A Feasibility Study of Micro-Satellites for Earth Observation

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There is a continuing desire to minimize the fabrication, launch, and operational costs of Earth-observing satellites, while still maintaining their high-resolution capabilities. Micro-satellites have been suggested as a method for obtaining these results using several different configurations; such as conventional filled aperture optics, distributed aperture systems, constellations and tethers. The ability of the different spacecraft types to achieve images of a specified resolution and quality are examined, as are their affects on the mass and size of the spacecraft. However, first a minimum spacecraft size is discussed and formulated for missions of this and similar types. This is used as a first order analysis to determine when micro-satellites may be applicable to a specific mission. These results, and those from the analysis of the different spacecraft types are then used to determine when, and if, it is beneficial to use a micro-satellite over a more conventional spacecraft design. It will be demonstrated that distributed aperture systems and deployable primary mirrors are generally the best approaches for high-resolution-imaging micro-satellites, but that distributed aperture systems are useful when replacing very large primary mirrors.

Introduction

There is an increasing interest to replace large spacecraft with small or groups of very small satellites to satisfy mission objectives, while being both economical and robust.

These small craft are sometimes called "micro" or "pico-satellites," and would operate as some type of harmonious system called a "constellation."[6] Thus there is a perceived benefit in total mass and an improvement in the fault tolerance of the system, if the functionality is distributed amongst many small craft rather than just one. Clearly distributing redundant capability between numerous craft would improve system robustness.

In addition some systems are naturally distributed because of operational issues. For instance to improve ground track revisit times or geometric constraints (e.g. GPS constellation) there may be an independent need to have multiple craft. The 'spirit' of "micro-satellites" is not simply to add redundancy by launching numerous craft using existing technology, nor simply scaling down an already distributed system, but to replace mass hungry or expensive subsystems with single or distributed subsystems, that are less expensive and offer either equivalent or superior performance. This study is a step toward bounding some of the system parameters to identify where there is some benefit to replacing a large monolithic craft.

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This effort began as a classroom project in a graduate level systems design course that was posed as a response to a request for a proposal for the development and design of a low-cost, high-resolution Earth imaging spacecraft, from the Air Force Research Laboratory (AFRL).[8]

Consequently, rather than discuss the appropriateness of small satellites in general, the bias here is particular to Earth reconnaissance. These requirements are to image the Earth in the visible and infrared, at resolutions in the range of 0.3 to 3 meters, and to keep the total system mass below 500kg. "Micro-satellites," as defined by the AFRL, are spacecraft that fall between 10 and 100 kilograms in mass, and are less then or equal to 0.5 meters in length per dimension.[6] The general requirement to observe the Earth implies that there is a need to collect substantial information on orbit and then transmit it to a receiver. Then at a minimum, the light collection aperture, power requirements and antenna size, all have lower bounds that in turn impact the spacecraft geometric size and mass. A wide variety of spacecraft configurations were considered including, single craft with both a monolithic mirror and deployable mirrors, as well as a constellation of craft.

This paper has several features: determining the minimum spacecraft geometric size for a micro-satellite mission, and determining how the Earth observation requirements affect these size and mass bounds. To determine whether or not micro-satellites are a feasible option for Earth observation missions several questions are answered. First, can a micro-satellite be used and constructed for this mission within the size constraints for this classification? Second, how do we want to observe the Earth (e.g. filled, sparse or interferometric apertures), and does one method apply itself better to micro-satellites than another? Three, can this system take an image? For instance even thought a particular observing strategy offers some benefits, does it truly satisfy all of the observation requirements? Lastly, is there a benefit to using micro-satellites for this mission, as compared to a more conventional spacecraft?

One result of this study is that there is a substantial mass benefit to taking a large primary mirror and dividing it into smaller apertures. These smaller apertures could either be on separate spacecraft or distributed on a single spacecraft with a large supporting structure. For this mission, since the distributed spacecraft would need to fly in close proximity, with great precision, there is no clear mass benefit over a single spacecraft. The set of distributed spacecraft would require an inordinate amount of redundancy in support subsystems (e.g. attitude control etc.)

Minimum Spacecraft Size

In order to ascertain the applicability of micro-satellites for Earth observation a model is constructed to determine the spacecraft size as a function of the data rate required by a particular mission. For the purposes of this study a spacecraft will consist of a bus and a payload. Accordingly, the two most general requirements for a working spacecraft will be correct operation of the bus and the payload. For the bus to function correctly, one of its duties is to transmit the data collected by the payload to the ground. Therefore, regardless of the type of payload and other considerations, the spacecraft must always be sufficiently 'large' to allow the data to be returned to the Earth.

