CREATING SMALL SATELLITE CONSTELLATIONS AROUND THE SUN

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ABSTRACT

This paper deals with the propulsion and trajectory issues of creating a small satellite constellation around the Sun. The possible uses for such a constellation, including stereoscopic imaging of the Sun and gamma ray burst localization, are briefly discussed. The focus of the paper is then narrowed to a range of orbits and payload masses. Spread rate, the most important parameter for creating a sun orbiting constellation, is explained and defined. Next, it is shown that impulsive chemical propulsion is better than low thrust electric propulsion for this mission class. Finally, a full discussion of the propulsion and trajectory options available is presented. Among the topics discussed are: launch options, including dedicated launch vehicles and secondary payload slots; propulsion architecture options, including concentrating the propulsion into a single bus and distributing the propulsion among the satellites; and propulsion systems options including solid, monopropellant, and bipropellant propulsion. Performance calculations are given for each option over a wide range of parameters. It is concluded that the best dedicated launch vehicle options are bus monopropellant propulsion, and distributed solid propulsion. It is also concluded that secondary payload slots offer a very low cost alternative to dedicated launch vehicles for payload masses up to 55 kg.

INTRODUCTION AND PURPOSE

Sun orbiting satellite constellations are a class of interplanetary mission which has received little previous attention. However, there are several scientifically important missions that can use such a constellation. This section briefly discusses stereoscopic imaging of the Sun and gamma ray burst localization missions in order to justify and give perspective to the technical body of this paper.

The Sun is of course the most observed star in the cosmos. However, Earth and Earth orbit based sensors can only view one face of the sun at a time. The simplest solution to the problem of having only one vantage point on the Sun is to spread a number of satellites around the Sun in orbits similar to Earth's. These satellites need only carry a few basic sensors which complement more advanced sensors on Earth and in Earth orbit thus allowing the satellites to be small. By having multiple vantage points, events on all sides of the Sun can be observed simultaneously and regions of the sun can be observed stereoscopicaly allowing the three dimensional nature of events in the regions to be seen. Such a solar observation mission would lead to a better understanding of stellar structure and behavior.

Unlike the Sun, gamma ray bursts are one of the most enigmatic astronomical phenomena observed and will remain so until their positions can be better localized. Gamma ray bursts are high intensity flashes of gamma radiation lasting form 0.1 to 10 sec. Gamma ray bursts arrive about once a day from locations randomly distributed across the sky. This distribution suggests that gamma ray bursts come from outside the galaxy. Their distance along with their intensity may make gamma ray bursts among the most energetic events ever observed. Unfortunately, gamma ray burst detection systems have not reached the ~1 arcsecond accuracy needed to uniquely identify them with astronomical objects which are observable in other parts of the EM spectrum. Until the objects that produce gamma ray bursts can be observed by radio, infrared, optical, UV, and X-ray telescopes it is unlikely that a definitive theory for the causes of these events will be established. A simple method for achieving subarcsecond gamma ray burst localizations proposed by Ricker[1] is to spread a number of small satellites containing gamma ray burst detectors around the Sun. By measuring to millisecond accuracy the difference in time of arrival of each burst at each satellite, precise localizations can be achieved, allowing for follow up observations in other parts of the EM spectrum. Understanding gamma ray bursts will lead to a better understanding of the structure and evolution of the universe.

The motivation and funding for this work is linked to a gamma ray burst localization mission but the material presented here should be general enough for application to any Sun orbiting constellation. This paper is intended to give a feel for the propulsion and trajectory issues involved in creating a small satellite constellation around the Sun. In addition, given a desired number of satellites, a period in which the constellation is to develop, and a payload mass, one should be able to size a constellation to first order from the performance section of this paper.

SCOPE OF FINAL ORBIT AND PAYLOAD

To fit this material into one paper it is necessary to place limits on the final orbits and payload size considered.

The satellite orbits in this paper will be near the orbit of Earth with one apse at Earth's radius and the other apse ranging from 0.66 AU to 1.5 AU. This rules out creating constellations in circular orbits nearer to the Sun for better solar observation or farther from the Sun for a larger baseline. In addition, the constellations considered here will be continuously evolving. These restrictions not only simplify the analysis but, as will be shown later, vastly reduce the mission cost.

Most information in this paper is independent of spacecraft size. However, the focus of this paper is on small payloads. A typical solar observation or gamma ray burst localization payload mass budget is shown in Table 1. This mass budget contains a power system capable of producing 40 watts at 1 AU from the Sun and an X-band RF communications system capable of 10 bits per second at 2 AU from Earth. The attitude control system provides for fine pointing during communication with Earth, momentum wheel desaturation, and for one or more high thrust propulsive maneuvers.

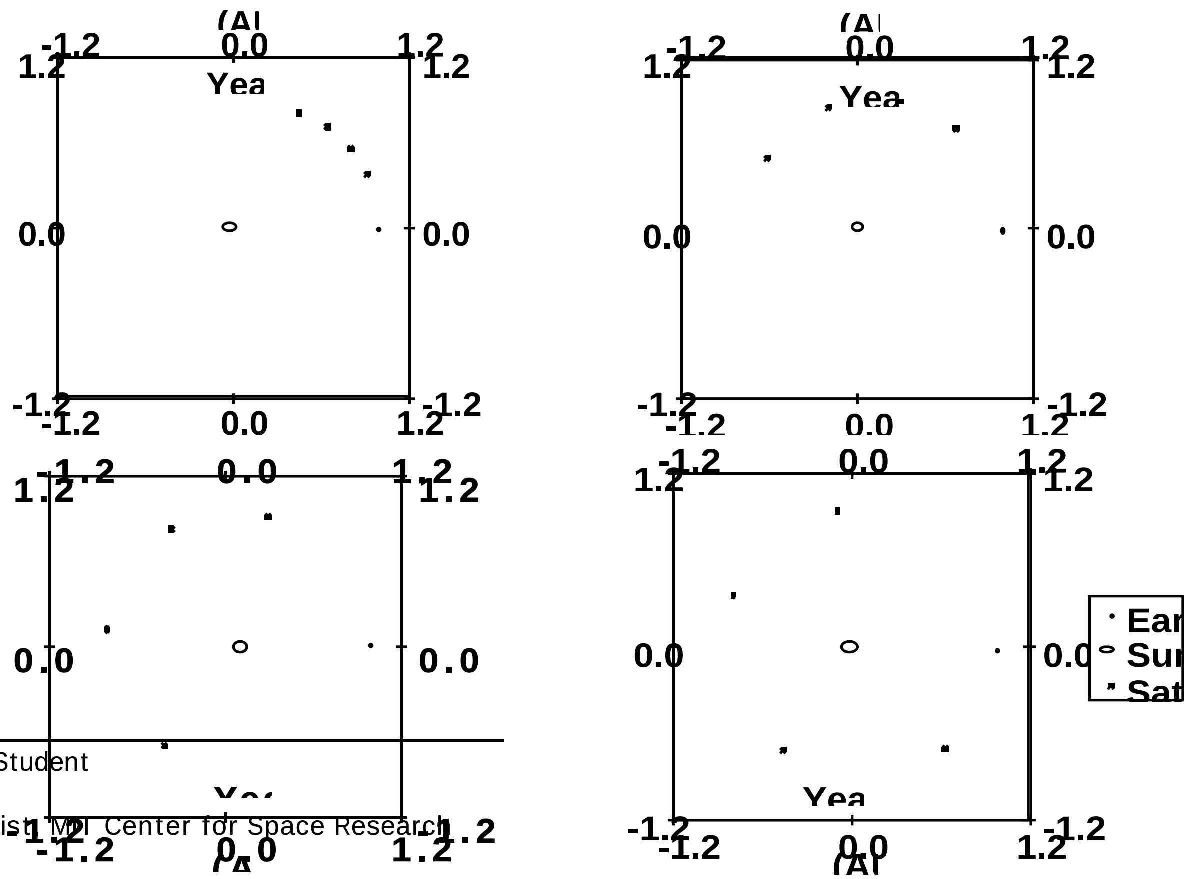

Fig. 1 The evolution of a satellite constellation with a maximum 75 deg/yr spread rate.

