PARAMETRIC DESIGN CURVES FOR PAYLOAD POWER AND MASS CAPABILITIES OF NON-GEO SMALLSATS BUSES/LAUNCHERS.

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ABSTRACT

There is a need to reduce cost and perform an optimization on smallsats and standardsats before options for a payload/bus/launch vehicle combination are selected. Bus manufacturers want naturally to protect their proprietary data; at the same time, a smallsat designer cannot decide on this combination without trade-off analysis. Geosynchronous bus manufacturers realized, some time ago, that it was in their interests to release some of the payload trade-off curves to potential customers.

This paper provides a methodology to reduce cost and optimize the selection of this combination, for Low & Medium earth orbits satellites. This generalized approach for elliptical orbits, is extended from a previous paper on circular non-GEO orbits. It provides equations for the net payload power and mass available to the system, for varying bus launcher sets, for elliptical (and circular) orbits, taking into account orbit eccentricity, mass to orbit, power generation/storage and fuel required for drag make-up. Some selected examples are provided for the payload power and mass for different launchers and the total payload equivalent mass at varying elliptical altitudes. This methodology is adequate for a first cut optimization, for elliptical LEO's and MEO's, with direct injection launches. Further refinements require detailed knowledge of the power system as well as other data of a given bus, which are best evaluated with the bus manufacturers.

1.0 INTRODUCTION

At the early stage of the development of a microsat/smallsat/standardsat, the system designer must consider all potential candidate of available buses and launch vehicles, for the best cost effective way to carry the payload into preferred orbit (eccentricity, altitude, inclination), reliability for different possible lifetimes, etc.

The question addressed here is how to pick optimum bus/launcher vehicle combination. Traditionally, one could go from the start, to a bus manufacturer who could undoubtedly provide all options for his own bus utilisation. This was practised 15 years ago, for geosynchronous satellites. Since then (Ref. 1 & 2), bus manufacturers have started to make available the payload Power-Mass capabilities of their buses, under a set of conditions such as lifetime, eclipse, etc.
Recently, the need for cost reduction and optimization of all satellites and particularly smallsats, microsats and minisats has reached paramount importance.

What are the payload power & mass capabilities of commercially available buses ?

Manufacturers normally publish bus payload capabilities that are designed and presumed optimized for specific missions, i.e. for a given orbit altitude, eccentricity ellipse, lifetime, reliability. There is a need for the smallsat systems designer to translate these capabilities to fit specified potential needs and to provide cost and performance. Curves similar to those available for geosynchronous satellite buses are required but not available for non-GEO orbits.

The methodology derived for GEO orbits was developed recently (ref. 3) for Low and Medium Earth Orbits, LEO & MEO, but only for circular orbits. This paper extends the methodology to more generalized orbits such as Elliptical Orbits, using some of the ground work developed in Ref. 4.

2.0 METHODOLOGY FOR GEO

The model is based on the concept of the power mass curve as described for geosynchronous spacecraft [Ref. 1 & 2].

This concept recognizes that, for a particular design of spacecraft bus, payload mass and power can, within limits, be traded off against each other. A plot of payload mass versus payload power defines a region within which particular combinations of mass and power can be accommodated.

For a given payload consisting of antenna systems and transponders and requiring some value for *P-M*, there are several satellite buses and several types and sizes of launchers. The selection of a specific set of options is dictated essentially by its feasibility, the most efficient bus utilization, and overall cost considerations. At the initial system design phase, many options for different *P-M* payload values can be generated as a result of various trade-offs. Such trade-offs are performed here for LEO, to optimize the payload for different requirements. In previous work [Ref. 1 & 2], a simple and effective method of optimizing the payload for all potential buses was developed for GEO satellites.

The payload is characterized by its mass and power, as shown in Figure 1, for a geosynchronous satellite. The payload *P-M* design value can be parametrically varied by the bus manufacturers in two ways:

1- For a given stationkeeping condition, dc power, i.e., solar sails, batteries for eclipse capability, and power conditioning circuits, can be traded against the mass made available to the payload.

2- For given dc power conditions, the reduction of number of years of stationkeeping from the design value, say from 10 to 7 years, can lead to savings in the fuel mass, and hence more mass is made available to the payload. Figs. 1 & 2 below illustrate the results for GEO satellites.