To transmit a set amount of data at a specified rate, and to other tolerances (e.g. efficiencies and ground station properties, see Appendix A), the communications subsystem requires a specific power. The relationship between the data rate and other system parameters is described in equations (1) − (3). The precise definition of these terms is provided in Appendix A. The specific power of the antenna, P_c, as seen in equation (1), is proportional to the data rate, R, inversely

proportional to the gain, G_t, of the transmitting antenna, and thus inversely proportional to the square of its diameter, D_a (e.g. equation (2)).

$$P_c = \frac{E_b / N_o k T_s R}{L_l G_t L_s L_a G_r}$$

(1)

$$G = \frac{\pi^2 D_a^2 \eta_c}{\lambda_c}$$

(2)

$$P_{sa} = \frac{\pi^2 D_{sa}^2}{4} \eta_{sa} G_c (1 - degreadtaion)^{lifetime}$$

(3)

Assuming a parabolic antenna, the minimum size of the spacecraft will be determined by modeling it roughly as a flying "pancake." This configuration represents the minimum, as it assumes ideal conditions; the solar array is always pointing towards the sun, while the antenna points toward an Earth bound receiver.
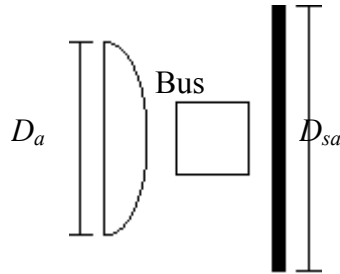

Figure 1a: A general spacecraft configuration consisting of a parabolic antenna with diameter D_a, and the solar array with diameter D_{sa}.
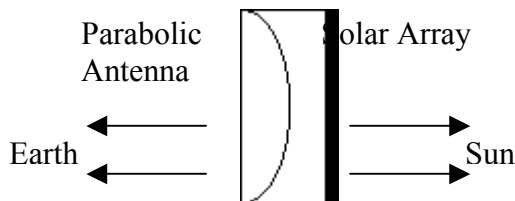
Figure 1b: A minimum sized spacecraft where D_a is equal to D_{sa}. The solar array is always assumed to be pointing directly at the sun, while the antenna is always pointing directly at the Earth. P_c is equal to P_{sa}.

As illustrated in figure 1a, the geometric size of the spacecraft, D_{sc} is the maximum of either D_a or D_{sa}. This spacecraft size will be at a minimum when the power supplied by the solar array (equation (3)) is equal to that required by the communications subsystem, P_c equal to P_{sa}.[†] The payload is then coupled to the size of the spacecraft through the amount of data that will need to be transmitted. The optimal configuration is illustrated in figure 1b. This sizing calculation is shown in figure 2, where for the parameters assumed in the appendix, [6] and an altitude of 400 kilometers, the power curves for three data rates, and the power supplied by the solar array are plotted.[‡] The solar array was assumed to be operating at beginning-of-life, so that there was no degradation.
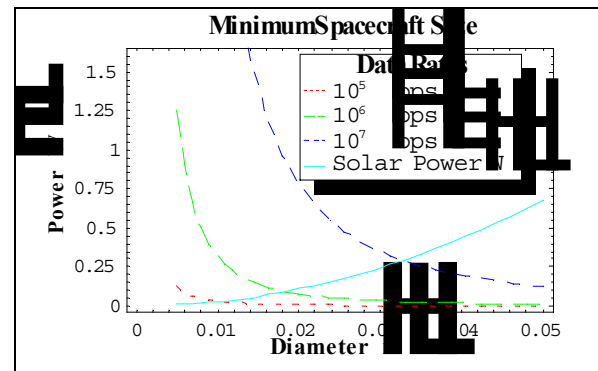
Figure 2: Solar power available as a function of diameter, and the power required for 3 data rates as function of antenna size, assuming an altitude of 400 km.

The comparison shown in figure 2 provides a minimum diameter of approximately two centimeters for a data rate of 10^6 bps, at this altitude and for the other constraints. As expected, a larger data rate increases the diameter, while a smaller data rate decreases the diameter. Therefore, the amount of data that needs to be transmitted over a time period for a mission can be used to give a lower bound on the appropriateness of small satellites.

In the minimum size calculation the distance over which the data is to be transmitted and other

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[†] RTGs will be ignored because of cost and regulation.
[‡] See appendix for choice of altitude.

assumptions can have a pronounced effect. The assumptions of both a range of data rates and a Low Earth Orbit (LEO) altitude foreshadow some of the subsequent study. In particular the optics requirements to timely map large surfaces increase the data rate and the need to collect a sufficient amount of light drives the satellite to lower altitudes or greater mirror diameters, D_m.

Independent of the optics requirement, the effect or sensitivity of transmission distance can be illustrated. The space loss for the communications system can be explicitly written as equation (4).

$$L_s = \left(\frac{\lambda}{4\pi S} \right)^2$$

(4)

Thus, the minimum size of a spacecraft at geostationary orbit can be calculated for a data rate of 10^6 bps. Using the same assumptions as above, a diameter of approximately 17 centimeters is calculated. This represents an 850% increase in diameter over a change of altitude of less then 35,000km. Therefore, based on the amount of data that needs to be transmitted there is a feasible design region that will be limited by distance and the power that can be provided.

Other considerations with regard to these calculations are to increase the capabilities of the ground station and to decrease the distance that the data has to be transmitted. The first solution, however, is limited, as the largest operational antennas are on the order of seventy meters, and three in number. Decreasing the transmission distance would be possible if a relay were used, such as TDRS, but for the purposes of this study all analysis was performed as if the spacecraft were communicating directly with a ground station.

