QUICK AIDS TO SATELLITE MISSION PLANNING

Prof U R Rao
Project Director, ISRO Satellite Systems Projects,
Peenya, Bangalore.

Y S Rajan
Scientist, ISRO Headquarters, Bangalore (India)

ABSTRACT :

From the available data on the State-of-art of design of various satellite subsystems, some thumnrules are given. These are used in determining the power and weight available for an experimenter in a given class of satellite. Such quick formulae would be useful for the potential experimenter or mission planner to conceive various options for a given mission.

1. INTRODUCTION :

Ever since the first Sputnik was hurtled into apce in 1957 more than about 3000 satellites have been launched into apce. USSR and USA alone launch about 150 to 200 satellites every year. Other countries put together, may launch about 15 to 20 satellites every year. Also introduction of space shuttle would increase these launches further due to reduction in launch costs. This being so, it is important for various mission planners and potential experimenters to have some idea about weight and power availability in a particular satellite mission. If planning is done well in advance knowing the launch vehicle capability, it is possible for the mission planners to conceive of various options. Also this helps various experimenters who would like to place experiments in satellites being planned by others, without having to wait for initially for too many details to be provided by satellite bus planners; experimenters can give a set of prospective experiments (by cutting the coat to available cloth !)

To the authors' knowledge only one document exists that relates satellite and payload parameters to weight class (Ref.1). This document however builds up the weight requirements from the required parameters of telemetry, power consumption of payload etc. However the attempt here is to use some thumrules and derive for a

...more

given satellite weight class the available power and weight for the payload. One attempt on these lines had been made by Sibila in Reference 2 & 3. Here however author had restricted the satellite missions to the three particular launch vehicles, namely Scout, Diamant and Black Arrow. Also the author of those papers had derived only the available weight of the payload, which the present authors consider as insufficient. Because there are many useful payload missions that are lighter in weight, but consume much power. Such a high power consumption has to becessarily reflect on satellite weight in the form of additional solar arrays, power conditioning systems and batteries; thus high power consumptions of the payload leads to reduction in available payload weight on a weight limited satellite. Hence it is considered important that both power and weight for the payload are derived for each satellite weight class. The volume available for the payload can be derived from the thumbrule given for packing density; most of the conceivable payloads will follow this pattern. Satellite missions for payloads which are unusually voluminous have to be considered separately, case by case.

2. METHODOLOGY AND CALCULATIONS

By presenting such thumbrules authors do not undermine the ingenuity of individual satellite designer nor are they oblivious to various subtelities of satellite missions. For example a sunsynchrous twilight orbit can produce maximum possible power for properly maunted solar arrgy- about 3 times what is available for a non-priented spinning satellite. Similarly highly elliptic orbits would present long shadows, thus requiring a need for additional storage batteries. These subteletics are pointed out separately.

Nominal (Universal) design is done as below: (Typical calculations are given in Table-1)

STEP (i) From the weight class of the satellite using the packing density as 270 kg/M^3 derive the satellite volume.

STEP (ii) For this, given volume derive the surface area of a cube. Multiplying this surface area by a factor 8Kg/M^2 will give the weight of structural portions of the satellite. (This is meant to include adaptors, primary structures, brackets and body mounted solar arrays)

....more

The above two steps utilise some thumbrules derived by the authors after a study of some of the state-of-the-art satellites like ANS, S^3, ESRO-IV etc.

STEP (iii) At this stage one has to size the available power from the spacecraft. Power can be derived by body-mounted solar cells, non-oriented solar paddles or oriented solar paddles. Also solar paddles can be made light weight and deployable (e.g UK's X-4 satellite, CTS etc) All these modes have their own advantages and disadvantages.

Body mounted solar arrays are simplest to construct. But they have the penalty of low array efficiency because of geometrical view angles, for incident solar energy. Deployable arrays provide a means to generate more power for the satellite. But one should not overlook the fact that additional arrays will cost the satellite in weight through weight of supporting booms and additional array weight. Thus addition of deployable arrays, driving mechanisms etc really become worthwhile when the satellites are heavier than say 250 kg. This point is discussed later in this paper while discussing Fig.1.

