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Sirius: A New Launch Vehicle Option for Mega-LEO Constellation Deployment

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ABSTRACT

Tremendous growth in the satellite communication market is expected within the next decade. In particular, the market for services based in low earth orbit (LEO) is booming. Large constellations of satellites will soon be deployed with capabilities ranging from modest paging services to high bandwidth, data transfer systems. Constellations in the latter category are referred to as Mega-LEO constellations. Deployment of Mega-LEO constellations will place tremendous demands on international launch capabilities. Current expendable booster capability, reliability, availability, and price are all issues.

This research tests the hypothesis that a new, low cost launch vehicle can be developed specifically to deploy a Mega-LEO constellation and still be economically competitive. A fictitious Mega-LEO constellation called *Orion* was created to set mission requirements. Aggressive launch cost goals and launch rates were established.

A new two-stage system with a reusable booster was designed to meet the challenge — *Sirius*. This paper includes the results of the conceptual vehicle design activity including both technical and economic data. Details on the multidisciplinary design optimization methodology employed are also included.

NOMENCLATURE

ACC	advanced carbon-carbon
APAS	Aerodynamic Preliminary Analysis System
ASDL	Aerospace Systems Design Laboratory
B	billion dollars (1996 U.S. dollars)
CABAM	Cost and Business Assessment Module
CAD	computer aided design
DoE	design of experiments
DSM	design structure matrix
GA	genetic algorithm
GEO	geostationary orbit
IRR	internal rate of return
LEO	low earth orbit
LH2	liquid hydrogen
L/Lref	ratio of vehicle length to reference length
LOX	liquid oxygen
M	million dollars (1996 U.S. dollars)
MDO	multidisciplinary design optimization
MR	mass ratio (gross weight/burnout weight)
OBD	optimization-based decomposition
POST	Program to Optimize Simulated Trajectories
RSE	response surface equation
RSM	response surface methodology
SSTO	single-stage-to-orbit
TDS	tether deployment system
Ti-Al	titanium-aluminide
TOS	transfer orbit stage
TPS	thermal protection system
TRF	technology reduction factor
TSTO	two-stage-to-orbit

INTRODUCTION

With the continued expansion of global telecommunications markets, many companies have realized the tremendous potential of using constellations of low to mid earth orbiting satellites to

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provide real time voice and data transmission to all parts of the globe. Compared to communications from geostationary orbit (GEO), low earth orbit (LEO) communication offers shorter signal lag time and lower power requirements. However, LEO-based satellites have smaller signal footprints and faster orbital velocities (not synchronized with the Earth's rotation). Therefore, LEO communications systems must include a number of satellites in many different orbital planes to achieve global or near-global coverage. These satellite 'constellations' work together with ground stations to form an integrated communications system.

LEO constellations of communications satellites have been categorized into three broad classes by number of satellites, satellite size, and constellation data capacity[1]. As shown in Table 1, 'Little-LEO' systems are targeted primarily at the global paging and messaging market. 'Big-LEO' systems include voice communications systems (i.e. satellite phones). 'Mega-LEO' constellations are very large systems capable of high bandwidth, broadband voice and data communications — an 'Internet-in-the sky'.

Table 1 - LEO Satellite Constellations

Class	Sat. Mass	# Sats	Market
Little-LEO	< ~100 lb.	< 50	messaging
Big-LEO	< ~300 lb.	< 100	voice
Mega-LEO	> 1000 lb.	> 200	broadband

Some constellations are already being deployed. Orbital Sciences' Little-LEO *Orbcomm* system for global paging and messaging already has a partial constellation in orbit (a few satellites) and is generating revenue. Motorola's Big-LEO *Iridium* system for global voice communications launched it's first satellites earlier this year. *Orbcomm* will be deployed by Orbital Sciences' own Pegasus and Taurus launchers. *Iridium* will be deployed by a combination of the Delta and international expendable launch systems.

Follow-on systems around the year 2000 envisioned by companies such as Teledesic will significantly increase the LEO data transmission capability. These Mega-LEO systems will require more, larger satellites operating in many orbital planes. By current estimates, Teledesic's Mega-LEO constellation will require 324 satellites of 2750 lb. each[2]. To initiate revenues, recoup startup costs, and be first to market, operators of a Mega-LEO constellation need their satellites to be launched quickly and efficiently — preferably within one or two years.

Timely deployment of such a large constellation is expected to quickly over-tax the available domestic and international launch capability. The current total international launch capacity is only approximately 45 - 50 launches per year (about 800,000 lb. to LEO)[3]. Of this total, most of the launches are already dedicated to government and military payloads, science missions, space station support, and other commercial launches. No more than a few launches per year would be available to support rapid deployment of a Mega-LEO constellation. Launch vehicle manufacturers are reluctant to invest in additional launch rate surge capability to meet what is perceived to be a one time launch requirement. (Mega-LEO maintenance and replenishment launches are expected fall well within today's launch capabilities). The current launch industry's solution to Mega-LEO deployment is to stretch the deployment to 5 - 6 years, manifest multiple satellites into larger launch packages, and use a mixed fleet of domestic and international boosters (e.g. Atlas, Delta, Proton, Ariane)[4].

Using existing launch vehicles, the cost of Mega-LEO deployment is expected to be high. Current expendable launchers place payloads into LEO for \$3,000 - \$4,000 per lb. of payload. Total launch cost for an entire constellation might be as high \$1B - \$1.5B.

This research tests the hypothesis that a new low cost launch system can be designed specifically for deploying a Mega-LEO constellation and still be profitable. This new system would (initially) be dedicated to launching the Mega-LEO constellation so competition for existing launch resources would be eliminated. The new system would also have the capacity to launch all satellites within two years. To be economically competitive, the *total* cost for design, development, hardware acquisition, and operations of the new vehicle would have to be the same or lower than the cost of deploying the constellation on existing boosters. This very aggressive cost constraint implies that the new system minimize new technology development cost, use streamlined design and development methods, have very low manufacturing costs, and use a minimal cost operations strategy.