Optics Systems

For this mission the results from the optics system represent the vast majority of the data that will need to be transmitted. Optics systems in general are limited in their ability to resolve an image by the Rayleigh criterion, and by the number of photons that can be collected. Both of these restrictions are functions of the orbital properties of the spacecraft, amongst which are its slant rage, R_s, and the nadir angle, η. While the Rayleigh criterion and photon count will be used to size various aspects of the optical system, the orbital elements along with mission objectives can be used to immediately calculate a data rate and thus find a minimum size for the spacecraft. In order to perform this calculation a "push-broom" imager was assumed at an altitude of 400 kilometers. A schematic of this type of imager is shown in figure 3.
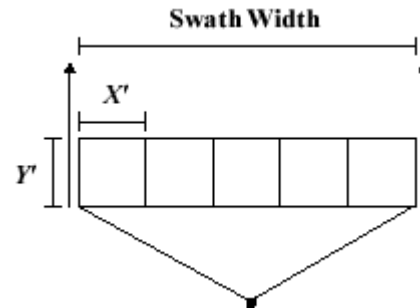
Figure 3: A schematic of a "Push Broom" imaging configuration.[7] Seen here, each square represents one ground pixel, resolution element). The arrows represent the scan direction, and the, darkened circle represents the spacecraft with the 'triangle' as its field of view.

Essentially a "push broom" imager assumes that information is collected in parallel over the swath width.

Size Implications due to Data Rate
The data rate is determined by the number of ground pixels being scanned per second, and the number of bits, b, used to encode them. The angular radius of the Earth, that is, what can be seen from orbit, limits the nadir angle. Upon calculating the varying orbital elements, a data rate can be found using equation (5). Here it is assumed that each ground resolution element is a square pixel of dimensions X' and Y', which are then equal in value.

$$R = \frac{2\eta R_s}{X'} \frac{V_g}{Y'} b$$

(5)

Figure 4 is a plot of the data rate, R, for resolutions, X', of 3 meters and 0.3 meters respectively, as the nadir angle, η, increases from zero to its limit. The maximum data rates for these systems are on the order of 4.4×10^9 bps and 4.4×10^{11} bps respectively. The impact on the Earth observing mission is dramatic; it requires orders of magnitude greater transmission capability then previously described from the communications subsystem. These data rates represent the raw data collected. If an acceptable amount of data loss is identified, modern compression schemes might be employed to reduce these values.
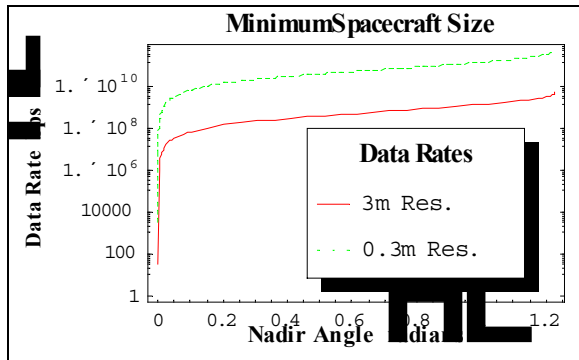

Figure 4: The data rate, R, as a function of the nadir angle, η, and the ground resolution, X'.

Using these new data rates, a minimum diameter can be identified particular to these mission requirements. This data rate comparison is plotted in figure 5. For a resolution, X', of 3 meters, the minimum size of the spacecraft is approximately 15 centimeters in diameter, however at a 0.3 meters resolution the minimum size grows close to 50 centimeters.
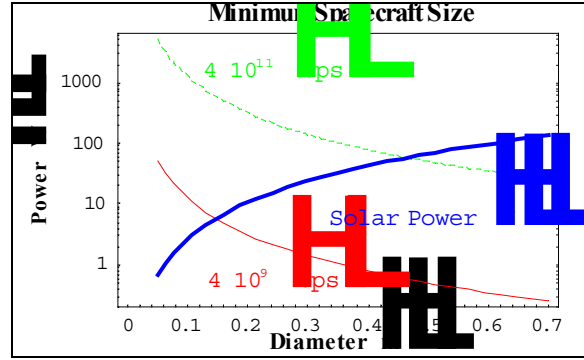

Figure 5: Solar power available as a function of diameter, and the power required for 2 data rates as function of antenna size, assuming an altitude of 400 km.

This then approaches the upper limit in size of micro-satellites, as they are generally defined as less then or equal to 0.5 meters in any direction. This size also represents the absolute minimum, and as the model becomes more realistic the solar array area will grow, and so unless the performance requirements are greatly reduced it would be difficult to fulfill this mission with a single micro-satellite. Consideration also has to be given to distributing the observation mission amongst multiple spacecraft. This effort will involve determining the minimum aperture of the optics and a more detailed study of how the optics collects the information.