Considering all these factors, it is desirable to calculate the solar array power for the nominal satellite design as being equivalent to the power available for spinning satellite in a circular orbit with surface area equal to the four sides of the cube defined in the step (i) & (ii). Even a three axis stabilised spacecraft will prodice roughly similar power, without specially extended arrays (which cost penalties in weight as discussed above)

Hence for estimating power output compute the surface area for four sides of the cube mentioned in step (ii). Multiply this surface area by 0.3 to get the projected area (equivalent to the case in a spinning satellite). Then using the thumbrule

$100 W/M^2$ of projected area

for the solar array power output, estimate the power output of the array. This 100 watts/$(\text{meter})^2$ of projected area takes into account currently realisable solar cell efficiencies and packing densities (allowing for some surface area for minimum space craft attitude sensors)

STEP(iv) For the solar array power output given above, multiply by 0.5 to derive the conditioned power available for the space craft. This takes into account charging requirements, eclipse durations etc in nominal cases.

...more

STEP (v) Now one has to apportion the weight and power requirements for satellite housekeeping systems like telemetry, tracking, command (TTC) attitude control systems, thermal control system etc. In most of the cases TTC weights can be roughly fixed between 8 to 10 kg with about 12 wqtts of power consumption. This weight estimates provides for redundant units. Similarly attitude control system for a three axis (body) stabilised case can be fixed around 25 kg with power consumption of . . . . Naturally very low weight class satellite (50 kg class) cannot carry such a weight and hence simple magnetic spin vector control is assumed for this class. The estimated weight for attitude control system provides for nominal hydrazine or cold gas. Any special life time or injection error correction requirements would have to provide for secondary propulsion fuel accordingly.

Weight for thermal control system also is provided for as shown in Table.1. A contingency of 10% of the total satellite is provided for to take into account balancing weights, harness, growth margin etc.

Based on the above steps weight and power available for the payload in 50 kg, 150 kg, 200 kg, and 400 kg class satellites are summarised in Table-1.

The results of table-1 is plotted in Fig-1.

3. DISCUSSION OF THE RESULTS

No attempt is made to make exact empirical formulae from Fig-1 because the curve itself can be directly used for sizing payload weight and power.

It is to be noted that the payload here is defined as the experimenters instruments, any memory devices like tape recorders and other special sybsystems that are needed for the experimenter's missions because tape recorder is not provided for in the telemetry system.

It is seen from the figure that the weight available for the payload is roughly increases linear fashion as the satellite weight increases. If the gross satellite weight is assumed as S(kg) then available payload weight W (kg) is approximately,

$$W = 0.5\,S - 30$$

...more

similarly the available power P (watts) is given approximately by

$$P = 0.113 \; S.$$

The power available for the payload starts decreasing from the linear relation when the gross satellite weight is above 300 kg. It is at this stage it is necessary to have deployable arrays to generate more power, as the weight available for the payload is substantial and one could afford to forego some of this for generating higher power. This is, of course the point for the satellite designer.

Let us briefly discuss how we can use this curve for the experimenter. As discussed earlier this curve gives the nominal value of power and weight available for the experimenter.

If the experimenter knows the satellite is a simple spin stabilised one then he can add about 10 kg more to the available payload weight than read off this curve and add about 5 watts more to the available power as attitude control system will consume less power. For all cases when the satellite weight is above 160 kg, below 100 kg three axis stabilisation cannot be assumed.

Similarly if the satellite is a twilight sunsynchronous orbit, then available payload power can be doubled than the Fig-1 indicates.

If the satellite is known to have oriented arrays similarly power available can be assumed to be doubled that indicated in the curve. But if the satellite gross weight is below 150 kg one has to reduce about 8 kg from the available payload weight for eriented array case.

Also if the experimenter needs some especially large area faee of solar cells for mounting instruments then for such an area should be multiplied by 30 W/M^2 and deducted from available payload power.