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Table 1 A typical small payload

Item	**Mass (kg)**
Structure / Thermal	12.0
Sensor(s)	10.0
Attitude Control	8.0
Power	7.0
RF Communications	3.5
Computer	1.2
Sub Total	41.7
20% Margin	8.3
Total	**50.0**

SPREAD RATE

Spread rate determines how quickly a constellation around the Sun develops and is one of the primary parameters in this paper. Spreading the satellites around the Sun with respect to Earth and each other, at an acceptable rate and minimum cost, is the interesting part of this mission class from a propulsion and trajectory stand point. A constellation is created by injecting each satellite in the constellation with a slightly different spread rate with respect to Earth and allowing them to spread around the Sun as shown in Figure 1. Therefore, it is important that the mechanism for achieving spread rate is thoroughly described and that spread rate is properly defined before discussing the details of this paper.

Spread rate is achieved by injecting the satellites into solar orbits with perihelions slightly lower than or aphelions slightly higher than 1 AU. With a perihelion lower than 1 AU, a satellite orbits the Sun faster than Earth and appears to move away from Earth in the direction of Earth's travel around the Sun. *Spreading ahead* of Earth will be defined as a *positive* spread rate. With a aphelion higher than 1 AU, a satellite will orbit the Sun slower than Earth and appear to move away from Earth in the opposite direction of Earth's travel around the Sun. Conversely, *spreading behind* Earth will be defined as a *negative* spread rate.

The most efficient way of changing the aphelion or perihelion of a satellite's orbit, and providing it a spread rate, is to exit the Earth frame with a V_{inf} parallel to the velocity vector of Earth. The exact change in period from that of Earth, and therefore the spread rate, can be computed from basic Keplerian orbital mechanics. The calculation of spread rate can be further simplified by linearizing the change in orbital energy around the Sun with respect to a small change in orbital velocity. Linearizing the resulting period equation yields[2]:

$$S \approx \frac{3V_{inf}}{\bar{a}} \qquad (1)$$

Where S is the spread rate, $\bar{a}$ is the semi-major axis of Earth, and V_{inf} is parallel to the velocity vector of Earth. If V_{inf} is not parallel to the velocity vector of Earth the result can be approximated by multiplying V_{inf} by the cosine of the angle between the two vectors. For hand calculations the constant $\bar{a}/3$ is equal to 27.5 when S is in deg/yr and V_{inf} is in m/s. Equation 1 closely approximates the full Keplerian solution, as shown in Figure 2, and only breaks down at exit velocities approaching solar escape (above spread rates of 300 deg/yr). Because of the accuracy and simplicity of equation 1 it is used as the definition of spread rate in this paper.

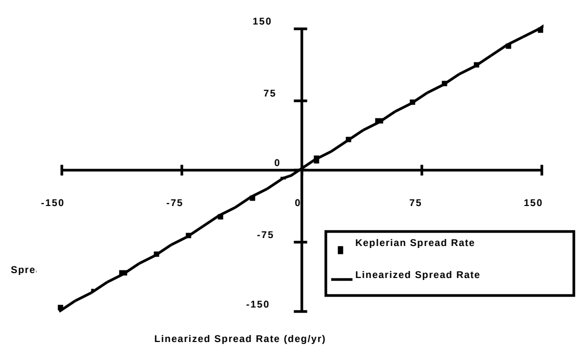

Fig. 2 Linearized vs. Keplerian spread rate.

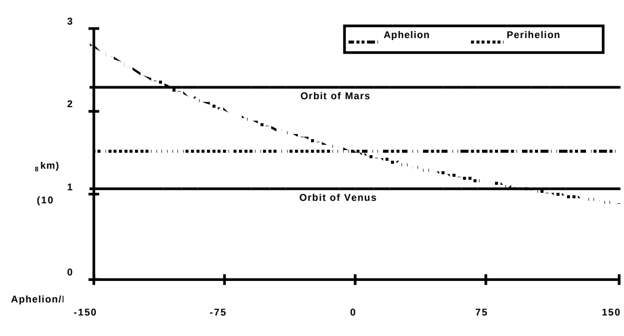

Fig. 3 Aphelion and perihelion vs. spread rate.

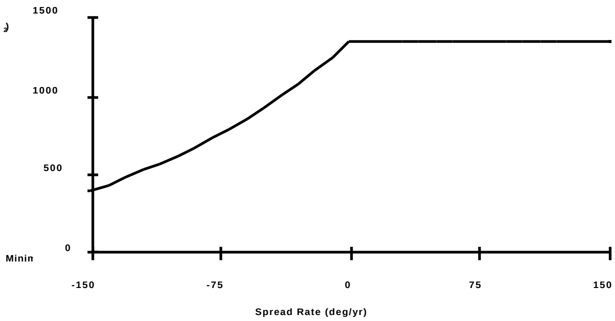

Fig. 4 Minimum solar flux vs. Spread rate.

The mechanism for creating spread rate can change the aphelion or perihelion of the satellite's final orbit considerably, as shown in Figure 3. For spread rates over ±100 deg/year a satellite will be going inside the orbit of Venus for positive spread rates and outside the orbit of Mars for negative spread rates. Satellites with high positive spread rates may suffer thermal problems while satellites with high negative spread rates require heavier power systems due to reduced solar flux, as shown in Figure 4. To a lesser extent negative spread rates also suffer from decreased data transmission rates. For the remainder of the paper positive spread rates are used when a choice exists.

IMPULSIVE VERSUS LOW THRUST

Two very different propulsion systems can be used to achieve a V_{inf} with respect to Earth and thus a spread rate: impulsive, low I_{sp}, chemical propulsion such as solids, liquid propellants, or low thrust, high I_{sp}, electrical propulsion such as arcjets, Hall thrusters, and ion engines. This section will explain why impulsive propulsion's ability to use the presence of Earth at injection gives it greater performance for the mission types considered.

Both impulsive and low thrust missions start when the satellites are injected at or above escape velocity. Next, each satellite must be provided with a different ΔV, corresponding to a different V_{inf} and spread rate, so that a constellation is formed. This ΔV is the primary propulsion for the mission. Without it the satellites might spread with respect to Earth, depending upon their injection velocity, but not with respect to each other thus forming a clump and not a constellation. The problem then is to use the least ΔV to create a given V_{inf}. The relation between ΔV and V_{inf} can be found from energy conservation:

$$\frac{1}{2}V_{inf}^2 = \frac{1}{2}V^2 - \frac{\mu}{r}, \text{ or}$$

$$V_{inf}^2 = V^2 - V_{esc}^2 \qquad (2)$$

Where V is the velocity of the satellite, μ is the gravitational parameter of Earth, r is the radius of the satellite from earth, and $V_{esc} = (2\mu/r)^{1/2}$ is the local escape velocity from Earth. Assuming that the satellite is launched at escape and is provided a ΔV to spread itself with respect to Earth and any other satellites launched with it equation 2 becomes:

$$V_{inf}^2 = \left(V_{esc} + \Delta V\right)^2 - V_{esc}^2 \qquad (3)$$

solving for ΔV:

$$\Delta V = \sqrt{V_{esc}^2 + V_{inf}^2} - V_{esc} \qquad (4)$$

ΔV can be minimized in equation 4 for a given V_{inf} by maximizing V_{esc}. V_{esc} is at its maximum just after injection when the satellites are closest to Earth and drops off rapidly as the satellites leave the Earth frame. When V_{esc} is zero ΔV is equal to V_{inf}. Impulsive propulsion can take advantage of the high V_{esc} near Earth by making its ΔV over a course of minutes or seconds immediately after injection. Low thrust propulsion must make its ΔV over weeks or months when the satellites are far from Earth and V_{esc} is nearly zero.