ORION

To quantify mission requirements for the launch vehicle design challenge, a fictitious Mega-LEO constellation was assumed. The specifications for the *Orion* Mega-LEO constellation are given in Table 2 and Figures 1 and 2.

Table 2 - Orion Constellation Requirements

Number of Satellites	400
Satellite Mass	1,575 lb. (ea.)
Number of Orbital Planes	20
Operational Orbit	420 nmi circ.
Orbital Inclination	85°
Deployment Period	2002 - 2004
Replenishment Period	2004 -2014
Launch Price Target	$3M per sat.

The *Orion* constellation includes characteristics typical of Mega-LEO constellations. It consists of 400 identical, high bandwidth, communications satellites deployed to 20 different orbital planes. Each orbital plane contains 20 equally spaced satellites in 420 nmi. circular orbits at 85^o inclination. Each satellite weighs 1,575 lb. First launch of the initial constellation deployment is to be required by January 2002 and deployment must be completed within two years. Launch is expected to be on a safe and reliable vehicle.

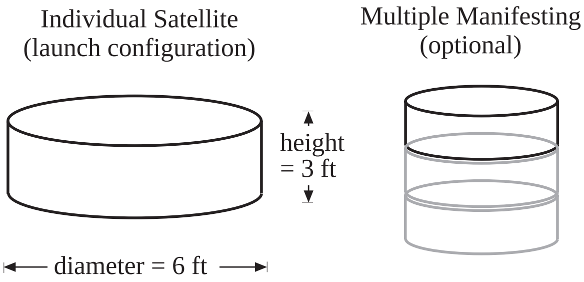


Figure 1 - Orion Satellite Physical Configuration

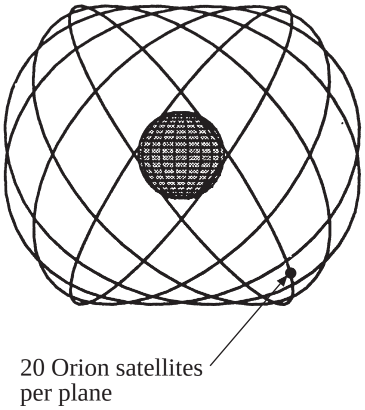


Figure 2 - Notional Orion Orbital Planes

Orion will pay a fixed deployment price of $3M per satellite for a total of $1.2B for the initial deployment. Periodic replenishment of spares will require an additional 20 satellites per year for 10 years following initial deployment. The cost of each replacement satellite is also $3M per satellite (for $0.6B in additional revenue). Some amount of this money will be available up front to develop a new system. Multiple manifesting of satellites is allowed.

The challenge of designing a new, low cost launch system for *Orion* was presented to a group of aerospace engineering graduate students at the Georgia Institute of Technology. The vehicle design data reported in this paper is a summary of their response.

VEHICLE DESIGN

Mega-LEO deployments offer very challenging goals for performance, operability, and affordability. In order to meet these goals, the *Sirius* design has made use of existing hardware, current technologies, and reusability. The design methodology employed sought to accomplish two objectives: integrate cost early in the design process, and delay as long as possible many of the design decisions. By thus preserving flexibility in the design, the design team was able to quickly explore a large design space in an efficient manner. In this fashion the methodology provided, not only a viable point design, but also a better understanding of the design space which assisted in decision making.

Table 3 - Computational Tools

Solid Modeling	IDEAS
Aerodynamics	APAS[5]
Trajectory Optimization	POST[6]
Weights & Sizing	Excel
Optimization	Genalg.f[7]
Statistical Analysis	JMP
Line Drawings	Canvas
Cost & Business	CABAM[8]

ANALYSIS PROCESS

Brainstorming

The design team was allowed great freedom in choosing the configuration of the vehicle. During the brain-storming process, many diverse vehicle options were examined on a macroscopic scale. Configurations considered included: bimese launch vehicles, expendable rockets with strap-on boosters, wing body configurations, and lifting bodies. Additional ideas which could be incorporated into any configuration were tether deployment, air launch assist, or sea launch options. Due to the large number of launches required for constellation deployment, the design team considered a fully expendable system to be cost prohibative. Recovering the first stage of a conventional two-stage-to-orbit (TSTO) design was eliminated due to concerns for protecting the airframe and engines during re-entry (this option was assumed to require a high staging Mach number in order to increase the amount of re-useable hardware). The bimese was considered too expensive and complex to meet system goals. The lifting body was eliminated because the vehicle shape would lead to poor packaging efficiency and difficult center of gravity managment issues. Following a down-select screening process, the design team selected the wing body configuration as potentially the most economically viable option.

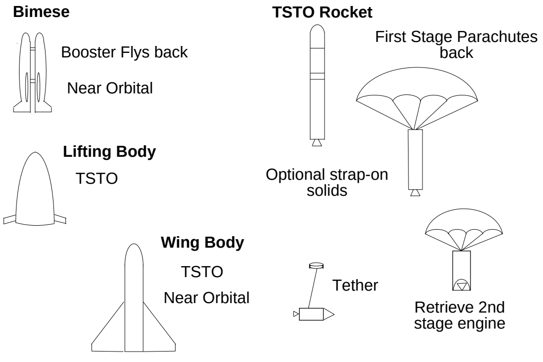


Figure 3 - Brainstorming Concepts

Design Variables

Once a baseline configuration was chosen, the team selected five design variables: number of satellites manifested per launch, first stage propulsion system, second stage propulsion system, structural materials, and booster fineness ratio. Table 4 provides a summary of all design variables and valid ranges.