It it is known that large deployable arrays are used then in such cases adeuate increase in power available and reduction in weight available should be done. Fig-2 gives a way of combining weight and power available into a single parameter.

...more

The above remarks will be useful in adapting the curve to different missions. Fig-1 by itself would be useful for the individual exoerimenters who has a few experiments in mind.

3.1 POWER WEIGHT RELATIONSHIP :

Now let us consider the utilisation of these for planning various satellite mission. With satellite bus concepts gaining momuntum, it will be useful to consider compatibility various payloads for a particular satellite bus. In order to do this power & weight compatibility an empirical parameter combining power and weight is given below.

To generate 100 watts of array power in a spinning satellite approximately 15 kg of solar array is required and 100 watts would entail about 25 kg of power conditioning and storage system (using 4w/kg relation given before) Thus 100 watts mean about 40 kg in satellite weight. So P watts of power has a relation of 0.4 P(kg) to weight. Hence if we coin a parameter A = W + 0.4 P, then A will indicate the available power cum weight combination on the satellite bus. This is plotted in Fig-2 thus transforming Fig-1 into a single curve.

An example of use of this curve is given below. Suppose there is one payload weighing 20 kg and consuming 50 watts of power, then its A-parameter is ,

$$20 + 0.4\ (50) = 40$$

Similarly a payload weighing 35 kg and consuming 13 watts of power also has a A parameter of 40. Both this can be accomadated in a satellite of 140 kg, by having additional solar arrays for the mission carrying former payload. From Fig-1, it would look that it is necessary to have a satellite above 400 kg for the high power consuming payload. While using this A-Parameter below 150 kg one has to be careful, because the weight of boom and weight for deployment mechanisam will be non-trivial compared to payload weight. Hence it will be better to deduct about 5 kg from available payload derived through A-parameters for this region. It should also be noted that A-Parameters are additive for multi-experiment missions.

3.2

....more

3.2 EXAMPLES OF SOME TYPICAL PAYLOAD :

Examples for some application payloads (Ref 3 & 4) are given in Table-2 by showing their A - Parameters and the satellite weight class they can accomodated. A similar attempt can be done for some space science payloads also.

It can be seen from Fig-2 that all payloads shown in Table-2 except number 8, can be accomodate within 225 kg satellite class with folded deployable arrays for high power class. Readers can appreciate the value of the A-parameter especially for the theme presented in Sibila's paper (Ref 3). All payloads in Table-2 excepting 3,6 and 8 can be accomodated in a 150 kg weight class of satellites. Any redundancies needed in the in the payload system will demand heavier satellites. One may infer from these Fig-2 and Table-2 that one can build an ERTS type mission with adequate redundancy of sensor with a satellite weighing about 400 kg.

4.0 CONCLUSION

Relations between gross satellite weight and available payload weight & power have been derived. Also an empirical parameter combining power and weight for the payloads is derived with typical examples pf shpwing its use for satellite mission planning.

TABLE - 2

payload	Weight (kg)	Power watts	A-parameter
1. Automatic picture transmission V Camera and Magnetic tape recorders (10kg, 5w)	20	20	28
2. Advanced Vidicor Camera system + tape recorders (10g, 5w)	20	27	30.8
3. Return Beam Vidicon Camera + Tape recorders (20kg + 10 watts)	40	60	64
4. High Resolution infrared radiometer + tape recorder (10kg, 5 watts)	16	10	20
5. Very high Resolution radiometer + tape recorders (10 kg, 5 watts)	20	10	24
6. Multispectral scanner + tape-recorders (20kg, 10w)	70	35	84
7. Microwave radiometer + tape-recorders (10kg, 5W)	30	25	40
8. Side looking radar + Tape recorder (20kg, 10w)	200	300	320

NOTE: Tape recorders are assumed redundant and of two classes of about 10kg, 5 watts and 20 kg, 10 watt types as indicated in the table, so that for multiple missions A-parameters can be adjusted.