The reduction in required ΔV by applying thrust near Earth can be explained in terms of basic physics. The purpose of a ΔV is to change the energy of an orbit in the Earth frame in order to give it a V_{inf}. The rate at which the energy of a satellite's orbit is changed is the force acting on the satellite times the velocity of the satellite with respect to Earth. Therefore a given impulse (force times time) will create the greatest change in orbital energy when the impulse is applied at the highest velocity relative to Earth. An impulse applied to a satellite at V_{esc} near Earth where V_{esc} is high will change the energy of the satellite's orbit far more than if the impulse is applied when the satellite is far from Earth where V_{esc} is approaching zero.

To quantify the advantage of high thrust propulsion consider a spread rate of 90 deg/yr. This spread rate requires a V_{inf} of 2475 m/s from equation 1. If the ΔV is made at a 300 km altitude just after injection to the escape transfer orbit, the required ΔV is 277 m/s. If the ΔV is made far from Earth, the required ΔV is 2475 m/s. Impulsive propulsion requires nearly an order of magnitude less ΔV than low thrust propulsion. Figure 5 shows the required ΔV versus spread rate assuming the ΔV is made at a 300 km altitude and escape velocity.

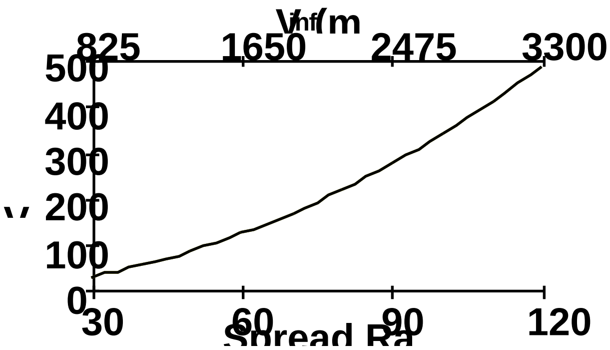

Fig. 5 ΔV vs. spread rate and V_{inf}.

To include the effect of the difference in I_{sp} between impulsive and low thrust propulsion it is assumed that to be equivalent, both types of propulsion must have the same wet mass to dry mass ratio. This assumption makes electric propulsion appear better than it is since the solar arrays and power converters needed for electric propulsion will make it heavier at the same mass ratio. While the assumption of similar mass ratios is poor it saves delving into the specifics of each technology. The ratio of I_{sp}'s to yield the same mass ratio is simply the ratio of the ΔVs:

$$\frac{I_{spelectric}}{I_{spchemical}} = \frac{V_{inf}}{\sqrt{V_{esc}^2 + V_{inf}^2} - V_{esc}} \quad (5)$$

Figure 6 shows the low thrust I_{sp} needed to produce the same mass ratio as several different types of chemical propulsion versus spread rate. Impulsive propulsion has a clear advantage over low thrust propulsion for the spread rates considered. Even at the same mass ratios and weights chemical propulsion is preferable because it is simpler and more flight proven.

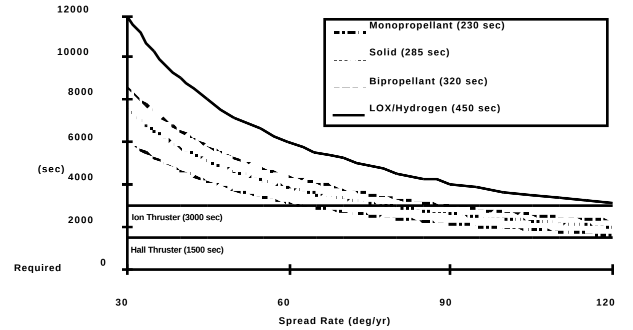

Fig. 5 Low thrust I_{sp} needed to match impulsive Isp.

Finally, this analysis shows why missions that circularize themselves into higher or lower orbits around the Sun are much more expensive than missions that do not circularize. To circularize at a solar distance near that of Earth's requires removing the extra ~V_{inf}, without the benefit of Earth's presence. This means that the circularization burn requires an order of magnitude more ΔV than the spreading ΔV. Missions with circular final orbits are therefore much more expensive than the ones considered in this paper.

MISSION OPTIONS

The remaining mission options can be divided into those that are launched on dedicated launch vehicles and those that are launched as secondary payloads. Several propulsion types will discussed for each case.

DEDICATED LAUNCH VEHICLE MISSIONS

A dedicated launch vehicle is the simplest way of placing a payload onto an escape trajectory. Many launch vehicles and their variants are available in the class needed to launch a constellation of Sun orbiting satellites, as is shown in Table 2. Since a large number of similar competitive launchers exist it is reasonable to assume that launch cost will be directly proportional to throw mass. Therefore all dedicated launch vehicle propulsion systems should be optimized for minimum throw mass.

Table 2 Launch vehicles capabilities to $C_3 = 0$[3,4,5]

Launch Vehicle	Throw Mass (kg)
Pegasus XL / Star 27	127
Taurus / Star 37	315
Taurus / Orion 38 / Star 37	340
Taurus XL / Star 37	360
Taurus XL / Orion 38 / Star 37	380
Taurus XLS / Star 37	400
Taurus XLS / Orion 38 / Star 37	425
LMLV 2	425
Taurus XL / AUS-51	430
Taurus XL / Orion 38 / AUS-51	445
Taurus XLS / AUS-51	515
Taurus XLS / Orion 38 / AUS-51	530
LMLV 3 (2 strap on)	590
LMLV 3 (6 strap on)	700

Position of Perigee

The ability of a dedicated launch vehicle to choose the launch time of day, and thus the position of perigee, is one of its primary advantages. The optimum position of perigee for a single satellite results in a V_{inf} parallel to the velocity vector of Earth. For convenience, position of perigee will be defined as the clockwise angle from the velocity vector of Earth. From Keplerian orbital mechanics the optimum position of perigee for a single satellite is:

$$\omega = \sin^{-1}\left(\left(1 + 4\frac{V_{inf}^2}{V_{esc}^2}\left(1 + \frac{V_{inf}^2}{V_{esc}^2}\right)\right)^{-\frac{1}{2}}\right) \mp \frac{\pi}{2} \qquad (6)$$

Where ω is the position of perigee. The second term in equation 6 is negative for positive spread rates and positive for negative spread rates. Unfortunately all of the satellites in a constellation will be given the same position of perigee if launched on the same vehicle. The optimum position of perigee for a constellation minimizes the average ΔV of all satellites. It is necessary to find the V_{inf}, and from that the ΔV, for each satellite to yield a component of V_{inf} parallel to Earth's velocity vector corresponding to the desired spread rate. The V_{inf} needed is found by solving:

$$V_{inf\ component} =$$

$$V_{inf}\cos\left(\omega - \sin^{-1}\left(\left(1 + 4\frac{V_{inf}^2}{V_{esc}^2}\left(1 + \frac{V_{inf}^2}{V_{esc}^2}\right)\right)^{-\frac{1}{2}}\right) \pm \frac{\pi}{2}\right) \qquad (7)$$

Where the second term in the cosine is positive for positive spread rates and negative for negative spread rates. Using equation 7 to optimize position of perigee for the minimum average ΔV for all satellites in constellations of different sizes and spread rates yields Figure 7. It is assumed in Figure 7 and for the remainder of the paper that the spread rates of each satellite in a constellation are evenly incremented up to that of the maximum spread rate satellite. For example, in Figure 1 each satellite has a spread rate of 18.75, 37.50, 56.25 and 75.00 deg/yr respectively. The positions of perigee versus spread rate for constellations of 4 to 8 satellites can be fit by the line $-0.2141\omega - 0.8740$ without significant error. Errors in position of perigee of three to four degrees result in required ΔV errors of less than a 0.1%. For negative spread rates the position of perigee is the one shown Figure 7 plus 180 degrees.