Size Implications of Resolvability: Rayleigh Criterion
One feature that limits the minimum size of the aperture is the effects of diffraction. The Rayleigh criterion can be used to determine a minimum diameter needed to resolve an image.[5] The minimum ground resolution, X', is described in terms of the aperture diameter, D_m, and the wavelength of light, λ, as shown in equation (6).

$$X' = \frac{2.44\lambda S}{D_m}$$

(6)

This aperture size can be immediately compared to the size of the spacecraft, to determine if it will dominate the system. For the 3 meters ground resolution the smaller, visible, wavelength requires a primary mirror of approximately 16 cm in diameter and the longer infrared wavelength

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results in an aperture of 72 centimeters. At a ground resolution of 30 centimeters, these diameters become 1.63 and 7.16 meters respectively. A plot of this comparison is shown in figure 6.
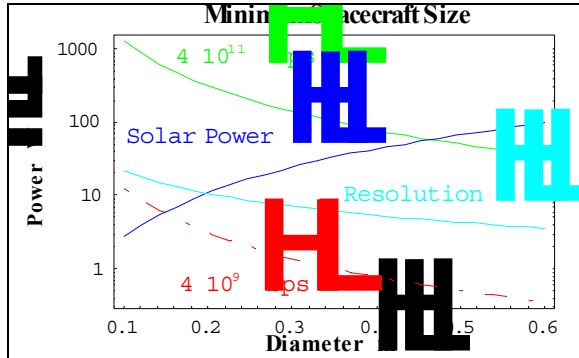
Figure 6: Solar power available and power needed to transmit data as a function of diameter of the array and the antenna, respectively, compared to the ground resolution as a function of the diameter of the aperture, assuming the Rayleigh criteria and an altitude of 400 km.

One consequence of this criterion is that as the ground resolution becomes finer (from 3 m to 0.3 m) the geometric size of the spacecraft transitions from being a feasible small satellite to a "large satellite." This "large satellite" may still be implemented monolithically, or as some type of constellation of small satellites.

Size Implications of Illumination: Photon Count

To resolve an image there is a minimum signal requirement; the number of photons that need to be collected. There is an absolute minimum of two photons per ground pixel, determined, by the Nyquist criterion, however, to actually determine some type of gray scale information, and to overcome noise in the system, the actual sample size will more realistically be on the order of 10^4 or more photons per ground pixel. To illustrate the effect of signal strength a minimum of 2 photons is assumed, with a ground resolution of 3 meters, at a representative infrared wavelength of 2.2 micrometers. From the earlier discussion, the aperture in this case is already at least 72 cm in diameter. This analysis was performed in the infrared because it is straightforward. The black

body radiation profile of the Earth will be used with a temperature of 290 Kelvin.

Using the black body profile of the Earth the energy emitted from the Earth at a specific wavelength and temperature can be calculated with equation (7). This energy is then integrated over a bandwidth, and divided by the energy per photon as seen in equation (8). The resulting value is the total number of photons being emitted from the Earth over this bandwidth. Equation (9) is the photon density as a function of altitude for a single ground pixel over the sample time.

$$E_\lambda = \frac{2\pi h c^2}{\lambda^5} \frac{1}{e^{ch/kT\lambda} - 1}$$

(7)

$$N_p = \frac{\lambda}{hc} \frac{1}{4\pi} \int_{\lambda\min}^{\lambda\max} E_\lambda d\lambda$$

(8)

$$N_p = \frac{\lambda}{hc} \frac{1}{4\pi} \frac{X'Y'}{S^2} t_s \int_{\lambda\min}^{\lambda\max} E_\lambda d\lambda$$

(9)

Initially a perfect system will be assumed where transmission losses of the signal through the atmosphere shall be neglected, and the detector will be assumed to be ideal with one detector pixel per ground pixel in a swath (cross track pixel). The integration time, equation (10), is the exposure time, which is just a function of the orbit. The integration time can vary, as it is technically defined as the exposure time multiplied by the ratio of detector pixels to pixels in a swath, as shown in equations (11) and (12). In other words the integration time can be improved by employing a better focal plane imager (e.g. a matrix imager – an array of "push brooms") that accumulates photons of the imager in successive arrays.

$$t_s = \frac{Y}{V_g}$$

(10)

$$V_g = \frac{2\pi R_E}{P}$$

(11)

$$t_i = \frac{Y}{V_g} \frac{N_m}{Z_c}$$

(12)

Figure 7 is a plot of the number of photons that can be collected for a 3 meters resolution at an altitude of 400 km, at a wavelength of 2.2 microns as a function of diameter. This plot uses a bandwidth of 0.4 microns. At this bandwidth the mirror collects 2 photons approximately when the aperture is large enough to achieve the spatial resolution requirements. Also shown is a bandwidth of 1.2 microns, still centered about 2.2 microns.
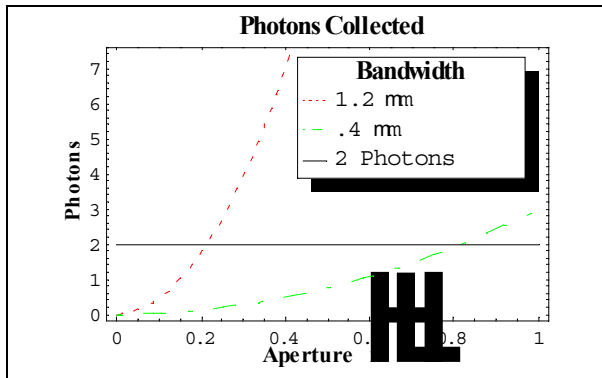
Figure 7: Number of photons as a function of the aperture diameter. This assumes a 3 meters resolution at an altitude of 400 km, at a wavelength of 2.2 microns.

Therefore, to collect a suitable number of photons the aperture, the bandwidth, or the complexity (array size) of the imager, must be greatly increased. Another alternative is to change the wavelength requirement, e.g. drop the infrared requirement and image in the visible spectrum using reflected and scattered sunlight.