Proofs 1+2

~~GROSS~~ QUICK ~~QUICK~~ AIDS TO SATELLITE MISSION PLANNING

Prof. U. R. Rao
Project Director, ISRO Satellite Systems Project, Peenya, Bangalore (India)
Y. S. Rajan
Scientist, ISRO Headquarters, Bangalore (India)

Abstract: From the available data on the state-of-art of design of various satellite subsystems, some thumbrules are given. These are used in determining the power and weight available ~~to~~ for an experimenter in a given class of satellite. Such quick formulae would be useful for the potential experimenter or mission planner ~~to think of~~ to conceive ~~to~~ various options for a given mission.

1. ~~II~~ ~~III~~ <u>Introduction</u> : Even since the first Sputnik was hurtled into space in 1957 ~~about 3000~~ more than about 3000 satellites have been launched into space. USSR and USA alone launch about 150 to 200 satellites ~~per~~ every year. Other countries put together, may ~~total~~ launch about 15 to 20 ~~launches~~ satellites every year. Also introduction of space shuttle ~~may~~ would increase these launches further ~~by~~ due to reduction in launch costs. This being so, it is important for various ~~exper~~ mission planners and potential experimenters to have some idea about weight and power availability

in a particular satellite mission.
If planning is done well in advance knowing the launch vehicle capability, it is possible for the mission planners to conceive of various options. Also this helps various experimenters who would like to place ~~put~~ experiments in satellites being planned by initially

having ~~to two~~ to wait for too
Others ~~-~~ without ~~too many questions~~
many details ~~put initially~~ be provided by bus
~~posed~~ to satellite ~~design~~ planners;
experimenters can give a set of prospective experiments ~~to suit the~~ (by cutting the coat to available cloth!).

To the authors' knowledge only one document exists that relates satellite and payload parameters to weight class. (Ref. 1) This document however builds up the weight requirements from the required parameters of telemetry, power consumption of payload etc. However the attempt here is to use some ~~thumbrul~~ thumbrules and derive for a given satellite weight class ~~the payload~~ the available power and weight for the payload. ~~[illegible]~~

One attempt ~~&~~ ~~[illegible]~~ on these lines had been made by Sibila ~~by~~ in Refs. 2 & 3. Here however authors had restricted the ~~payload~~ satellite missions to the ~~set~~ ~~of~~ three particular launch vehicles,

(5)

namely, Scout, Diamant and Black Arrow. Also the ~~authors~~ ~~in~~ of those papers had derived only the available weight of the payload, which ~~we~~ the present authours consider as insufficient, because there are many useful payload missions that are lighter in weight but consume much power. Such a high power consumption has to necessarily reflect on satellite weight ~~either~~ in the ~~form~~ of additional solar arrays, ~~and~~ power conditioning systems and ~~storage syst.~~ ~~this power condition~~ batteries; Thus high power consumptions of the payload leads to reduction in available payload weight in ~~a satellite~~ a weight limited satellite. Hence it is considered important that both power and weight for the payload are derived ~~in terms of~~ for each

Satellite weight class. The Volume available for the payload can be derived from the thumbrule given for packing density; ~~assumed for any other~~ most of the conceivable payloads will follow this pattern. ~~They~~ Satellite missions for payloads which are unusually ~~voluminous~~ ~~vol~~ voluminous have to be considered separately, case by case.

~~2.0~~

~~2. Approach~~

2. ~~APPROACH~~ METHODOLOGY AND CALCULATIONS

~~[illegible]~~ By presenting

Such thumbrules authors have do not undermine the ingenuity of individual satellite designer nor are they oblivious to various subtleties of satellite missions. For example a sunsynchronous ~~orbit ca~~ ~~a~~ twilight orbit can produce maximum possible power for ~~a~~ properly ~~orientely~~ ~~oriented~~ mounted solar array ~~is~~ about 3 times what is available ~~to~~ for a non-oriented ~~cased~~ spinning ~~case~~ satellite.

Similarly highly elliptic orbits would present ~~[illegible]~~ long shadows thus ~~causing~~ ~~[illegible]~~ requiring a need for additional storage batteries. These

Subteleties are pointed out separately.

