ice in orbit (the water then being converted to useful propellants through electrolysis in the orbiting fascility).

The vehicle, sized for three "chunks", carries 120 MT of payload in a river barge fashion. Some vehicle characteristics are listed here along with economic metrics and sensitivity to \$50/kg propellant costs in Table 9:

- Dry mass = 6.0 MT
- H2/O2 mass = 163.8 MT
- Krypton mass = 1.2 MT
- IMLEO= 291.0 MT
- Structural mass fraction = 0.035
- Propellant fraction = 0.567
- Mass ratio = 2.31
- Flight rate factor = 2.38
- Average unit production cost = \$2129/kg

Table 9 shows that the Dual-Mode comes very close to meeting the 2.5¢/kW-hr goal when propellant costs are limited to \$50/kg. Figure 17 compares the LCC breakdown of the *Swarm* and

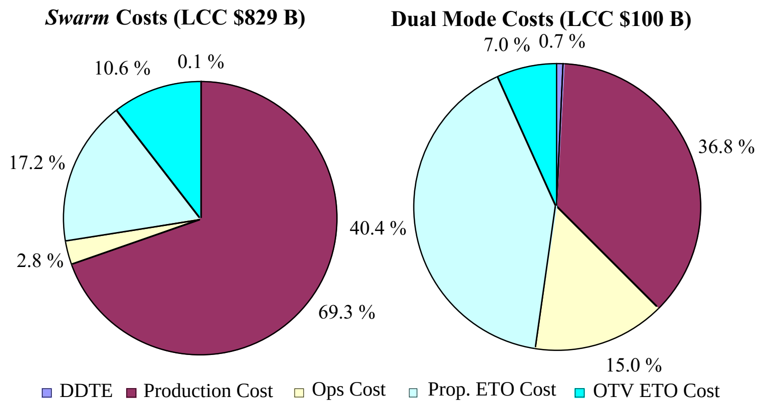

Figure 17 - Life Cycle Cost Breakdown.

Dual-Mode OTVs. The expendable *Swarm* is dominated by production cost. In contrast, LCC of the re-usable dual-mode OTV is dominated by ETO transportation costs, with propellant transportation being the largest contributor. Both pie charts show that operations costs and DDT&E are small for projects as large as SSP.

<u>All Chemical Propulsion</u>

This concept was analyzed as a companion to the Dual-Mode concept described above. The purpose was to determine the penalty for removing the high Isp return propulsion trip and replacing it with two additional LOX/LH2 burns. Results showed that the dry weight penalty for removing the SEP system and increasing the size of the cryogenic propellant tanks was negligible. The vehicle is sized for the same payload as the Dual-Mode concept.

Some vehicle characteristics are listed here along with economic metrics in Table 10:

- Dry mass = 6.0 MT
- H2/O2 mass = 179.9 MT
- IMLEO = 305.9 MT
- Structural mass fraction = .032
- Propellant fraction = .588
- Mass ratio = 2.43
- Flight rate factor = 2.50
- Average unit cost = $2134/kg

Again, this concept comes very close to meeting SSP goals if propellant is available for $50/kg. Figure 18 compares the economics of SEP (*Swarm*), Dual-Mode, and All-Chemical. The total transportation price is the combined ETO and in-space prices. SSP goals of $800/kg combined and 2.5¢/kW-hr for transportation are shown. The lines represent different masses for

Table 9 - Dual-Mode Metrics.

Transportation Cost	15% IRR	25% IRR	Propellant @ $50/kg
Per Unit Mass ($/kg)	846	1223	581
Per Unit Energy (¢/kW-hr)	4.02	5.81	2.76

Table 10 - All-Chemical Metrics.

Transportation Cost	15% IRR	25% IRR	Propellant @ $50/kg
Per Unit Mass ($/kg)	881	1276	596
Per Unit Energy (¢/kW-hr)	4.19	6.06	2.83