Another feature to consider is the signal to noise ratio of the sample. Equation (13) is the signal to noise ratio when there are no loss terms, and the only efficiency to be considered is the quantum efficiency, which will taken as 60%.[5] Again, this is an ideal case, and the true signal to noise ratio will be less.

$$SNR = \sqrt{S_i Q t_s}$$

(13)

Using a sample value of 100 the signal to noise ratio is less then one. At this altitude, and for a resolution of 3 meters, for a signal to noise ratio of 2, approximately 8000 photons must be collected. Therefore, while the Rayleigh criterion specifies the aperture needed to achieve a spatial resolution this size may be less then that required. While the Rayleigh criterion determines the aperture size, no matter what type of optical system is used, its total area must be greater then or equal to that of a filled aperture system that will collect enough photons to resolve the image. High-resolution images in the infrared are not possible with filled aperture optics until the size of the mirror is considerably larger then the minimum spacecraft size.

One consequence of this study of the optics requirements is that while the data rate does impact the minimum size of the craft in terms of communications and power, the ability to collect light (i.e. the photon count) and the ability to resolve a ground object (i.e. Rayliegh's criterion) drive the geometric size of the payload, and for higher resolutions, the craft. It is not clear at this point how this size translates into system mass, but it is clear the optics configuration is critical. Since the optics components here are the "largest" in terms of geometric size, they are examined in more detail. The optics configuration may be traditional, (e.g. a Cassegrainian telescope with usual mirrors). Or it could be more unusual in that it might consist of inflatable or membrane surfaces, apertures that are sparse in that they are distributed along a truss or tethered structure, or the optics system may consist of a constellation of free flying craft. In order to ascertain or compare the relative merits of a least some of these configurations, the ability to form an optical surface in space is studied to connect the geometric size requirement of the optics with mass.

Mirror Mass

The mass of the optics system is likely to be considerable, especially for high resolutions

where large apertures are needed. Large mirrors can be extremely heavy. For instance the 2.4 meters mirror on the Hubble Space Telescope has a mass of 826 kg.[9] This mirror would consume all the mass of a "micro-satellite!" Clearly some alternative to a simple filled aperture configuration such as distributed or sparse aperture, interferometric or deployable optics, including deployable-segmented and membrane type mirrors need to be at least considered. Our purpose here is to lay a foundation to compare the masses of numerous optical configurations.

It can be shown that a function exists for the mass of a mirror as a function of its diameter. For hard and composite type (including porous or honeycomb reinforced glass) mirrors or any other mirror that maintains its shape by bending stiffness, the mass of the mirror is proportional to the diameter of the mirror to the fourth power. This relationship is shown in equation (14), where C_m is a constant that depends upon the technology of the mirror and has units of kg/m^4.[1]

$$m = C_m D_m^4$$

(14)

It is important to note several aspects of equation (14). First the mass of the mirror increases, not as the area of the mirror, but as the area squared! Second, this relationship holds under numerous loading scenarios. This fourth order polynomial comes about because traditional mirrors are 'plate like structures,' and are governed by a fourth order differential equation. Hence, they yield a fourth order solution.[3] For instance mirrors are usually designed to have a specified displacement limit, e.g. $\lambda/10$, as it is re-oriented in a gravitational field. There are also other loads on the structure as it is launched. What ever these loads or specified limits (e.g. displacement or yield), the scaling of the mirror will be as in equation (14). Finally, it should be noted that this solution is for simple passive mirrors and does not take into account any supporting mirror structure or the rest of the components of the optical payload. If the mirror was actually designed to always have a 'supporting structure', e.g. a back

truss, it would have to be included in the 'mirror mass'.

The "technology" of the mirror can be described through a constant. The constant can be "calibrated" by knowing the mass and diameter of existing mirrors. The mass constant will vary for different types of mirrors and deflection limits. Ikonos' primary mirror is 0.7 meters in diameter and has a mass of 13.4 kilograms.[10] This is a composite or porous mirror that has a predicted mass constant of 64.75 kilograms/meter^4, while in contrast a solid glass mirror has a predicted constant of approximately 1000 kilograms/meter^4. Some representative values are listed in table 1.

Table 1: Mirror mass technology

Mirror	Diameter (m)	Mass (kg)	Cm (kg/m^4)
Solid	1	1000	1000
Hubble	2.4	826	25
Ikonos	0.7	13.4	56

Both the Hubble and Ikonos mirrors represent superior technologically when compared to a 'solid mirror'. Differences in the constant C_m may be due to technology (e.g. lightening holes), specified displacement limit or supporting structure. A plot of a solid mirror type against a composite can be seen below as figure 8.
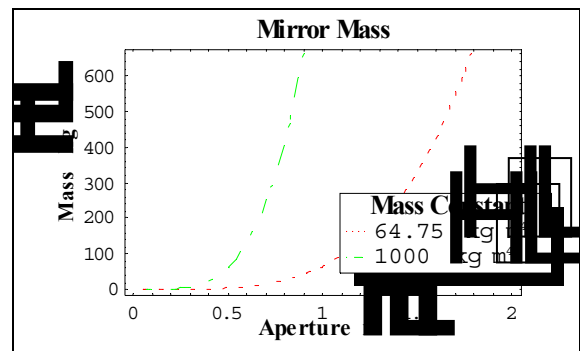
Figure 8: Mirror mass as function of size and type of technology.