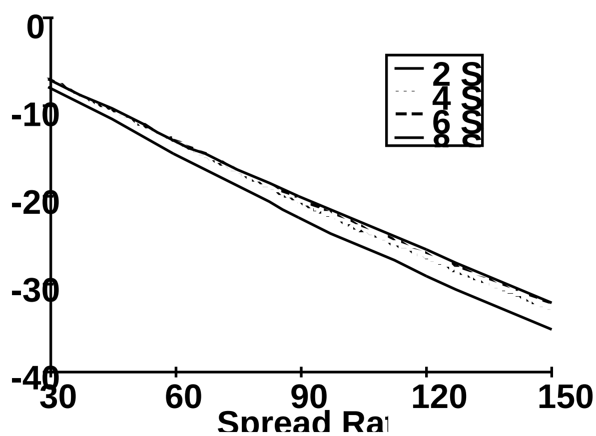

Fig. 7 Position of perigee vs. positive spread rate.

Bus Propulsion

In a bus propulsion system the spreading ΔV for each satellite is provided by a single propulsion bus. A bus / satellite stack is injected with the V_{inf} of the maximum spread rate satellite. After the maximum spread rate satellite is released, the bus decelerates the stack to the V_{inf} required by the next highest spread rate satellite and releases that satellite. The process of decelerating and releasing satellites continues until all of the satellites are released. The satellites are decelerated from the maximum spread rate so that the largest ΔVs are made first when the stack is closest to the Earth. A wait time is required between each ΔV for releasing a satellite and performing a collision avoidance maneuver.

The advantage of a bus propulsion system is that it requires purchasing only one propulsion system and is therefore lighter and lower cost than a distributed propulsion system which requires purchasing multiple propulsion systems. The disadvantage of a bus propulsion system is that it must be more reliable than a distributed propulsion system. If a bus propulsion system fails the entire mission may be lost.

For performance calculations the bus is considered an augmented satellite. Because the bus is placed into a solar orbit with the satellites it makes sense to include a set of sensors on the bus and make continued use of its attitude control, communications and power system after its initial task is done. The mass budget for the bus will be the mass of the propulsion system plus the mass budget in Table 1. Modeling is done numerically by integrating the equations of motion in the Earth frame in order to capture the finite burn duration losses. The wait time between burns is assumed to be 10 minutes.

Bus Monopropellant

The monopropellant system for the primary payload bus monopropellant propulsion option is modeled in this section and all later sections with the following assumptions. The monopropellant system is assumed to have a nominal I_{sp} of 230 sec at an inlet pressure of 200 psi. Thrust is assumed to be proportional to tank pressure. The tank pressure is modeled with a 4 to 1 adiabatic isentropic expansion starting at 400 psi and 300 K. The adiabatic isentropic expansion is justified because all of the propellant is used over a short period of time, preventing significant heat transfer. The effect of inlet pressure on I_{sp} is given by[6]:

$$I_{sp} = I_{sp\,nominal}\left(1 - 0.005\frac{P_{nominal}}{P}\right) \qquad (8)$$

Where P is inlet pressure. In addition to the nominal propellant, 10% more propellant is included as trapped propellant and flight performance reserve. The mass of the propellant tank is assumed to be 10% of the mass of the propellant. The mass of the thruster is modeled by[6]:

$$M_{thruster} = 0.068745\ F^{0.55235} \qquad (9)$$

Where mass is in kg and force is in N. Equation 9 is a fit to existing monopropellant thrusters. Finally, the mass of the latch valves, filters, pressure transducers, lines, line heaters, and fittings is assumed to be 5 kg.

The results of modeling the primary payload bus monopropellant propulsion system are shown in Figures 8 and 9. Throw mass versus initial acceleration based on nominal thrust is shown in Figure 8 for a 90 deg/yr spread rate. The optimum is caused by the superposition of two competing processes: decreasing thrust leads to higher finite burn duration losses, and increasing thrust leads to increased thruster mass. This optimum, however, is relatively wide. Using the optimum acceleration of 0.05g the throw mass versus spread rate is shown in Figure 9.

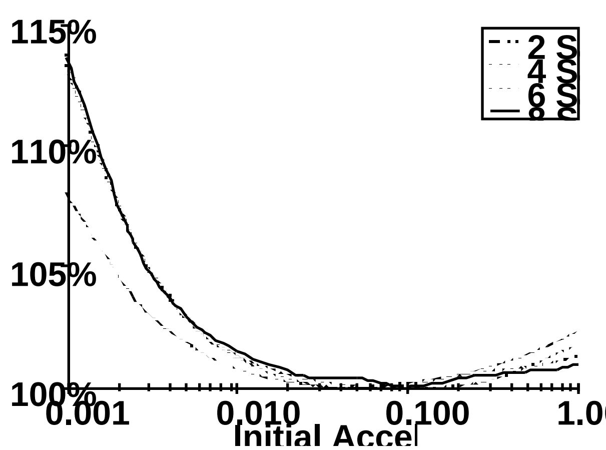

Fig. 8 Throw mass vs. initial acceleration for bus monopropellant at a 90 deg/yr spread rate.

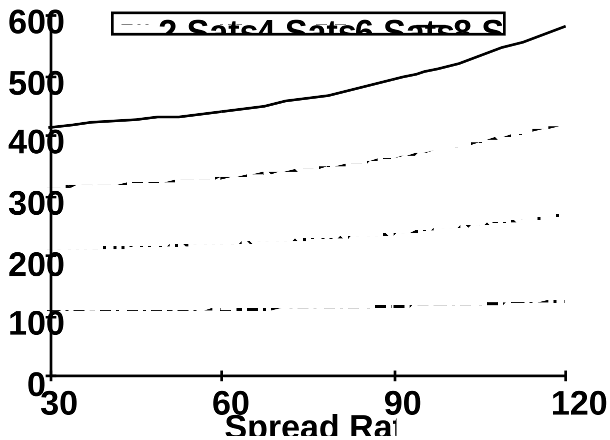

Fig. 9 Throw mass vs. spread rate for bus monopropellant.

Bus Bipropellant

The bipropellant propulsion system for the primary payload bus bipropellant propulsion option is modeled in this section and all later sections with the following assumptions. The bipropellant thruster is assumed to have a nominal I_{sp} of 290 sec at an inlet pressure of 220 psi. Higher performance dual mode engines are being developed but are not generally available in the thrust range required and are expensive to qualify so are not considered. A pressure regulated nitrogen pressurant system is assumed. In addition to the nominal propellant, 10% more propellant is included as outage and flight performance reserve. The mass of the propellant tank is assumed to be 10% of the mass of the propellant. The mass of the thruster is modeled by:

$$M_{thruster} = 0.18989\ F^{0.48667} \quad (10)$$

Where mass is in kg and force is in N. Equation 10 is a fit to the masses of the 4 to 800 N bipropellant thrusters manufactured by Kaiser Marquardt[7]. Finally, the mass of the latch valves, check valves, filters, pressure transducers, pressure regulators, lines, line heaters, and fittings is assumed to be 10 kg.