~~To start with,~~ ~~The~~ nominal (universal) ~~case can be~~ design is done as below: ~~(A typ~~ (Typical calculations are given in Table - 1)

STEP (i) From the weight class of the satellite using the ~~for~~ packing density as 270 Kg / M^3, derive the satellite volume.

STEP (ii) For this ~~the~~ given volume ~~easiest~~ derive the surface area of a cube. Multiplying this surface area by a factor 8Kg / M^2

will give the weight of structural portions of the satellite. ~~(By this it is ne~~ (This is meant to include adapters, ~~to launch~~ primary structures, brackets, and body mounted solar ~~arrays (to etc)~~ arrays)

~~(ii)~~ The above two steps utilise some thumbrules derived by the authors ~~by~~ after a study of some of the state-of-the-art satellites like ANS, S3, ESRO-IV etc.

STEP (iii) At this stage one ~~has~~ to size the available power.

from the spacecraft. Power can be derived by body-mounted solar cells, non-oriented solar paddles or oriented solar paddles. ~~Again~~ Also solar paddles can be made light weight and deployable (e.g. UK's X-4 satellite, CTS etc.) All these modes have their own advantages and disadvantages.

Body mounted ~~cells~~

Solar arrays are simplest to construct. But they have the penalty of low array efficiency because of geometrical view angles ~~from the Sun~~ for ~~the solar rays~~ incident solar energy. Deployable arrays provide a ~~meed~~ means to ~~true~~ generate more power for the satellite. But one should not overlook the fact that ~~these~~ additional arrays will cost the satellite in ~~terms~~ weight through weight of supporting booms and ~~adds~~ additional array weight. Thus ~~these~~ addition of deployable arrays, driving mechanisms etc really become worthwhile when the satellites are heavier than say 250 kg. This point is discussed later in this paper while discussing Fig 1.

Considering all these factors, it is desirable to calculate ~~consider~~ the solar array power for the nominal satellite design as being ~~derived for a~~ equivalent to the power available for spinning satellite in a circular orbit ~~for eq~~ with surface area equal to the four sides ~~projection~~ ~~derived in~~ ~~steps (i) and (ii).~~ of the cube defined in the step (i) and (ii). Even a three axis stabilised spacecraft will produce roughly similar power, ~~with~~ without specially extended arrays (which cost ~~further~~ penalties in weight as discussed above).

Hence for ~~calculating~~ estimating power output ~~calculate~~ compute the surface area for four sides of the cube mentioned in Step (ii) Σ. Multiply this surface area by 0·3 to get the projected area (equivalent to the case in a spinning satellite) Then using the thumb rule

100 W / m^2 of projected area

for the solar array power out-put, estimate the power output of the array.

~~This 100W/m² ... for the~~

This 100 watts/(meter)² of projected area takes into account currently realisable solar cell efficiencies and packing densities (allowing for some surface area for minimum spacecraft attitude sensors)

<u>Step (iv)</u> For the solar array power output given above, multiply by 0·5 to derive the conditioned power available for the spacecraft. This takes into account charging requirements, ~~et~~ eclipse~~s~~ durations etc in nominal cases.

Step (v) Now ~~comes the~~ one ~~Questi~~ ~~B~~ has to apportion the weight and power requirements for ~~Spacecraft~~ Satellite housekeeping systems like telemetry, tracking, ~~and~~ command (TTC) attitude control systems, thermal control system etc. In most of the cases TTC weights can be roughly fixed ~~at~~ ~~IOT~~ between 8 to 10 kg with about 12 Watts of power consumption. This weight estimates provides for redundant units. Similarly attitude control system for a three axis (body) stabilised case can be fixed around 25 kg with power consumption of - - - - -. Naturally very low weight class satellite (50 kg class) cannot carry such a weight and hence simple magnetic spin vector control is assumed for this class. The estimated weight for attitude control systems provides for nominal hydrazine or nitrogen cold gas. Any special life time or injection error correction requirements would have to provide for secondary propulsion fuel accordingly.