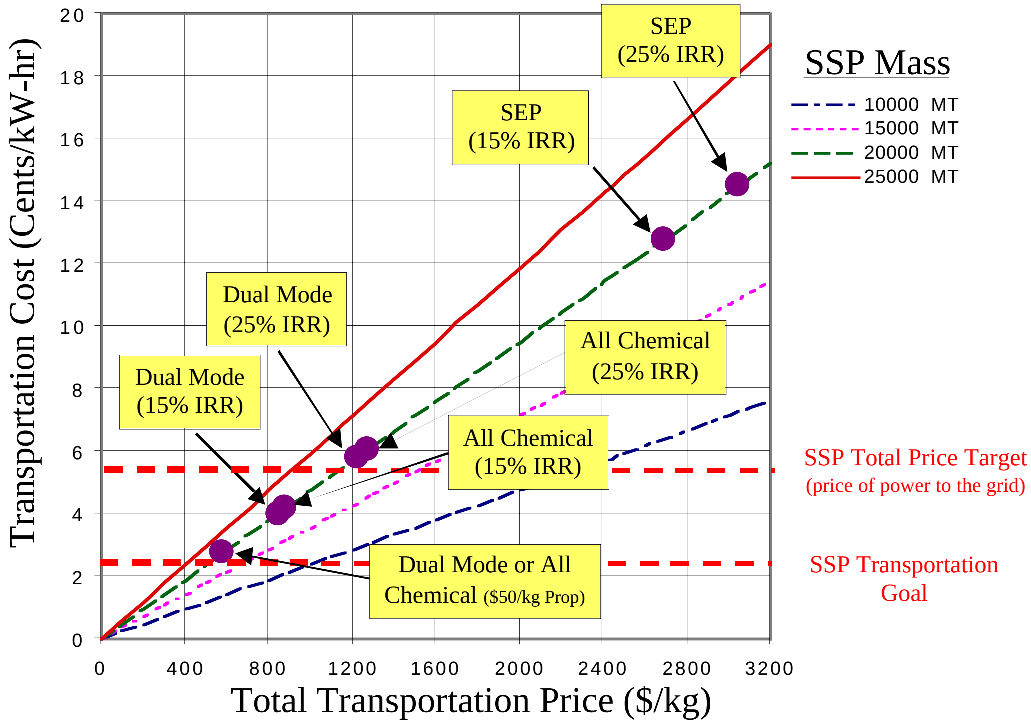

Figure 18 - Concept Economic Comparison.

SSP satellites. The points shown assume a 20,000 MT mass. The sensitivity to extremely low propellant costs ($50/kg delivered to LEO) is also shown in Figure 18.

SUMMARY

The results from the ETO study supported the conclusion that, the GEO-based SSP scenario produced an attractive economic scenario for a potential RLV developer, even when revenues are limited to only $400/kg. With proper support from the government, the sustained, high traffic mission model from the SSP creates a steady revenue source that enables RLV, Inc. to recoup startup costs, offer lower prices, and still provide an adequate return on investment of 15 to 25%.

Results from the in-space study showed that the most economical OTV is highly reusable, has a short turn-around time, a long vehicle life, and low propellant cost requirements. These characteristics result in a low fleet size and therefore lower debt requirements. These characteristics also lower the Initial Mass in Low Earth Orbit (IMLEO) and therefore lower deployment costs. The goal of $400/kg is very aggressive and difficult to achieve.

Finally, end-to-end price in $/kg, is not the best metric for measuring SSP economic success in transportation. This is because of the inverse relationship of price and total cost (the product of price and number of flights) with changes in required ETO flights. A better metric is total transportation cost expressed in cents per kilowatt-hour. Several of the concepts showed combined launch prices within or near the $800/kg goal without achieving the 2.5¢/kW-hr needed to meet the 5¢/kW-hr goal, needed for SSP economic viability.

RECOMMENDED DESIGN GUIDELINES

1. Small Fleet Size/Long Life

Low fleet sizes are of paramount importance in reducing in-space transportation costs to SSP through reductions in OTV procurement costs. Fleet sizes are kept at a minimum by utilizing highly reusable OTVs with high thrust. High thrust systems lower the trip times and maximize the number of times an OTV can be reused within the time frame required to place one satellite on station. Expendable systems are not competitive because of the dominance of total production costs on LCC even with an assumption of very low unit production costs.

2. Low Propellant Costs

Low propellant costs are crucial to reducing in-space transportation costs through reductions in ETO transportation costs borne by the in-space provider. Propellant costs are kept low by reducing the mass and/or the cost of the propellant. The mass is reduced through high specific impulse systems. The cost is reduced by utilizing propulsion systems which use abundant (cheap) propellants and/or very low-cost ETO transportation systems for bulk transportation of propellant.

3. Reduce IMLEO

Reducing IMLEO is important to reducing in-space transportation costs because of the direct dependency on ETO transportation costs. IMLEO is reduced through low fleet sizes (see 1 above) and low propellant mass (see 2 above). Reducing the FRF for a concept results in fewer ETO flights and therefore a higher ETO transportation price on a unit basis, but lower overall ETO transportation costs.

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