Table 4 - Design Variables

Number of Satellites	1 to 9
First Stage Engine	RD-O120, RD-190, RD-120
Second Stage Engine	Orbus-21D, TOS, STAR-63D
Structural Material	Al, Graphite Epoxy, Ti-Al
Fineness Ratio	6, 7, 8

Number of satellites manifested represented a trade between vehicle performance and operational cost. It is obvious that this design variable must be an integer, since launching a fraction of satellite is not practical. The design of the orbital planes limited the maximum number of satellites per launch to twenty. However, pre-conceptual calculations showed that manifesting more than nine satellites per launch resulted in impractically large vehicles (well over 1M lb. gross weight). The first variable, satellite manifesting, was therefore set at integer values from one to nine satellites per launch.

Three existing engines were selected for possible use in the first stage booster. All three engines selected, the RD-O120, the RD-190, and the RD-120, were Russian designed engines. These engines brought to the design high performance at extremely low cost. Two hydrocarbon-fueled engines were selected because it was first suspected that the higher propellant bulk densities would result in a smaller dry weight, and therefore cheaper system. The RD-O120 was the only liquid hydrogen-fueled engine chosen for study. The engines were numbered engine one through three, respectively.

The second stage systems chosen were off-the-shelf expendable solid propellant motors. The three systems chosen were: the Orbus-21D, the Transfer Orbit Stage (TOS), and the STAR-63D. As with the first stage engine, these engines were numbered engine one through engine three, respectively.

The fourth design variable was structural material. Three materials were chosen for study: Aluminum, Graphite Epoxy, and Titanium-Aluminum (Ti-Al) alloy. These materials were also assigned a number from one to three, respectively. The effects of different materials were realized by applying Technology Reduction Factors (TRFs) to booster airframe component weights and airframe development and production costs. The Ti-Al material, considered "hot structure", had the additional benefit of reducing Thermal Protection System (TPS) requirements.

The last design variable chosen was fuselage fineness ratio (fuselage length-to-diameter ratio). This metric was chosen to study the effects of ascent aerodynamics on the vehicle design. The selected fineness ratios were: 6, 7, and 8. Each value of fineness ratio required a separate aerodynamic model.

Design Structure Matrix

Figure 4 shows the Design Structure Matrix (DSM) used in the variable trade studies. This figure shows the flow of information between the different disciplines and the optimizer. Vertical lines indicate input to each of the disciplines or modules. Similarly, the horizontal lines indicate outputs. The modules are evaluated sequentially in descending order from the upper left to the lower right. Therefore, lines above and to the right of the diagonal represent feedforward loops. The lines below and to the left of the diagonal are the feedback loops. Normally, there is an undesirable feedback between trajectory optimization and weights. In this work, Optimization Based Decomposition (OBD) is used to break internal feedback loops between disciplines which would otherwise require iteration. The feedback loop is broken by adding an additional intermediate variable in the optimizer and enforcing a compatibility constraint. In this case, the feedback of required mass ratio (MR req'd) from performance to weights and sizing was broken by adding the mass ratio guess, MR guess (variable #6) and enforcing the constraint that it be equal to MR req'd (#9).

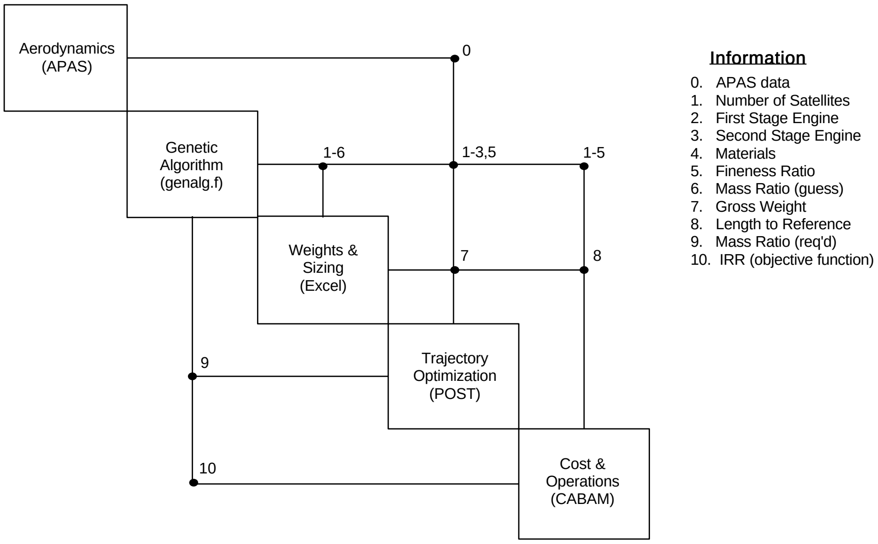


Figure 4 - Design Structure Matrix

Coupling Variables

Several other internal variables were important and required to provide coupling between modules. The first discipline, weights & sizing, required only an initial guess for the required mass ratio in addition to the five design variables. Performance received only four of the design variables, plus aerodynamic data for each of the fineness ratios, and the gross weight predicted from the weights & sizing module.

The third discipline, Cost, requires all of the subcomponent weights calculated by the weights & sizing module. This would have required a prohibitive number of coupling variables if all of this information had to be passed within the MDO environment. This undesirable situation was avoided simply by fact that the two analyses (weights & sizing and cost) were both run on the same computer platform (Excel spreadsheets in this case). A copy of the weights & sizing spreadsheet was literally cut and pasted onto the cost estimation tool, CABAM. By passing the design variables and the photographic scaling variable, length to reference length (L/Lref), CABAM could recreate any design and therefore quickly analyze the life cycle cost of the system. The photographic scaling ratio, L/Lref, was therefore generated by the weights & sizing module and fed forward to the cost module. The performance module returned required mass ratio and the cost module returned IRR to the optimizer. The required mass ratio was then compared to the initial mass ratio guess to enforce the compatibility constraint. IRR was utilized as the objective function of the optimization process.