The results of modeling the primary payload bus bipropellant propulsion system are shown in Figures 10 and 11. Throw mass versus initial acceleration is shown in Figure 10 for a 90 deg/yr spread rate. Using the optimum initial acceleration of 0.02g, throw mass versus spread rate is shown in Figure 11.

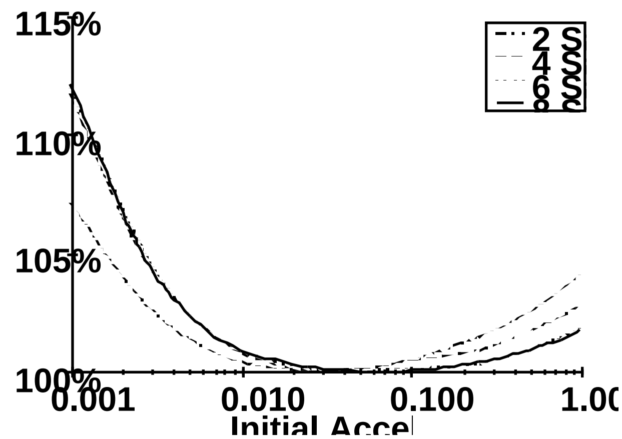

Fig. 10 Throw mass vs. initial acceleration for bus bipropellant at a 90 deg/yr spread rate.

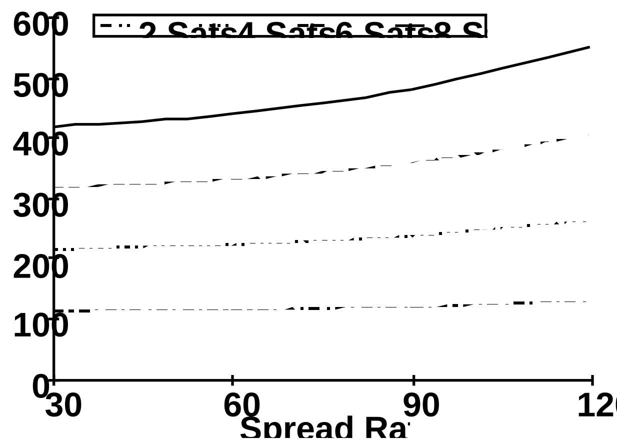

Fig. 11 Throw mass vs. spread rate for bus bipropellant.

Distributed Propulsion

In a distributed propulsion system the spreading ΔV for each satellite is provided by the satellite's own propulsion system. The satellite stack is injected with the V_{inf} of the lowest spread rate satellite. Starting with the maximum spread rate satellite each satellite in turn separates from the stack and makes its spreading ΔV. As with the bus propulsion system the satellite release order is arranged so that the largest ΔVs are made closest to Earth. Again, a wait time is required between releasing satellites but it will be shorter than for the bus propulsion option because no collision avoidance maneuver is performed.

The advantage of a distributed propulsion system is that the failure of one satellite propulsion system does not prevent the deployment of the other satellites. In addition, burns are made in parallel reducing finite burn loss. The disadvantage of a distributed propulsion system is that it costs more than a bus propulsion system because more propulsion systems must be purchased. The throw mass will also be higher for a distributed propulsion system.

The mass of the satellites is assumed to be the mass of the propulsion system plus the mass budget in Table 1. Monopropellant and solid propulsion are considered in the following sections. However, bipropellant propulsion is neglected because it will yield no performance advantage at this size while incurring significant cost and complexity. Modeling is done numerically by integrating the equations of motion in the Earth frame in order to capture the finite burn duration losses. The wait time between releases is assumed to be 5 minutes.

Distributed Monopropellant

All of the assumptions made for the monopropellant propulsion system in the bus monopropellant section are retained in this section.

The results of modeling the primary payload distributed monopropellant propulsion system are shown in Figures 12 and 13. Throw mass versus initial acceleration based on nominal thrust is shown in Figure 12 for a 90 deg/yr spread rate. Using the optimum initial acceleration of 0.05g, the throw mass versus spread rate is shown in Figure 13.

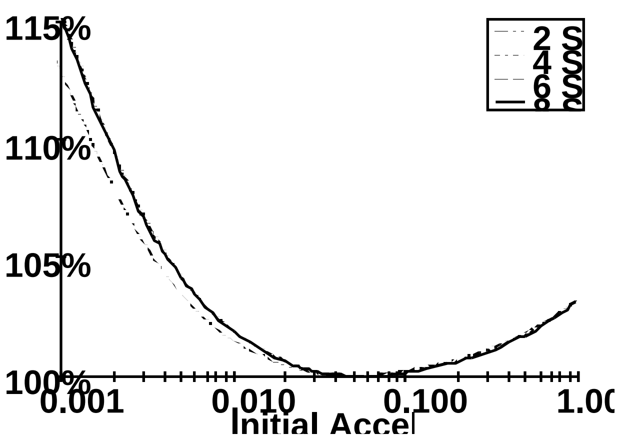

Fig. 12 Throw mass vs. initial acceleration for distributed monopropellant at a 90 deg/yr spread rate.

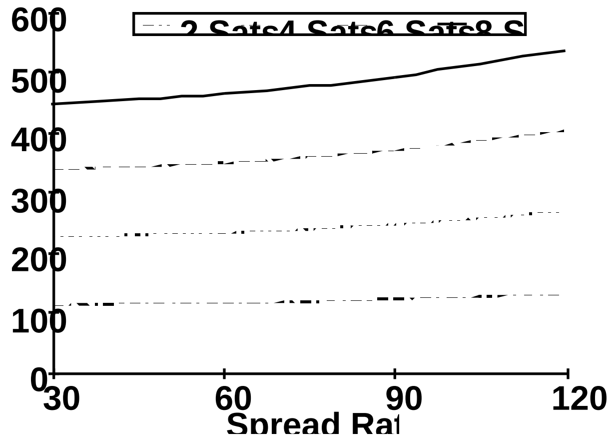

Fig. 13 Throw mass vs. spread rate for distributed monopropellant.

Distributed Solid

A simple way of providing distributed propulsion is through small solid motors such as the Star 5. For performance calculations a generic solid is assumed with a mass fraction of 50% and an effective Isp of 265 sec. Since the action time of a solid is very rapid the performance can be modeled as Keplerian. The solid motors are sized for the ΔV of the largest spread rate satellite. For lower spread rates the satellites are allowed to coast to a higher altitude where their spreading ΔV is the same as that of the maximum spread rate satellite. Coasting to a higher altitude saves the complexity of off loading the solid motors. Even if the motors were off loaded this maneuver would still be necessary because most solid motors can only be off loaded 20%. The throw mass versus spread rate is shown in Figure 14.

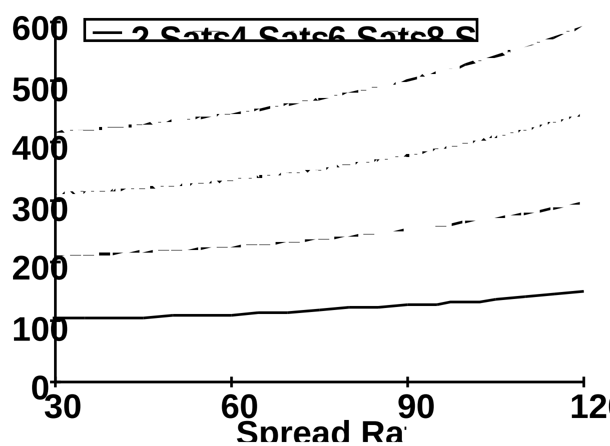

Fig. 14 Distributed solid throw mass vs. number

of satellites and spread rate.