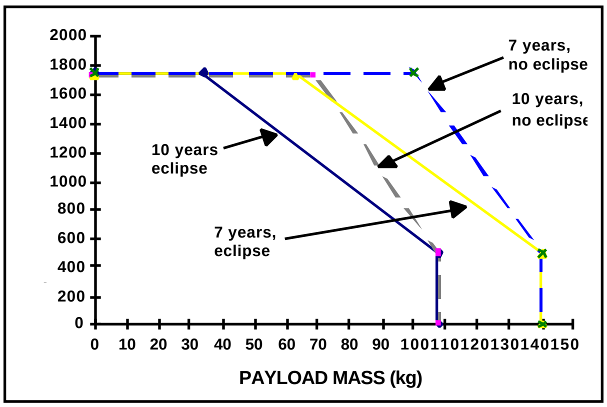

Figure 1 Parametric variation of payload P-M with lifetime for GEO satellites, with and without eclipse.

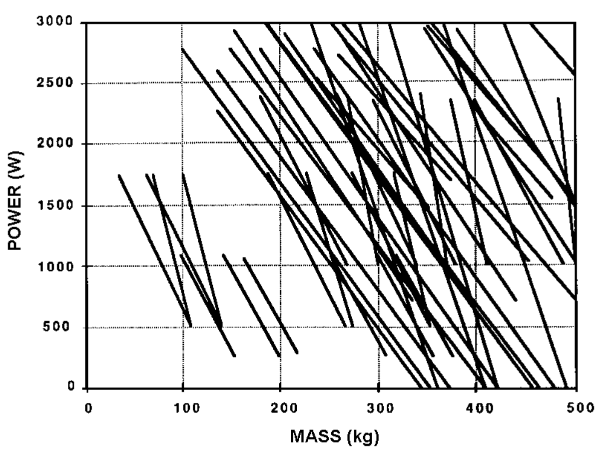

Figure 2 Buses P-M parametric curves for all lifetimes, for GEO satellites [Ref. 3].

3.0 METHODOLOGY EXTENDED FROM CIRCULAR TO ELLIPTICAL ORBITS

A previous paper described a set of coupled equations which could be used to derive the Payload mass and power accommodation available for particular combinations of spacecraft bus and launch vehicle. Detailed equations were presented for computing the following:

- Maximum Launch Vehicle payload mass to a given orbit.
- Power subsystem (solar array plus battery) mass required to support a given Payload power consumption in a given orbit including the effects of eclipse and illumination angle.
- Fuel mass required for drag make-up in a given orbit.

The equations presented are valid for circular orbits only. In order to extent the validity to include elliptical orbits it is necessary to account for the effects of a non-zero eccentricity on the launch vehicle performance (mass to orbit), the spacecraft power generation and the fuel required for drag makeup.

Launch Vehicle Performance

All elliptical orbits (including circular) have a specific mechanical energy of 2GM/a where: "G" is the Gravitational constant, "M" is the mass of the earth and "a" is the semi-major axis. Parametric plots of mass to a circular orbit versus altitude, derived from manufacturers data, can be used for orbits with non-zero eccentricity by defining an equivalent circular altitude:

$$h_c = (h_a+h_p)/2$$

where: h_a is the altitude of apogee
h_p the altitude of perigee.

Power Generation

Extension of the power generation calculations to elliptical orbits must take into account the effect of the orbit eccentricity on the eclipse duration. Since, for elliptical orbit at an arbitrary inclination, the argument of perigee is continuously changing, the eclipse geometry becomes rather complex if the orbit is allowed to evolve naturally. A desired argument of perigee can be maintained by use of spacecraft propulsion but only at the expense of carrying a significant amount of fuel. At an inclination of 63.4 degrees the rate of change of argument of perigee become zero. If this condition is combined with the additional restriction that the argument of perigee be 270 degrees (apogee over the north pole), a relatively simple expression for maximum eclipse duration can be derived, which also applies for circular orbits at any non-sunsynchronous inclination.

Maximum eclipse will occur when the sun is in the plane of the orbit. If the solar declination is zero, then the true anomalies of the eclipse positions $<_{e1}$, $<_{e2}$, will be given by:

$$\cos(<_1) = R_e/r_{e1}$$

$$\cos(180-<_2) = R_e/r_{e2}$$

where R_e is the earth equatorial radius
r_{e1}, r_{e2} are the radii at the eclipse positions