Weight for thermal control system also is provided for as ~~shown~~ Table-1. ~~2-2a~~ A contingency of 10% of the total satellite is provided for to take into account balancing weights, harness, growth margin etc.

Based on the above steps ~~wei~~ ~~payload~~ weights and power available for the payload in 50 kg, ~~100~~ 150 kg, 200 kg, and 400 kg class satellites are summarised in Table-1.

The results of Table-1 is plotted in Fig-1.

3. DISCUSSION OF THE RESULTS

No attempt is made to

make exact empirical formulae from Fig-1 because ~~of~~ the curve itself can be directly used for sizing payload weight & power.

¶ It is to be noted that the payload here is defined as the experimenter's instruments, ~~and~~ any memory devices like tape recorders, and other special subsystems ~~sensors~~ that are needed for the experimenter's missions because tape recorder is not provided for in the telemetry system.

It is seen from the figure that the weight available for the payload ~~rough~~ is roughly increases ~~the~~ linear fashion as the satellite weight increases. If the gross satellite weight is assumed as S (kg) then available payload weight W_p (kg) is approximately,

$$W_p = 0.5\,S - 30$$

~~$S =$~~

~~W_p (kg) =~~

Similarly ~~if~~ the available power ~~is~~ P ~~(kg)~~ (watts) is given approximately by

$$P = 0.113\,S.$$

The power ~~The~~ available for the payload starts decreasing from the linear relation when the gross satellite weight ~~goes~~ is above 300 kg. ~~to~~ It is at this stage it is necessary ~~imperative~~ to have deployable arrays to generate more power, as the ~~payload~~ weight available for the payload is substantial and one ~~was~~ could afford to forgo ~~those~~ some of this for generating higher power. This is, of course, the point for the satellite designer.

Let us briefly discuss how we can use this curve for the experimenter. ~~If~~ As discussed ~~befor~~ earlier this curve gives the nominal value of power & weight available for the experimenter. ¶ If the experimenter knows the satellite is a simple spin stabilised one then he can add about 10 kg more to the available payload weight than read off this curve and add about 5 watts ~~S~~ more to the available power as attitude control system will consume less power. ~~(except for 50kg class where no~~ for all

(A) Also if the experimenter needs some ~~clear~~ especially large area free for mounting ~~his~~ instruments of solar cells then for such an area ~~he can~~ ~~use about~~ should be multiplied by

30 W/m^2 and

deducted from available payload power.

If it is known that large deployable arrays are used then in such cases adequate ~~modification~~ increase ~~of~~ in power available and reduction in weight available should be done. Fig-2 gives a way of combining weight and power available into ~~one~~ a single parameter.

Go to (20)

(19)

cases when the satellite weight is above 100 kg. Below 100 kg ~~xxx~~ three axis stabilisation cannot be assumed.

¶ Similarly if the satellite is ~~is~~ a ~~xxx~~ ~~sy~~ twilight sun synchronous orbit, ~~x~~ then available payload power can be doubled than the Fig-1 indicates. ~~For a~~

~~Similarly~~ ~~x~~

¶ ~~For ori~~

¶ If the satellite is known to have oriented arrays similarly power available can be assumed to be double that indicated in the curve. But if ~~below 100 kg gross weight satelli~~ the satellite gross weight is below 150 kg one has to reduce about 8 kg from the available payload weight for oriented array case.

→ (A)

The above remarks will ~~give some~~ be useful is adapting the curve to different missions. Fig-1 by itself would be ~~quite~~ ~~[illegible]~~ useful for the individual experimenters who has a few experiments in mind.

3.1 Power-weight relationship

Now let us consider the utilisation of these for ~~a~~ ~~satellite~~ ~~[illegible]~~ planning various satellite mission. With satellite bus concepts gaining momentum, it will be useful to consider compatibility various payloads for a particular satellite bus. ~~This~~ In order to do this power & weight compatibility an empirical parameter combining power and weight is given below.