Comparison of Dedicated Launch Vehicle Options

Figure 15 compares bus monopropellant and bipropellant propulsion by superimposing Figures 9 and 11. Over most of the range shown monopropellant and bipropellant propulsion are nearly equivalent. Bipropellant propulsion has an increasing advantage at high spread rates and large constellation sizes because of its high I_{sp}. Monopropellant has an advantage at lower spread rates and smaller constellation sizes because its fewer components result in a lower dry mass. Monopropellant propulsion is always preferable to bipropellant when the two are nearly equivalent because of monopropellant propulsion's low cost (1/3 to 1/4 of bipropellant propulsion) and simplicity.

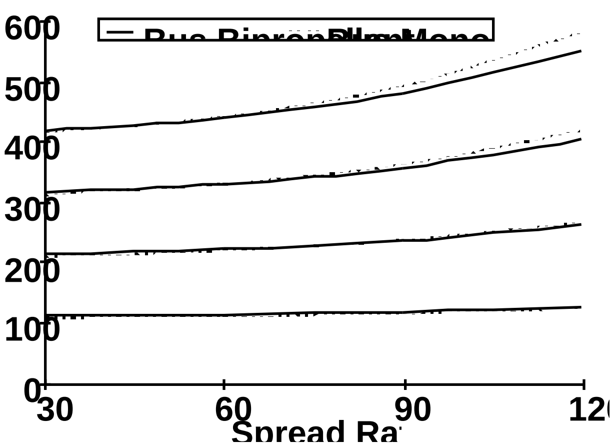

Fig. 15 Comparison of bus bipropellant and monopropellant.

Figure 16 compares distributed monopropellant and solid propulsion by superimposing Figures 13 and 14. At lower spread rates solid propulsion has an advantage due to its high I_{sp} and low mass fraction. This advantage is lost at high spread rates because it is assumed that the solid motors are not off loaded for the lower spread rate satellites in a constellation. Even if off loading is assumed it has a minor effect on the throw mass since most solids can only be off loaded 20%. When monopropellant and solid propulsion are nearly equivalent solid is preferable because of its low cost and simplicity.

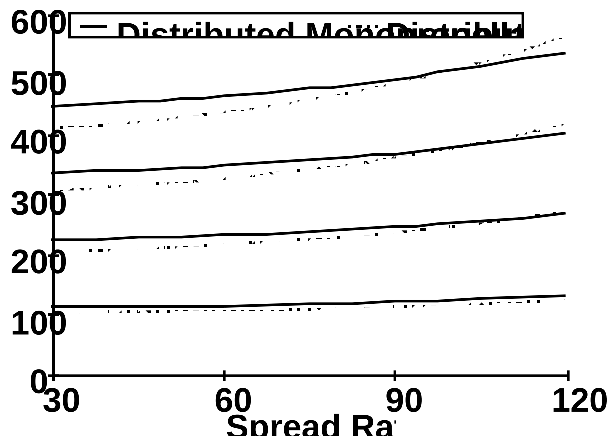

Fig. 16 Comparison of distributed monopropellant and solid.

Finally, Figure 17 compares the best bus option, bus monopropellant, and the best distributed option, distributed solid, by superimposing Figures 9 and 14. Over the spread rates shown both are nearly equivalent so a decision between the two must be made on different criteria. Bus monopropellant offers lower cost at higher complexity and risk of a single point failure. Distributed solid offers simplicity and lower risk at higher cost. The results of this trade will be dependent on the priorities of the mission being considered.

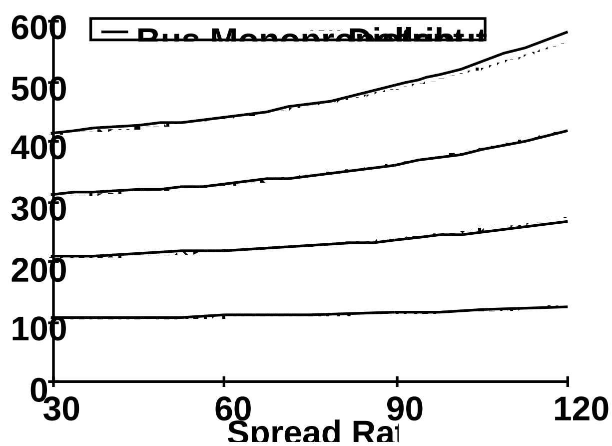

Fig. 17 Comparison of bus monopropellant and distributed solid.

SECONDARY PAYLOAD MISSIONS

Frequently, commercial missions are injected into geosynchronus transfer orbit (GTO) with a small amount of excess capacity. This excess capacity can provide a very economical way of launching a Sun orbiting constellation. Unfortunately, this capacity is rarely used because no standard procedure exists for purchasing and utilizing it, with the exception of the Ariane Structure for Auxiliary Payloads (ASAP). The Ariane 4 ASAP ring can launch several 50 kg payloads simultaneously while the Ariane 5 ASAP ring is even more capable. The Ariane 5 ASAP ring is designed to launch up to eight 80 kg, 65×65×80 cm payloads for a fraction of the cost of a dedicated launch vehicle. Thus the Ariane 5 ASAP ring can loft an entire constellation at a great cost savings. The Ariane 5 ASAP capabilities will be assumed for the remainder of this paper.

While ASAP payload slots are inexpensive, there are two disadvantages to launching a constellation as a secondary payload from GTO. The most obvious is the cost of getting from GTO to escape: about 776 m/s (assuming a 300 km × 35,786 km transfer orbit) in addition to the spreading ΔV. Under ideal conditions a 90 deg/yr spread rate would require 1053 m/s.

The second disadvantage arises from the nature of a secondary payload. Secondary payloads do not choose the launch window; the primary payload does. Depending on the time of launch the position of perigee of the transfer orbit could be very different from the optimum position of perigee for the desired spread rate. An off optimum position of perigee will result in a V_{inf} that is not parallel to the velocity vector of Earth. Correcting for a nonparallel V_{inf} requires a larger V_{inf} or making the ΔV at a true anomaly off perigee, both of which require a larger than ideal ΔV. Optimizing ΔV by varying true anomaly for a 90 deg/yr spread rate for all positions of perigee results in Figure 18. The optimum true anomaly versus position is shown in Figure 19. If a specific direction of spread is required the ΔV can cost up to 3.40 km/s. However, if both positive and negative spread rates are acceptable the cost can be reduced to a maximum of 2.23 km/s. The minimum cost versus position of perigee for bi-directional spreading is shown in Figure 20.

Fortunately, Ariane 5 GTO launches are restricted to a 45 minute launch window per day[8]. The start of this window varies slightly with time of year. The range of launch times during a year translates to positions of perigee from approximately 90 to 120 deg, as show in Figures 18, 19 and 20. This range of positions of perigee is nearly ideal for negative spread rates. For -90 deg/yr the maximum cost in this range is 1.35 km/sec or only 300 m/s greater than ideal. With holding orbits an ideal ΔV can be achieved, as discussed in the next section.