Genetic Algorithm

The choice of variables demonstrates a common occurrence in engineering practice. It is very often the case that the design variables on which a team wishes to perform trade studies are integer values. This poses a problem when trying to perform optimizations because most optimization techniques do not handle integer variables well, if at all. For this study, all of the design variables were integers. Therefore, the logical choice of optimizer was a Genetic Algorithm (GA). The compatibility constraint was implemented through the use of an exterior penalty function within the objective function routine. Table 5 shows some of the settings for the GA.

Table 5 - Genetic Algorithm Settings

Population Size	300
Maximum Generations	40
Number of Seeds	5
Cross-over Probability	90%
Mutation Probability	10%

Face-centered central composite DOE experimental arrays were generated for the weights and sizing, performance, and cost analysis disciplines. From these three DOE experimental arrays, second-order polynomial Response Surface Equations (RSEs) were fitted to gross weight, L/Lref, MR req'd, and IRR. The use of Response Surface Methodology (RSM) allowed simple integration of several cross-platform disciplinary codes. Additionally, the resulting optimization could be executed very quickly.

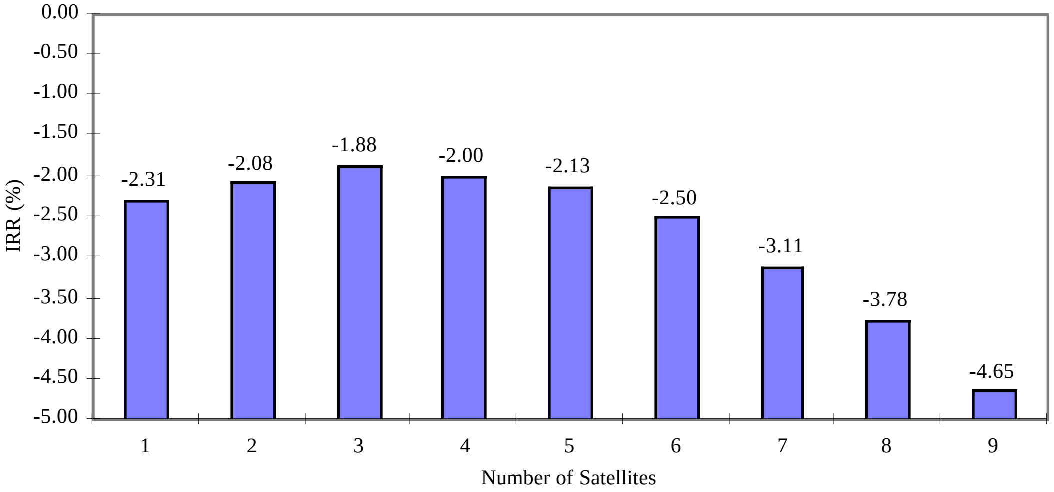
Response Surface Equations

The weights and sizing spreadsheet was used to generate RSEs for gross weight and photographic scaling ratio. Both responses were expressed as functions of six variables, (the five design variables and an initial MR guess). POST was used to generate an RSE for required MR. CABAM was used to generate an RSE for IRR. The IRR was expressed as a function of six variables: the five design variables and the photographic scaling ratio.

TRADE STUDIES

Once the response surfaces for each discipline were generated, the system-level optimization process began. Because the use of response surfaces allowed fast optimization, more information could be gained from the design space than just the optimal design variable settings. By sequentially constraining each of the design variables and re-performing the optimization, technology trade studies could be performed. That is, one design variable can be held constant and the other four design variables can be 'sub-optimized' around that setting. A sweep of these sub-optimums can be created for each design variable. In this manner, the designer can determine the effects on the objective function (IRR) and other vehicle parameters (such as gross weight) due to non-optimal design variable choices in a single variable assuming the others have been re-optimized.

Figure 5 illustrates the advantage of this type of information. As can be seen, the optimum manifesting is three satellites. However, with a constellation configuration of 20 satellites per orbit, an additional flight with a partial payload would be required for each orbital plane (6 flights of 3 satellites plus 1 flight of 2 satellites). Figure 5 shows that the penalty, in terms of program IRR, for selecting a manifestation strategy of 4 satellites per launch is less than 0.2%. The design team in this case determined that this penalty was acceptable when compared to the additional costs associated with the partially loaded flights. This decision extends even further since the optimal first stage engine is different for the two manifestation strategy choices. It follows that the designer must adopt the RD-O120 engine if it is desired to manifest 4 satellites per launch to avoid partial payload flights in the initial constellation deployment.



# Sat	FS Eng	SS Eng	Matl	Fineness	MR guess	Wg (klbs)	L/Lref	MR	IRR (%)
1	RD-O120	STAR-63D	Ti-Al	8	3.90	81.27	0.76	3.88	-2.31
2	RD-O120	STAR-63D	Ti-Al	8	4.54	154.26	0.85	4.50	-2.08
3	RD-190	TOS	Ti-Al	8	5.75	293.06	0.74	5.74	-1.88
4	RD-O120	TOS	Ti-Al	8	3.46	204.14	0.83	3.45	-2.00
5	RD-O120	TOS	Ti-Al	8	3.78	248.53	0.88	3.79	-2.13
6	RD-O120	TOS	Ti-Al	8	4.03	289.18	0.92	4.04	-2.50
7	RD-O120	TOS	Ti-Al	8	4.22	324.83	0.96	4.20	-3.11
8	RD-O120	TOS	Ti-Al	8	4.29	347.17	0.98	4.31	-3.78
9	RD-O120	TOS	Ti-Al	8	4.29	362.41	0.99	4.32	-4.65

Figure 5 - Satellite Manifesting Trade Study

An additional benefit of performing these type of trade studies is improved confidence in the optimization process. If only the design variable of interest is constrained and control of all other variables is assigned to the optimizer (as was done in this case), the optimum *within each design variable sweep* will also be the *global* optimum. Therefore, each design variable trade study should reproduce the same optimal design variable settings. Figure 10 illustrates how this information is useful. The optimum in this figure is for 4 satellites and the RD-O120 first stage engine at a fineness ratio of 8. In performing the optimization at 8, the GA did not select the combination of design variables with 3 satellites per launch and the RD-190 engine (as was selected in each of the other one-variable sweeps). We know from performing the additional optimization that this combination is feasible (meets the compatibility constraint) and has a better IRR. Had only one optimization been performed, the designers would have assumed (incorrectly) that the optimum shown in Figure 10 was the global optimum for and the absolute 'best' answer that could be found.