The important point here is that this fourth order scaling of the main mirror is critical. One interpretation is that as the aperture needs

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increase, extensive effort should expended to improved technology and thereby decrease the scaling factor Cm. Another interpretation is that the mass used for the primary mirror is an "opportunity" or is available to "fund" other less traditional types of light collection.

Less Traditional Primary Surfaces

This is a discussion of some alternative optics configurations in an attempt to reduce system mass.

Distributed Collection
Using this fourth order relation the savings in mass that segmenting a mirror can have can be calculated. The mass of the mirror can be calculated as it is segmented into a series of smaller mirrors of which the total area is constant, and is plotted as figure 9 below.
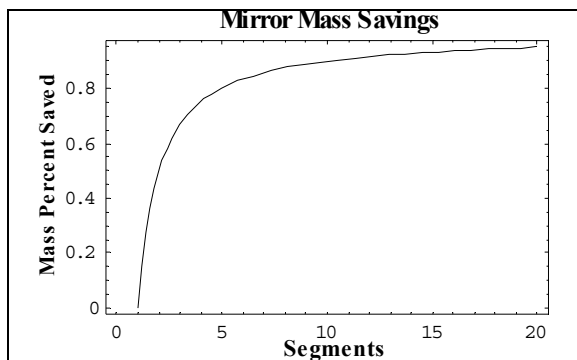

Figure 9: Mass savings of a segmented filled aperture mirror.

This curve is irrelevant of the type of mirror used, and only assumes equation (14). A savings of 90% can be obtained once the mirror has been segmented into ten pieces. This mass "savings" comes about because the mirror thickness can be reduced for smaller mirrors. It assumes that as the mirror is segmented, perhaps to put each mirror on a separate free-flier, that there is no additional cost. Thus this mass "savings" is what can be used for the additional mass of additional spacecraft.

 For a mirror such as Hubble's this would be a savings of hundreds of kilograms, however for Ikonos this would reduce the mass of the primary mirror by only 12 kilograms. Therefore, it may not always prove of value to segment a primary mirror. For a micro-satellite mission there is still the size constraint on the primary mirror, therefore using distributed aperture system applied over a constellation will be examined.

To compare the savings in relation to other system parameters, the primary will be assumed to be some percentage of the total mass of the spacecraft. Each time the mirror is segmented, the original spacecraft will be replaced with a bus and a mirror of the size determined by the number of segments to maintain the total net aperture. This is plotted as figure 10 below. For purposes here this is equivalent to setting the primary mirror mass as the mass of the payload.
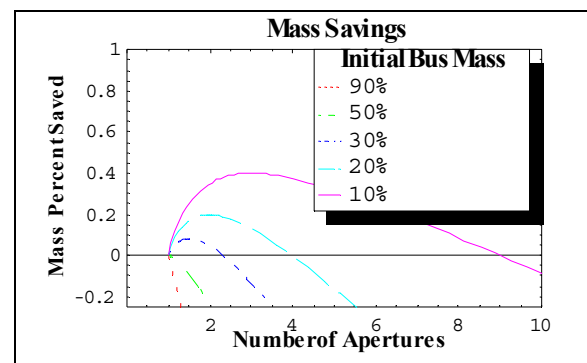
Figure 10: Mirror savings by segmenting as function of the initial mirror mass to system mass.

Unless the mass of the primary mirror makes up approximately 50% of the total original system mass, this solution actually requires more total mass! Even then, the mass savings is small, and peaks between two and three segments. This type of constellation however is nonsensical; in general if the payload of a spacecraft could be reduced in size so to could the bus size. The maximum size that each bus could be as the primary mirror is distributed amongst the spacecraft, if the constellation of spacecraft is to have a total net mass less then or equal to the mass of the original spacecraft being replaced, is plotted as figure 11 below.

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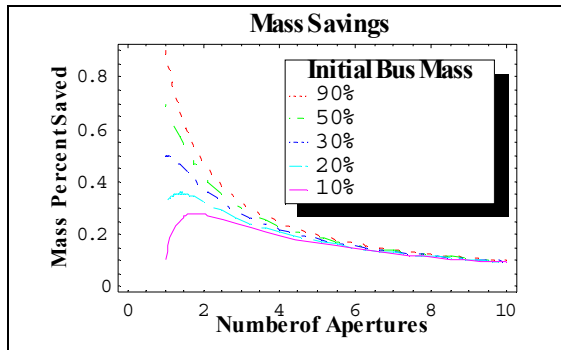
Figure 11: Mass savings as function of number of apertures, for different bus masses of the single spacecraft system.

If the primary mirror is initially 70%, or greater, of the total system mass, then the spacecraft bus mass can increase from its initial value. Therefore, replacing a spacecraft with a constellation of spacecraft can be beneficial. This implies that the ideal use for a distributed aperture system would be to replace an extremely large mirror. Figure 11 can also be used to determine a mass limit for each micro-satellite if a group of them were to replace a spacecraft such as Ikonos.