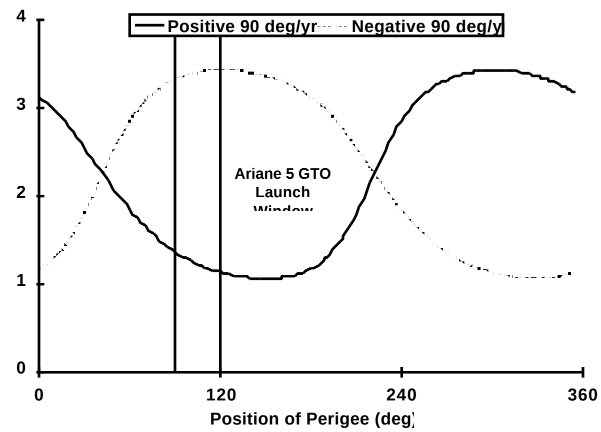

Fig. 18 Cost vs. position of perigee for a ±90 deg/yr spread rate.

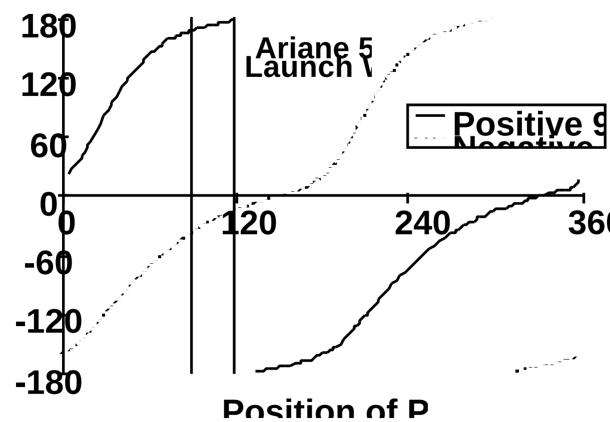

Fig. 19 True anomaly vs. position of perigee for a ±90 deg/yr spread rate.

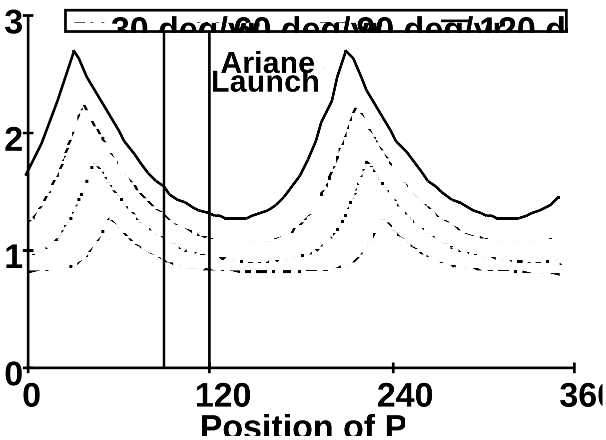

Fig. 20 Cost vs. position of perigee for several spread rates.

Holding Orbits

The purpose of a holding orbit is to move a suboptimum position of perigee to an optimum position of perigee using a combination of the motion of Earth around the Sun, as shown in Figure 21, and line of apsides rotation due to Earth oblateness. After deploying the primary payload, the ASAP ring releases each of the secondary payload satellites. The satellites then wait in a holding orbit until the position of perigee moves to the optimum point and make their escape ΔV.

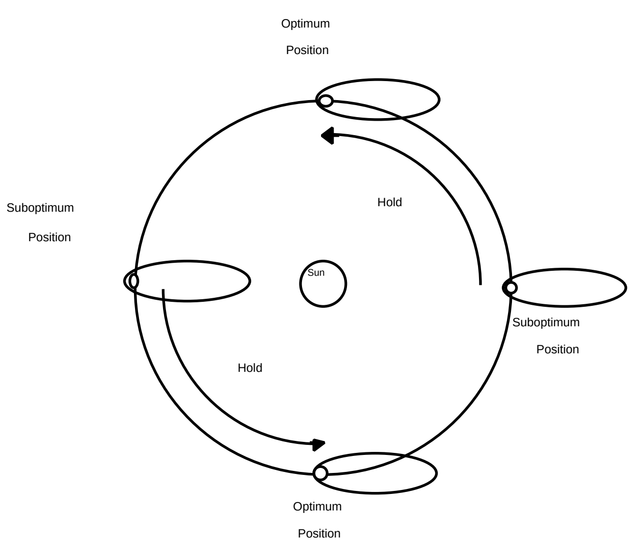

Fig. 21 Holding orbit concept.

The simplest holding orbit is GTO. The rate of change in position of perigee in GTO is relatively slow (~0.66 deg/day) because the oblateness of the Earth causes the holding orbit to precess. However, a slow rate of change in the position of perigee is preferable to the complexity of moving to a higher orbit in order to achieve a higher rate of change. For example, the cost versus hold time for a 90 deg/yr spread rate is shown in Figure 22. A wait of no more than 65 days yields the ideal ΔV for a -90 deg/yr spread rate.

The primary disadvantage of holding in GTO is radiation exposure. A satellite in GTO will spend a significant fraction of its time in the Van Allen radiation belts. It is assumed in this paper that performance is the primary mission driver and that the radiation exposure from holding in GTO is acceptable. Therefore, the performance calculation made in the following sections assume that holding in GTO will be used to achieve the ideal ΔV.

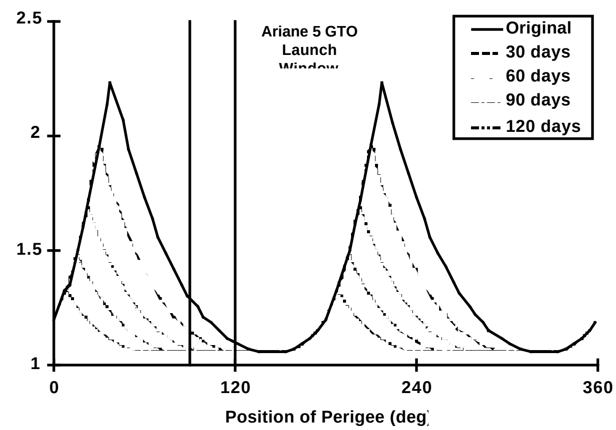

Fig. 22 Cost vs. hold time and starting position of perigee for a ±90 deg/yr spread rate.

If the radiation exposure or hold time is not acceptable a mission can be designed for immediate launch from GTO. Assuming the worst case, the cost of immediate launch in will be an increase in no greater than 30% in ΔV. At a 90 deg/yr spread rate this will cost ~6 kg of payload†. Another option is to design for the average ΔV in the launch window and accept a slightly higher or lower spread rate depending upon the actual position of perigee. The average ΔV will cost less than ~3 kg at a 90 deg/yr spread rate. Finally, if some hold time is acceptable, the greatest decreases in ΔV derive from the earliest part of the holding time.

† For a 60 deg/yr spread rate the cost is 4.5 kg.

Bipropellant and solid propulsion systems are considered in the following sections for producing the required ΔV. Monopropellant is neglected because it is not capable of significant payload at these ΔVs.

Liquid Propulsion

The secondary payload bipropellant propulsion system is modeled in the same way as the previous bipropellant propulsion system was modeled. The only exception is that the mass of valves and lines is assumed to be 5 kg due to the small size of this system. The mass of each satellite is assumed to be the mass of the propulsion system plus the mass budget in Table 1. Modeling is done numerically by integrating the equations of motion in the Earth frame in order to capture the finite burn duration losses.

The results of modeling the secondary payload bipropellant propulsion systems are shown in Figures 23 and 24. Because the total mass is limited to 80 kg, payload mass is used as a performance metric. The payload will be sized by the highest spread rate satellite so only it needs to be modeled. Payload mass versus initial acceleration is shown in Figure 23 for a 90 deg/yr spread rate. Using the optimum initial acceleration of 0.1g the payload mass versus spread rate is shown in Figure 24. These results are conservative because of the high outage and flight performance reserve assumed earlier.