Table 6 summarizes the results of the trade studies. The optimum case is shown along with the design variable choices selected by the design team to select a fully manifested configuration.

Table 6 - Design Variable Selection

Design Variable	Optimum	Final Design
Number of Satellites	3	4
First Stage Engine	RD-190	RD-O120
Second Stage Engine	TOS	TOS
Structural Material	Ti-Al	Ti-Al
Fineness Ratio	8	8

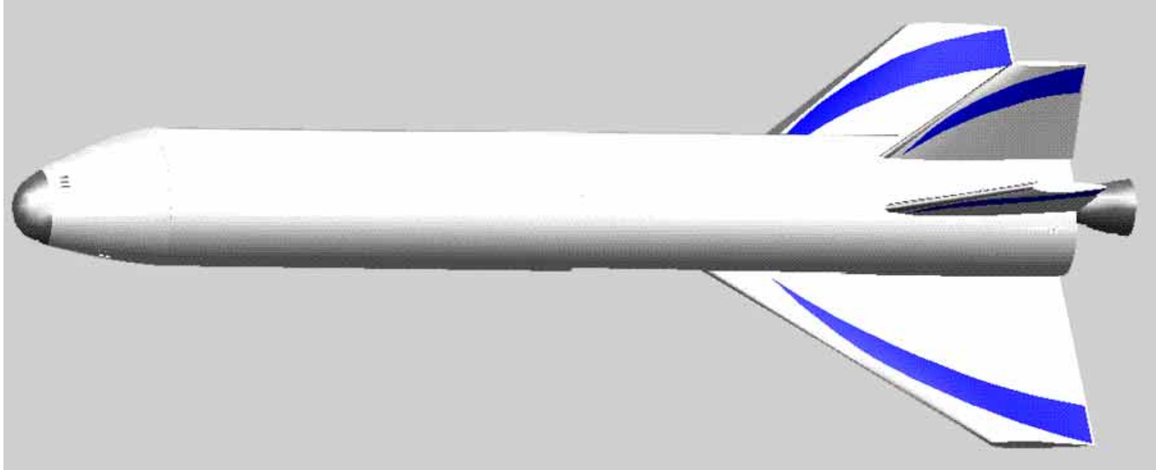
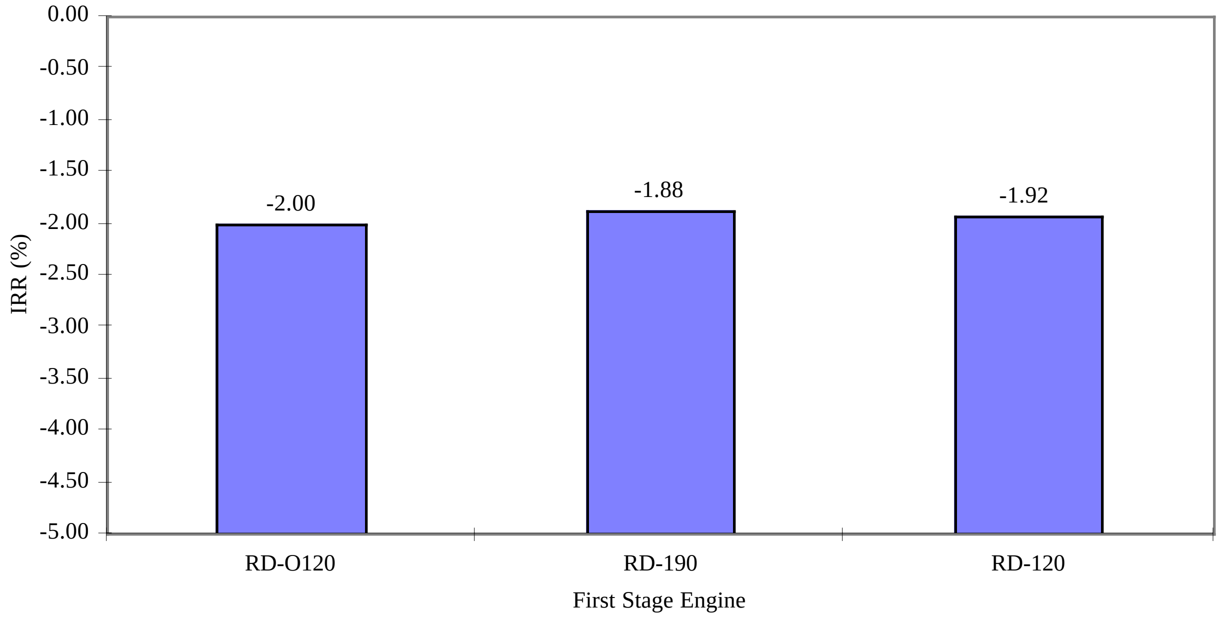
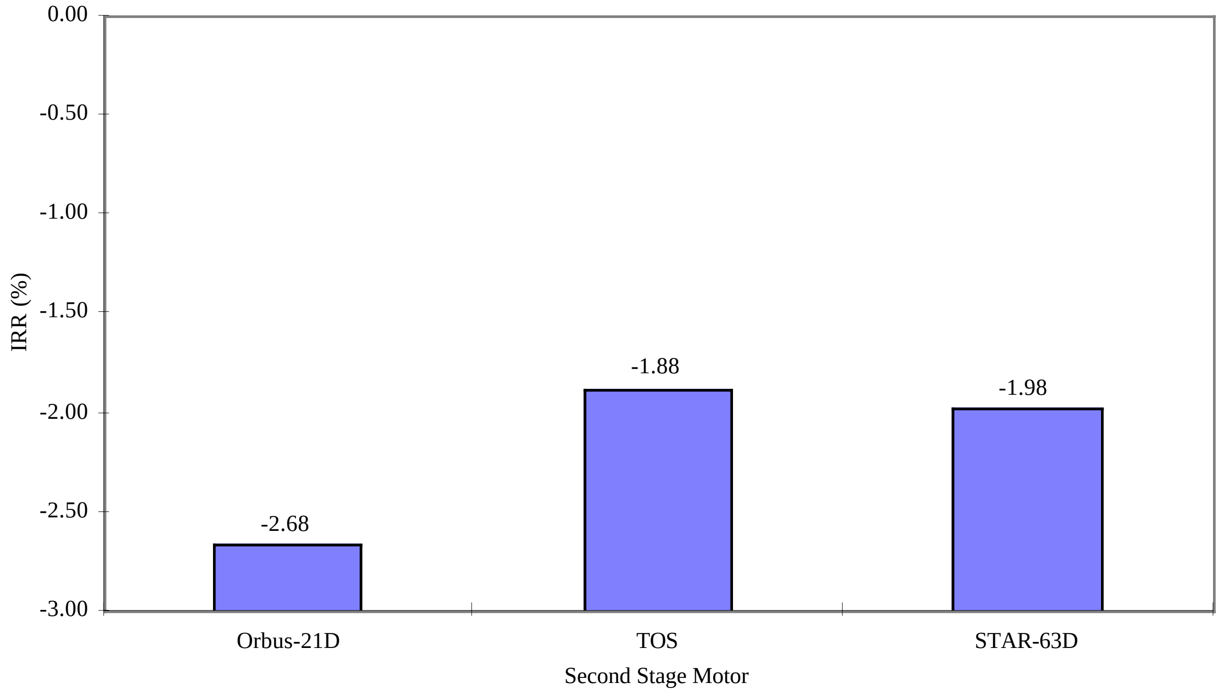