In general for elliptical orbit:

$$r = a(1-e^2)/(1+e.\cos(<))$$

Thus:

$$<_{e1} = \cos^{-1}[R_e/(a(1-e^2)-e.R_e)]$$

$$<_{e2} = 180-<_{e1}$$

The eccentric anomalies E_{e1}, E_{e1}, at the eclipse boundaries are given by:

$$E_{e1} = \cos^{-1}[(e+\cos(<_{e1})/(1+e.\cos(<_{e1})]$$

$$E_{e2} = \cos^{-1}[(e+\cos(<_{e2})/(1+e.\cos(<_{e2})]$$

The mean anomalies M_{e1}, M_{e2}, at the eclipse boundaries are given by:

$$M_{e1} = E_{e1} - e.\sin(E_{e1})$$

$$M_{e2} = E_{e2} - e.\sin(E_{e2})$$

The eclipse duration, T_{emax}, is given by:

$$T_{emax} = (M_{e1} - M_{e2})/n$$

where n is the mean motion given by:

$$n = \%(GM/a^3)$$

Drag make-up Fuel

Since the altitude of an orbit with non-zero eccentricity will vary continuously, the atmospheric density and hence the drag will also vary around the orbit. At altitudes above 1000 km, drag can normally be neglected thus only that portion of the orbit below 1000 km need be considered. An approximation to the fuel required for drag make-up can be derived by splitting this portion of the orbit into several sections, computing the drag at the start point of each section and assuming that this drag applies over the segment. By utilising the symmetry about perigee, computing the drag at three positions provides a total of six segments. The expressions derived are also valid for circular orbits since in this case the same drag will be computed at each position. The computation proceeds as follows:

The true anomaly at the cut-off altitude
$<_{lim}$, is given by:

$$<_{lim} = \cos^{-1}[a(1-e^2)/(e.r_{lim}-1/e)] \qquad h_a > h_{lim}, h_p < h_{lim}$$

$$= 180 \qquad h_a < h_{lim}$$

$$= 0 \qquad h_a > h_{lim}, h_p > h_{lim}$$

where r_{lim} is the radius at the altitude cut-off.

Two intermediate positions, subscript 2 and 3 between perigee and the cut-off are chosen as:

$$<_2 = <_{lim}/3$$

$$<_3 = 2.<_{lim}/3$$

The corresponding radii are given by:

$$r_p = a(1-e)$$

$$r_2 = a(1-e^2)/(1+e.\cos(<_2))$$

$$r_3 = a(1-e^2)/(1+e.\cos(<_3))$$

The altitudes are:

$$h_p = r_p - R_e$$

$$h_2 = r_2 - R_e$$

$$h_3 = r_3 - R_e$$

The atmospheric densities are given by:

$$D_p = \text{antilog}(x.h_p^2 + y.h_p + z)$$

$$D_2 = \text{antilog}(x.h_2^2 + y.h_2 + z)$$

$$D_3 = \text{antilog}(x.h_3^2 + y.h_3 + z)$$

Where x, y, z are constants relating atmospheric density to altitude for a particular solar condition.

The orbital velocities are given by:

$$V_p = \%[2.GM(1/r_p - 1/2a)]$$

$$V_2 = \%[2.GM(1/r_2 - 1/2a)]$$

$$V_3 = \%[2.GM(1/r_3 - 1/2a)]$$

The eccentric anomalies are given by:

$$E_2 = \cos^{-1}[e + \cos(<_2)/(1+ e.\cos(<_2)]$$

$$E_3 = \cos^{-1}[e + \cos(<_3)/(1+ e.\cos(<_3)]$$

$$E_{lim} = \cos^{-1}[e + \cos(<_{lim})/(1+ e.\cos(<_{lim})]$$

The mean anomalies are given by:

$$M_2 = E_2 - e.\sin(E_2)$$

$$M_3 = E_3 - e.\sin(E_3)$$

$$M_{lim} = E_{lim} - e.\sin(E_{lim})$$

The time of flight from perigee to position 2 is given by:

$$t_{p-2} = M_2/n$$

The time of flight from position 2 to position 3 is given by:

$$t_{2-3} = (M_3 - M_2)/n$$

The time of flight from position 3 to the cut-off is given by:

$$t_{3-lim} = (M_{lim} - M_3)/n$$

The fuel mass required over the mission duration t, from perigee to position 2 is given by:

$$M_{f1} = [0.5\ D_p\ C_d.A.V_p^{\ 2}/(I_{sp}.g)\ ].(t_{p-2}/T).t$$

Where C_d is the drag coefficient of the spacecraft

A is the cross-sectional area of the spacecraft

I_{sp} is the specific thrust of the spacecraft propulsion system

g is the acceleration due to gravity

The fuel mass required over the mission duration t, from position 2 to position 3 is given by:

$$M_{f2} = [0.5\ D_2\ C_d.A.V_2^{\ 2}/(I_{sp}.g)\ ].(t_{2-3}/T).t$$

The fuel mass required over the mission duration t is given by:

$$M_{f3} = [0.5\ D_3\ C_d.A.V_3^{\ 2}/(I_{sp}.g)\ ].(t_{3-lim}/T).t$$

The total fuel required is given by:

$$M_f = 2(M_{f1} + M_{f2} + M_{f3})$$

4.0 SELECTED RESULTS

The various power/mass dependency relationships described above are solved iteratively for each a particular combination of payload, orbit, mission life, spacecraft bus and launch vehicle. This is to determine if a solution exists within the limiting constraints imposed on payload mass, bus power subsystem mass, orbit maintenance fuel mass and total spacecraft mass. A probability of success at the end of the mission is computed from the combine of payload and bus reliability. Dimensional limits are tested by adding payload and bus dimensions and comparing the result to the launch vehicle fairing dimensions. Pointing limits are tested by comparing the payload requirement to the bus capability.

To illustrate the behaviour of the model, computations were made with two bus/launch vehicle combinations, encompassing one spacecraft bus design and two launch vehicles. In each case the payload mass and power capabilities were calculated for circular sunsynchronous orbits with a noon node, over a range of altitudes and for three mission durations and, for comparison, for an elliptical orbit at an inclination of 63.4 degrees, for a single mission duration. The bus, designated Bus 1, is intended to represent a typical medium sized, dual redundant, small bus, with a dual axis articulated array. Launch vehicle 1 is the Pegasus XL and launch vehicle 2 the standard Taurus.

Figures 3 and 4 show plots of payload power versus payload mass for the two combinations of bus and launch vehicle at selected altitudes and lifetimes. For elliptical orbit cases the altitude referred to is the equivalent circular altitude. In each case the boundaries are seen to be linear. In figure 3 the altitude and lifetime are such that the launch vehicle performance is the deciding limit. The sloping upper boundary indicated that payload mass can be traded against payload power. In figure 4 the altitude and lifetime are such that the assumed limits on power subsystem mass and payload mass are reached before the total mass limit since, in these cases, the launch vehicle has significant excess capacity. The result is that no trade-off between payload mass and power is possible indicating that this combination is suboptimum. The general features are the same for both the circular and elliptical cases, a longer eclipse duration for the elliptical orbit, however, leads to the power subsystem mass limit being reached for a lower value of available payload power.

Figures 5 and 6 show a wider range of results plotted by assuming that payload power (solar panels and batteries) can be equated to mass at 0.1 kg/W and combined with payload mass to form an equivalent payload mass. This assumption seems to provide a reasonable fit for the specific performances assumed for the array and batteries.