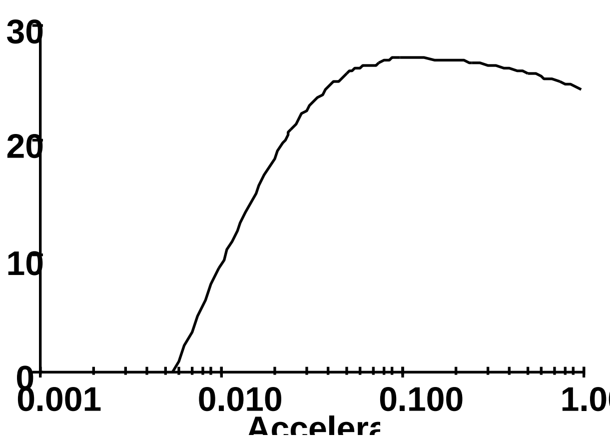

Fig. 23 Payload mass vs. initial acceleration for secondary bipropellant at a 90 deg/yr spread rate.

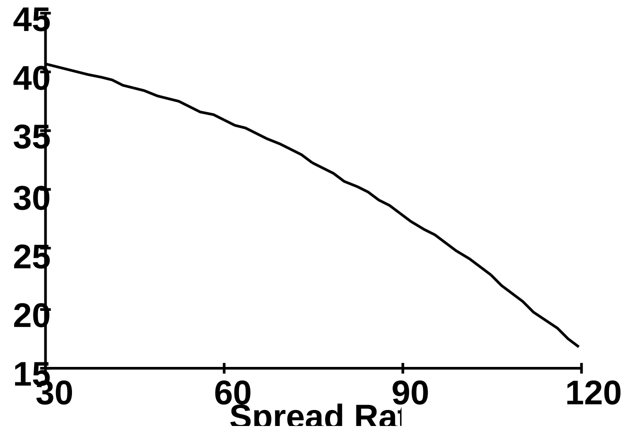

Fig. 24 Payload mass vs. spread rate for secondary bipropellant.

Secondary Payload Solid Propulsion

With its inherent simplicity and low cost, solid propulsion is an attractive option. The analysis of the secondary payload solid propulsion system is easier than that of the secondary payload liquid propulsion system because the high thrust and short action time of solids make finite burn losses negligible. Assuming a generic solid with a mass fraction of 0.9 and an I_{sp} of 285 sec yields the payload mass versus spread rate shown in Figure 25. Figure 25 also shows the capability of the Star 13A using up to 20% off loading to achieve different spread rates. The Star 13 motors manufactured by Morton Thiokol are low cost and ideal for this mission class. The real payload may be somewhat smaller than shown in Figure 25 due to additional structure resulting from the high burn out accelerations (≥10 g) of small solid motors.

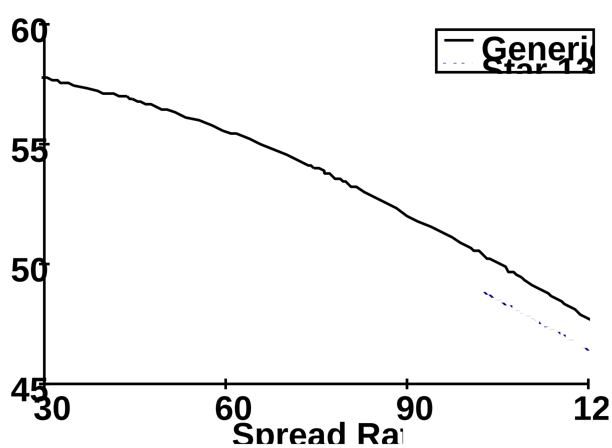

Fig. 25 Payload mass vs. minimum spread rate for secondary payload solid propulsion.

Comparison of Secondary Payload Options

Figure 26 compares secondary payload bipropellant and solid propulsion by superimposing Figures 24 and 25. Solid propulsion has an advantage due to its high mass fraction and short action time which minimizes finite burn duration losses. Solids are also simpler and more economical than bipropellants. The only disadvantages of solid propulsion are its lack of flexibility and high accelerations.

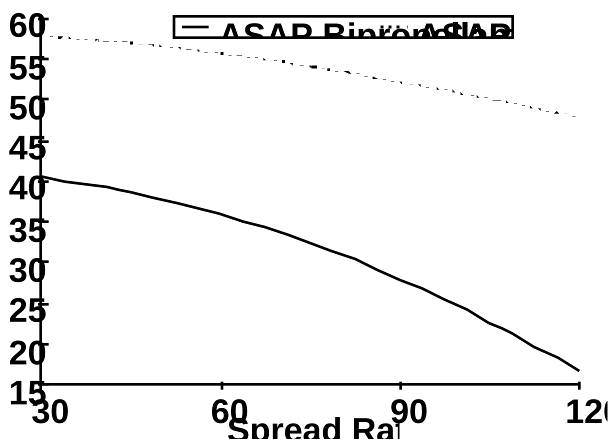

Fig. 26 Comparison of secondary payload bipropellant and solid.

Finally, it is necessary to compare dedicated launch vehicles and secondary payload slots. While secondary payloads are limited to ~55kg payloads, it offers a significant advantage over dedicated launch vehicles. As with most space applications, launch is the largest component of mission cost for Sun orbiting constellations. ASAP slots cost only a fraction of an equivalent dedicated launch vehicle for this application. Therefore, secondary payload slots offer a very low cost alternative to dedicated launch vehicles for payload masses up to 55 kg.

CONCLUSIONS

This paper has shown that for Sun orbiting constellations high spread rates, and circular final orbits are unacceptable in terms of vehicle systems and propulsion. It has been shown that impulsive chemical propulsion will yield higher performance than low thrust electrical propulsion. Through extensive modeling of mission scenarios, it has been shown that bus monopropellant propulsion is superior to bus bipropellant propulsion, that distributed solid propulsion is superior to distributed monopropellant propulsion, and that secondary payload solid propulsion is superior to secondary payload bipropellant propulsion. Finally, it is concluded that secondary payload slots offer a very low cost alternative to dedicated launch vehicles for payload masses up to 55 kg.

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1. Ricker, G.R., “The Energetic Transient Array (ETA): A Network of “Space Buoys” in Solar Orbit for Observations of Gamma-Ray Bursts”, *High Energy Astrophysics in the 21st Century*, American Institute of Physics, 1990, pp. 375-387

2. Patel, B.T. et al, “Electric Propulsion for an Interplanetary Astrophysics Mission”, 24th International Electric Propulsion Conference, IEPC-95-239, September 1995

3. Orbital Sciences Corporation, *Commercial Pegasus Payload User’s Guide*, Orbital Sciences Corporation, Dulles, VA, 1993, Section 3.0

4. Orbital Sciences Corporation, *Commercial Taurus Launch Vehicle Payload User’s Guide*, Orbital Sciences Corporation, Dulles, VA, 1996, Section 3.0

5. Lockheed Missiles & Space Company, Inc., *Lockheed Launch Vehicle User’s Guide*, Lockheed Missiles & Space Company, Inc., Sunnyvale, CA, 1995, Section 2.5

6. Brown, Charles D., *Spacecraft Propulsion*, American Institute of Aeronautics and

Astronautics, Washington, DC, 1996, Chapter 4

7. Kaiser Marquardt, *Rocket Engines And Propulsion Systems*, Kaiser Marquardt, Van Nuys, CA, 1990

8. Arianespace, *Ariane 5 User's Manual*, Arianespace, Every Cedex, France, 1991, Section 8.5