Figure 6 - Sirius Solid Model



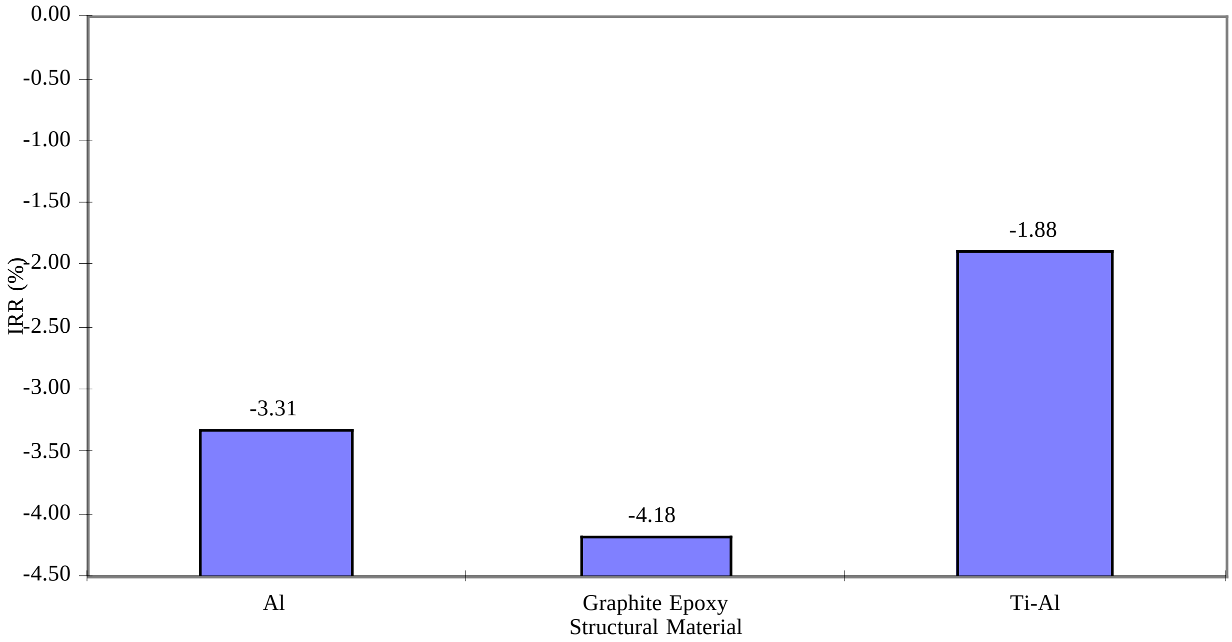
FS Eng	# Sat	SS Eng	Matl	Fineness	MR guess	Wg (klbs)	L/Lref	MR	IRR (%)
RD-O120	4	TOS	Ti-Al	8	3.46	204.14	0.83	3.45	-2.00
RD-190	3	TOS	Ti-Al	8	5.75	293.06	0.74	5.74	-1.88
RD-120	3	TOS	Ti-Al	6	5.94	309.33	0.85	5.96	-1.92