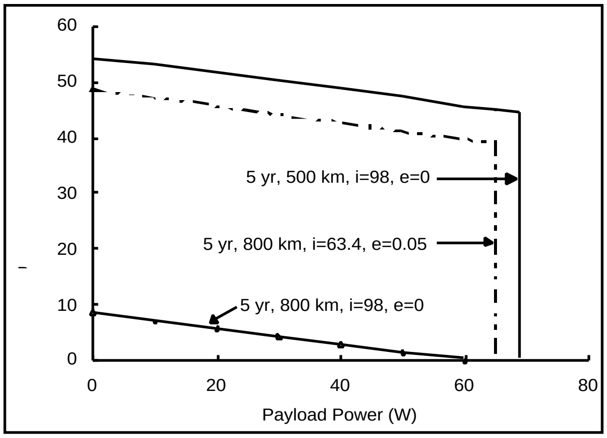

Figure 3 Payload Power vs Mass Bus 1 LV1

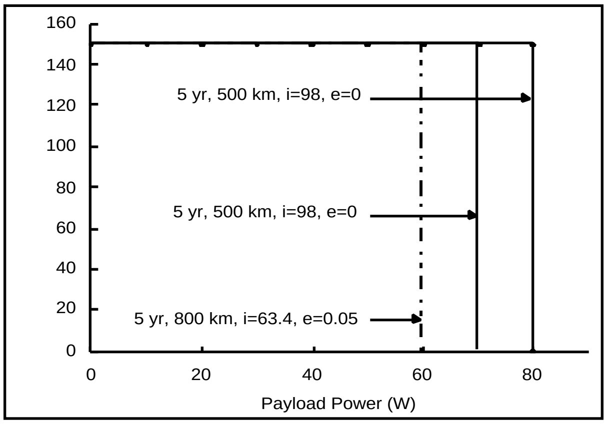

Figure 4 Payload Power vs Mass Bus 1 LV2

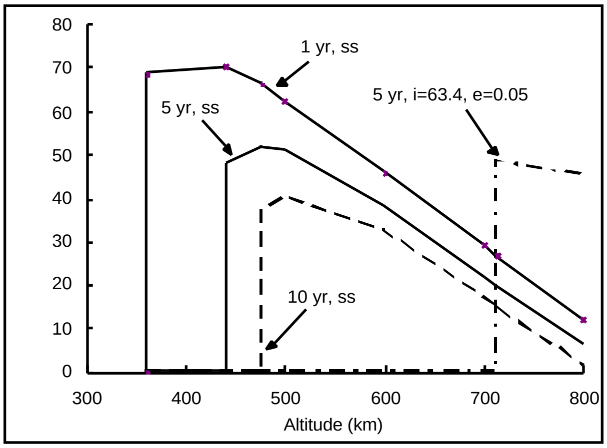

Figure 5 Equivalent Payload Mass vs Altitude Bus 1 LV1

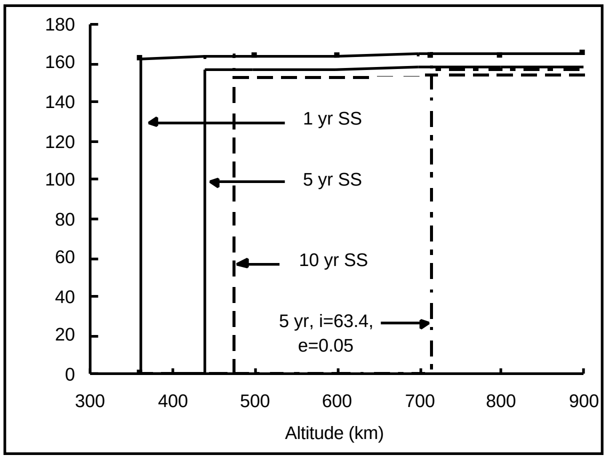

Figure 6 Equivalent Payload Mass vs Altitude Bus 1 LV2

The cases illustrated in figure 6 show, at the lower altitudes, the effect of the limit set on fuel capacity. Since, when the fuel required exceeds the limit, the mission cannot be accomplished. The payload equivalent mass is plotted as zero. As the altitude is increased the fuel mass quickly ceases to be the limiting condition and is replaced by a the limit on total spacecraft mass imposed by the launch vehicle performance. This causes the equivalent payload mass to reduce with increasing altitude. The elliptical case, whilst showing the same behaviour, reaches the fuel limit at a much higher equivalent circular altitude than for the circular cases. This is because the atmospheric density around perigee and hence the drag, is much higher. This more than offsets the low drag encountered around apogee.

The case illustrated in figure 5 shows the effect of using a more powerful launch vehicle. In this case the lower altitude cut-off is again set by the fuel limit. At higher altitudes the payload and power subsystem mass limit can both be accommodated and the only variation is a slight increase in power with increasing altitude due to the reducing eclipse duration.

5.0 CONCLUSION

The need to reduce smallsat cost demands that the system planner performs a trade-off analysis and an optimization, for all potentially desirable payload power and mass. This is achieved by considering combinations of all candidate bus/launcher pairs, under all potential conditions for orbit types, altitudes, inclinations, eclipses, lifetimes and reliabilities. Such a methodology, known as the "Payload Power-Mass Diagram" was developed fifteen years ago, for geosynchronous orbits, and more recently for circular low and high earth orbits. This paper extends the methodology to a more generalized approach to

Elliptical Orbits, using the specific mechanical energy concept. This method works and provides a practical mean to reducing smallsat cost.

This work applies to direct injection launches only. This restriction means that use of the bus propulsion to raise the initial orbit, which could be a useful strategy for maximising mass to orbit, is not addressed here. For the several parameters that are obtained by curve fitting, the model is restricted to the range over which the curve fit remains valid.

The configuration and technology of the solar array and batteries is also usually provided. But details required for the model such as the power subsystem mass and the specific performance of array and batteries are not. In most cases missing information can be inferred or reasonable assumptions made. This is the best a system designer can do, at the early planning phase, to narrow down and optimize a compatible bus/launch, vehicle/payload selection. At the final decision stage, the bus/launch vehicle manufacturers can best provide more accurate and final results.

Values of all the needed input parameters for a wide range of bus designs are rarely provided by the manufacturers. In many cases only partial information is available in the open literature. Typically payload mass and power capabilities are quoted together with total launch mass, sometimes in the context of a particular mission and sometimes more generally. Bus manufacturers have a need to protect their proprietary data. At the same time, they felt, fifteen years ago, that it was in their interests that potential customers for GEO spacecraft are provided with enough data to allow them initial payload/bus/launch vehicle optimisation. It is probably only a matter of time until they will provide similar bus data for non-GEO applications, now that smallsat are in greater demand.

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