Membrane Mirrors
To illustrate the difference between a surface that maintains its shape though bending (e.g. a plate) and one that maintains its shape by tension (e.g. a membrane), a membrane mirror's mass would scale as in equation (15). A membrane mirror (just for the surface) grows as the area of the membrane. It does not have to become thicker as the size is increased.

$$m = C_m D^2$$

(15)

This scaling calculation, especially for the membrane ignores the 'supporting compressive structure' that is required to support the membrane surface. Figure 12 is a plot of a membrane type mirror against a composite mirror, neglecting the supporting structure of both.
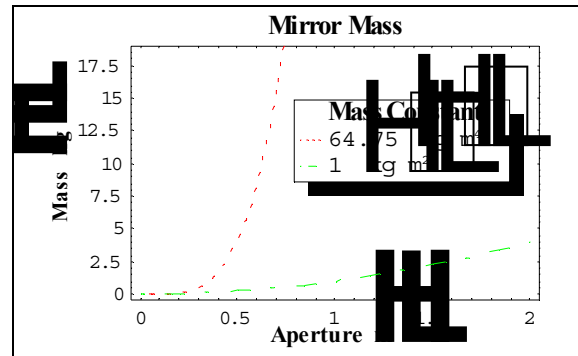
**Figure 12:** Membrane mirror mass plotted against composite mirror mass.

Typical mass coefficients are on the order of less then 1 kilogram per square meter, using materials such as aluminized Mylar. For mirrors of this type there is no gain to segmenting the mirror, aside from reducing the size of an individual mirror. A membrane allows a mirror to be deployable, and because of its low mass properties is ideal for a micro-satellite. That being said, a deployable segmented hard mirror would also be a solution, however, it would still way more then a membrane type mirror, especially when supporting structure is taken into account.

Tethers
Aside from increasing the aperture of a mirror, the only other method to increase the photon count at some altitude would be to increase the sample time.[§] This can be accomplished by "stacking" the detector pixels, having more detector pixels then ground pixels, or by having an orbital velocity less then that of the local circular orbit. One proposed method to accomplish the later is a tethered spacecraft. If the tether is assumed to be a rigid beam, then the assumption is made that the orbital velocity of any point on the tether is that of the center of mass of the system. Therefore, the payload, which would be at the lower end of the tether, would have an orbital velocity less then the local circular velocity at that orbit. However, the

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[§] Sample time is also referred to as integration time, but because the number of detector pixels has been set equal to the along track pixels, this time is just the exposure time, which has been termed the sample time.

benefit of this is negligible as seen in figure 13 below.
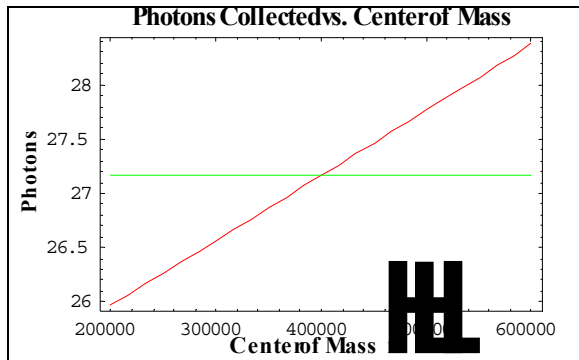
Figure 13: The number of photons collected for an optical payload at 400 Km, as the center of mass of a tethered spacecraft varies from 200 to 600 Km.

Here the diagonal line represents the number of photons collected as the center of mass ranges from 200 to 600 kilometers. The horizontal line represents the number of photons this system would collect at 400 kilometers. Changing the location of the center of mass of the system by 200 kilometers has a very small on effect on the total number of photons collected. In order to raise the center of mass by this amount the tether would have to be greater then 200 kilometers in length, greater then an order of magnitude larger then any previously tested tether. The increase in complexity of the system, relative to the increase in signal properties provided by the tether, hardly warrants its use.

Tethers have also been suggested as a means to lower, the orbit of the telescope, the payload, and thus decrease the diameter, D_m, mirror. However, atmospheric drag would still be significant unless the upper mass were at an altitude of less then 400 kilometers. Furthermore, if the optics payload were at an altitude of 300 kilometers, the primary mirror would still be required to have a diameter of greater then 1 meter, based on the Rayleigh Criterion. Thus, again because of complexity, mass, and other issues the benefit of a tether is negligible.

Discussion of System Mass

By imposing some limitations on the design of a micro-satellite a quick analysis can be performed using figure (11) to determine if there is a benefit to using micro-satellites over a more conventional approach. For instance if the maximum dimensions for a micro-satellite and its primary mirror are specified, the number and mass of spacecraft needed to replace a craft such as Ikonos can be determined. In the same way, the number of spacecraft needed to achieve a greater spatial resolution, for a specific wavelength, can be determined. Using Ikonos, 0.7 meters diameter primary mirror, it would take at least three spacecraft to achieve this aperture. Ikonos had a net mass of 726 kilograms, 171 kilograms of which are made up by the optical payload, the primary mirror as stated before was 13.4 kilograms. As the mass of the primary mirror makes up less then 2% of the total system mass it will be assumed that the entire optics payload will scale as D^4, the same as the mirror. This is not an entirely unreasonable assumption as it assumes that a maximum bending deflection is being allowed. So, assuming that optics payload scales as the mirror, then the optics payload makes up 24% of the spacecraft, at three segments the new bus mass is just less then 31% of the original system mass, or 222.4 kilograms.