Figure 7 - First Stage Engine Trade Study



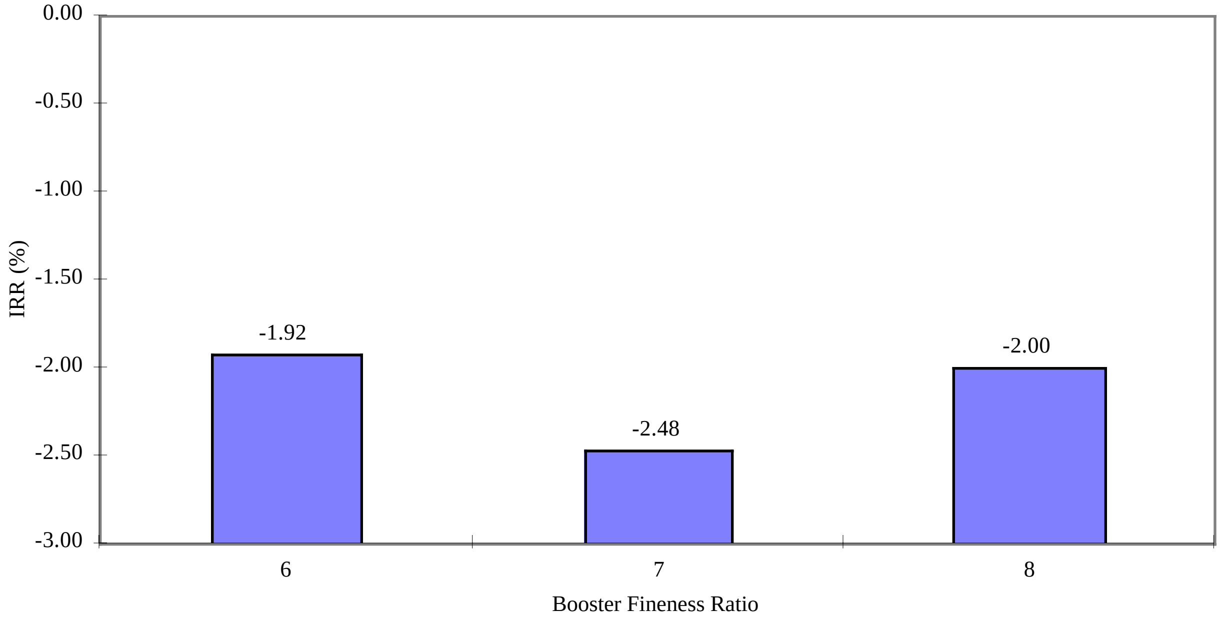
SS Eng	# Sat	FS Eng	Matl	Fineness	MR guess	Wg (klbs)	L/Lref	MR	IRR (%)
Orbus-21D	3	RD-120	Ti-Al	6	5.43	286.04	0.82	5.42	-2.68
TOS	3	RD-190	Ti-Al	8	5.75	293.06	0.74	5.74	-1.88
STAR-63D	3	RD-O120	Ti-Al	8	5.05	223.11	0.91	5.07	-1.98

Figure 8 - Second Stage Engine Trade Study



Matl	# Sat	FS Eng	SS Eng	Fineness	MR guess	Wg (klbs)	L/Lref	MR	IRR (%)
Al	2	RD-O120	TOS	8	2.25	167.81	0.67	2.24	-3.31
Graphite Epxy	2	RD-O120	TOS	8	2.38	140.26	0.68	2.35	-4.18
Ti-Al	3	RD-190	TOS	8	5.75	293.06	0.74	5.74	-1.88

Figure 9 - Structural Material Trade Study



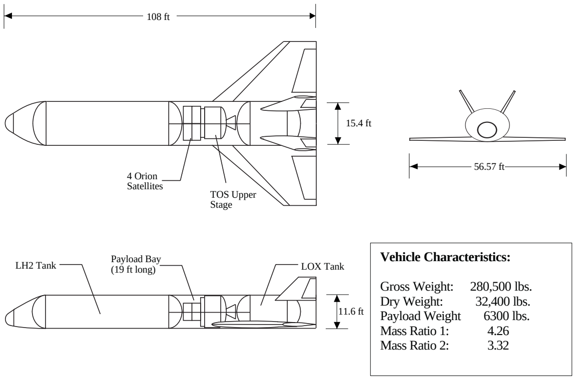
Fineness	# Sat	FS Eng	SS Eng	Matl	MR guess	Wg (klbs)	L/Lref	MR	IRR (%)
6	3	RD-120	TOS	Ti-Al	5.94	309.33	0.85	5.96	-1.92
7	3	RD-O120	TOS	Ti-Al	3.14	185.33	0.83	3.25	-2.48
8	4	RD-O120	TOS	Ti-Al	3.46	204.14	0.83	3.45	-2.00

Figure 10 - Booster Fineness Ratio Trade Study

VEHICLE DESIGN RESULTS

Figure 11 is the 3-view drawing of the converged *Sirius* concept corresponding to the selected design variables. *Sirius* is a two-stage-to-orbit (TSTO) all-rocket system. The first stage is a reusable wing body design powered by an RD-O120 LOX/LH2 engine. The second stage is an existing expendable Transfer Orbital Stage (TOS) solid propellant motor. A Tether Deployment System (TDS) is used to both circularize the satellites in the final orbit and de-orbit the expended upper stage. The TOS, the TDS, and four *Orion* satellites are carried in the booster payload bay during ascent.

Following a vertical liftoff from Wallops Island, VA, the booster returns un-powered for a horizontal landing in Roosevelt Roads, PR. The *Sirius* booster is post-processed and loaded on a C-5 transport aircraft for return to the Wallops facility.



Vehicle Characteristics:

Gross Weight:	280,500 lbs.
Dry Weight:	32,400 lbs.
Payload Weight	6300 lbs.
Mass Ratio 1:	4.26
Mass Ratio 2:	3.32