To generate 100 watts of array power in a spinning satellite approximately ~~6 kg~~ 15 kg of solar array is required and 100 watts would entail about 25 kg of power conditioning ~~and~~ storage system [using 4 W/kg relation ~~formula~~ given before.] Thus 100 watts mean about 40 kg in satellite weight. ~~Thus~~ So P watts of power has a relation of 0.4 P (kg) to weight. Hence if we coin a parameter A = W + 0.4 P, ~~as~~ then A will indicate the available power cum - weight combination ~~of~~ ~~for~~

on the satellite ~~B~~ bus. ~~Thus~~ ~~for a payload consuming~~ ~~power 20 (watts) and weighing~~

~~To consider~~ a This is plotted in Fig - 2 thus transforming Fig - 1 into a single curve.

An example of use of this curve is given below. Suppose there is one payload weighing 20 kg and consuming 50 watts of power, then its A-parameter is ~~60 kg~~.

20 + 0.4 (50) = 40.

Similarly a payload weighing 35 kg and consuming 13 watts of power also has a A parameter of 40. Both this can be accomadated in a satellite of ~~120~~ ~~150 kg~~ 140 Kg ~~or beyond~~. by ~~properly~~ having additional solar arrays for ~~Case 1.~~ the mission carrying former payload.

From Fig - 1, it would look that it is ~~very~~ necessary to have a satellite above 400 kg for the high power consuming payload. ~~A word of caution is that~~ while using this A - parameter ~~for~~ below 150 Kg one has to be careful ~~due to the~~ because the weight of boom ~~and driving~~ and ~~etc~~ weight for deployment mechanism will be non-trivial compared to payload weight.

Hence it will be better to deduct about 5 kg from available payload derived through A - parameters for this region. It should also be noted that A parameters are <u>additive</u> for multi-experiment missions.

3.2 Examples of some typical payload

Examples ~~shown~~ for some application payloads (Ref 38) are given in Table 2 by showing their A parameters and the satellite weight class they can accomodate. A similar attempt can be done for some space science payloads also.

~~B~~ Tab

Table - 2

Payload	Weight (Kg)	Power watts	A-parameter
1) Automatic Picture Transmission ~~APT~~ TV Camera and Magnetic tape recorders (10 kg, 5W)	~~18~~ 20	20	28
~~2) RBV~~			
2) Advanced Vidicon Camera System + tape recorders (10g, 5W)	20	27	30.8
3) Return Beam Vidicon Camera + Tape recorders (20kg + 10 watts)	40 ~~kg~~	60	64
4) High Resolution infra red radiometer + tape recorders (10 kg, 5 watts)	16 ~~kg~~	10	20
5) Very high Resolution radiometer + tape recorders (10 kg, 5 watts)	20	10	24

Payl. -	-		
6) Multispectral Scanner + tape recorders (20 kg, 10 W)	70	35	84
7) Microwave radiometer + tape recorders (10 kg, 5 W)	30	25	40
8) Side looking radar + tape recorder (20 kg, 10 W)	200	300	320

Note: Tape recorders are assumed redundant and of two classes of about 10 kg, 5 watts and 20 kg, 10 watt types as indicated in the table, so that → (B)

25

(B) { for multiple missions A-parameter can be adjusted.

25

It can be seen from ~~figure~~ Fig-2 that all payloads shown in Table-2 except number 8, can be accomodated within 225 Kg satellite class with folded deployable arrays for high power class.

~~&~~ Readers can appreciate the value of the A-parameter especially for the theme presented in Sibila's paper (Ref. 3). All payloads in Table-2 except ~~for~~ nos 3, 6 and 8 can be accomodated in a 150 Kg weight class of satellites. Any ~~set~~ redundancies needed in the payload system will demand heavier satellites. One may infer from these ~~curves and~~ Fig-2 and Table-2 that one can build an ERTS type mission with adequate redundancy of sensor with a satellite weighing about 400 kg.

4.0 CONCLUSION

Relations between gross satellite weight and available payload weight & power have been derived. Also an empirical parameter combining power and weight for the payloads is derived with typical examples of showing its use for ~~satellite~~ satellite mission planning.