The next limit that would be set would be the system mass. For the purposes here 200 kilograms will be chosen. Therefore to replace the spacecraft more then three satellites would be needed, with this limit. If the limit were made more severe, for instance 50 kilograms then approximately 15 satellites would be needed. Before continuing, the number of mirrors needed with these restrictions to replace a single mirror capable of a 0.3 meters resolution at 0.5 microns will be determined. This would require a single mirror of 1.63 meters in diameter, or 14 mirrors at 0.44 meters in diameter. This brings about the second problem in replacing a single spacecraft with a constellation, each bus must provide the

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same functionality as that of the original spacecraft's. Therefore, they must meet the same lifetime requirements, pointing, provide power to the varying subsystems, and so on.

Conclusions

Deployable mirrors offer advantages to larger Earth observing spacecraft as well as micro-satellites, but they reduce the need for precision formation flying, light recombination, and general spacecraft function issues when used onboard micro-satellites.

Until such systems become operational, micro-satellites are most likely not a cost effective solution for high-resolution images of the Earth. Taking such images is problematic because of the integration or sample time. To collect the number of photons needed to resolve such an image a large mirror aperture is needed, a problem that can be avoided by looking away from the Earth. Missions such as TPF and SIM may have dwell times on the order of hours to days, whereas for a spacecraft orbiting the Earth these times are on the order of microseconds. Micro-satellites may be used for a plethora of space-based observations, and data gathering missions, however, at the present, they are restricted for high-resolution image gathering of the Earth because of optical limitations.

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Appendix A

Variable	Description	Value
P_c	Communications Power	
E_b/N_o	Received Energy to Noise Density Ratio	15
K	Boltzmann's Constant	
T_s	System Noise Temperature	135 K
R	Data Rate	10^6 bps
G_t	Transmit Antenna Gain	Equation (2)
G_r	Receive Antenna Gain	Equation (2)
D_a	Transmit Antenna Diameter	
D_r	Receive Antenna Diameter	5.3 Meters
L_l	Line Loss	0.5
L_a	Propagation and Polarization Loss	0.9
L_s	Space Loss	Equation (3)
η_c	Antenna Efficiency (Equation 2)	0.6
λ_c	Communications Wavelength	
η_{sa}	Solar Array Efficiency (Equation 4)	.25
S	Altitude	
G_c	Solar Constant	
S_w	Swath Width	
X'	Cross Track Resolution	
Y'	Along Track Resolution	
V_g	Ground Track Velocity	
b	Bits per pixel	
R_s	Slant Range	
η	Nadir Angle	
D_m	Mirror Diameter	
Z_c	Cross Track Pixels	
Z_a	Along Track Pixels	
Z	Total Number of Pixels	
E_λ	Luminosity	
h	Plank's Constant	
c	Speed of Light	
T	Temperature (Earth)	
λ	Wavelength (Observed)	
N_p	Photon Number	
R_E	Radius of Earth	
N_m	Number of Pixels	
P	Orbital Period	
m	Mirror Mass	
C_m	Mirror Mass Constant	
SNR	Signal to Noise Ratio	
S_i	Signal	
Q	Quantum Efficiency	0.6
t_s	Sample Time	
t_i	Integration Time	

Appendix B

An orbit of 400 kilometers was assumed throughout this study. This assumption is based upon orbital requirements of a spacecraft for one year of operation. To calculate the propellant requirement for this the spacecraft was assumed to be placed in its orbit, and required to carry on it the amount needed to maintain orbit for a year, and then de-orbit the spacecraft. Atmospheric drag was taken to be the only orbital loss term, and average values were used for the atmospheric density (Mathematica, MeanDensity function). Thus, the sum of the two was considered to be total requirements for the system. Calculations were non-dimensional, and performed as a function of the initial spacecraft mass. This can be seen as figure ? below. All equations used can be found in the orbital equation reference.
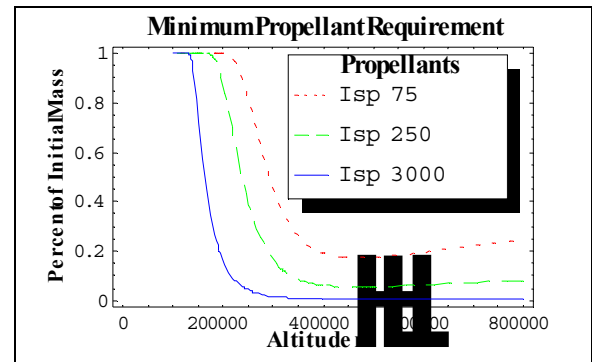
Figure B1: Propellant requirements in terms of initial mass, as a function of altitude. The Isp of 75 seconds is for a cold gas system, 250 a hydrazine, and 3000 and ion.

Plotted are the requirements using cold gas, hydrazine and ion propulsion schemes. Cold gas being the most commonly used propellant onboard small spacecraft, because of its cost and simplicity. Here, it can be seen that the minimum requirements for a cold gas system, are at approximately 400 kilometers. Hence, the reason this altitude was used.