Figure 11 - Sirius 3-view

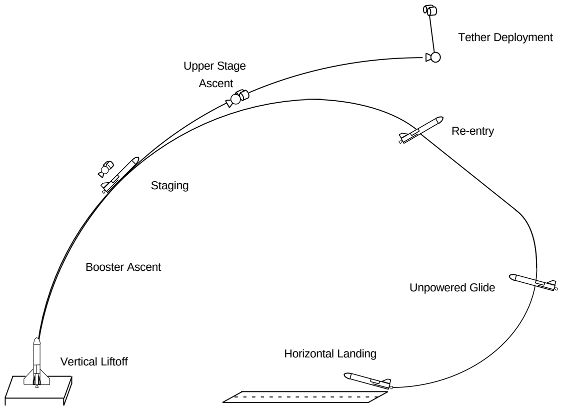


Figure 12 - Mission Flight Profile

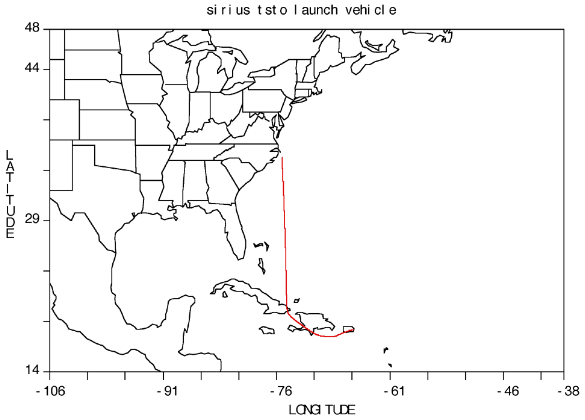


Figure 13 - Booster Re-entry Trajectory

The gross weight of the launch system is approximately 280,500 lb. and the dry weight is 32,400 lb. *Sirius* uses a moderate lift-to-drag wing body configuration which trims subsonically at 5° and hypersonically at 40° with body flap deflection. The booster required mass ratio is 4.26. The booster stages at 15,450 fps at 349,000 feet, and the upper stage inserts the payload into a transfer orbit of 417 nmi x 102 nmi x 85° inclination. The vehicle is primarily constructed of Ti-Al hot structure which eliminates the need for tile thermal protection system (TPS) over much of the vehicle. Areas that experience the highest heat rates are protected with advanced carbon-carbon (ACC).

A life cycle cost and business economic analysis using CABAM showed that the *Sirius* system was able to make a 1.1% internal rate of return (IRR) with only the *Orion* satellite deployment missions as a source of revenue (assuming that only one satellite is manifested per launch in the replenishment phase of the contract). Aggressive development and test assumptions were used. For example, the prototype was assumed to be converted into the operational vehicle. By entering additional commercial launch markets, requiring the *Orion* corporation to pay 90% of the launch price for the initial constellation deployment in advance, and allowing for multiple manifestation of satellites in the replenishment phase, the *Sirius* program can achieve a remarkable IRR of 39.2%. Entry into these new markets at an optimum payload delivery price of $850/lb. will not only improve profitability, but will also serve to stimulate the growth of future space markets (such as space tourism, space-based manufacturing, and human exploration of the solar system).

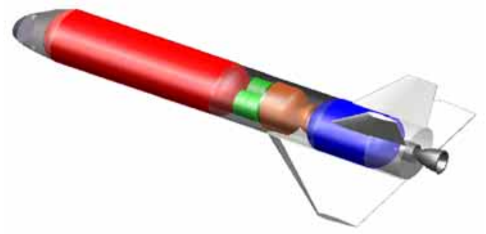


Figure 14 - Sirius Isometric View

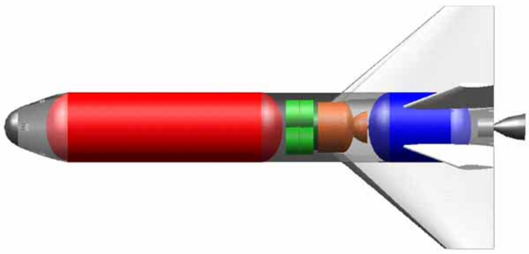


Figure 15 - Sirius Top View

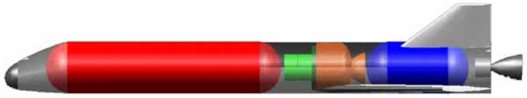


Figure 16 - Sirius Side View

Figure 17 - Sirius Rear View

CONCLUSIONS

Methodology Conclusions

1. RSE generation inherently incurs losses of information and accuracy. Careful attention must therefore be taken to analyze the fit and accuracy of the RSE. If the limitations of the RSE are understood, the RSE can be used to perform many calculations in a short time, to integrate these calculations onto a single platform, and to learn a great deal about the design space.

2. The most serious disadvantage of this process, in our opinion, is the feeling of alienation experienced by the disciplinarians since they are removed from the optimization process and replaced by a polynomial equation. The solution to this problem is to ensure that the disciplinarians understand the process and know how the information that they are generating is being used. It is extremely important, therefore, to involve the disciplinarians in evaluating the results of the optimization process.

3. The advantages of this process are the ability to analyze a large portion of the design space, the ability to directly handle discrete variables, the ability to perform disciplinary analyses in parallel, and the ability to design to cost. In this manner, point designs are generated in a very speedy and efficient manner. Trade studies are easily performed and assist the designers in making informed decisions.

Vehicle Conclusions

1. It is possible, using a methodology similar to one presented here, to design a new launch system specifically for deploying a Mega-LEO constellation. However, if such a system is used only for the constellation deployment, it is unlikely that the project would be funded due to little or no profit potential.

2. Once such as system is built, entry into current launch markets at launch prices substantially below those offered today will stimulate future launch markets and provide additional profits to offset the significant initial investment. The profitability of the launch system when these additional markets are considered is sufficient enough to attract investors.

3. The demands on a launch system from deploying large constellations requires a high degree of re-usability in the design. Use of existing hardware components can reduce development time and costs.

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Layout and Packaging (CAD) - Created and maintained solid model.

Performance - Optimized ascent and re-entry trajectories.

Hosung "Mike" Lee
Economics and Cost - Used cost estimating relationships and other business parameters to generate life cycle cost data and key economic variables.

Operations and Facilities - Assessed vehicle operational scenario including fleet size, turnaround time, and operating manpower requirements.

Kimberly Steadman
Aerodynamics - Provided vehicle aerodynamic model and generated aerodynamic data throughout the vehicle flight regime.

Aeroheating - Estimated vehicle peak surface temperature and evaluated thermal protection scheme.

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