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Mass Comparisons of Electric Propulsion Systems for NSSK of Geosynchronous Spacecraft

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SUMMARY

A model was developed and exercised to allow wet mass comparisons of three-axis stabilized communications satellites delivered to geosynchronous transfer orbit. The mass benefits of using advanced chemical propulsion for apogee injection and north-south stationkeeping (NSSK) functions or electric propulsion (hydrazine arcjets and xenon ion thrusters) for NSSK functions are documented. A large derated ion thruster is proposed which minimizes thruster lifetime concerns and qualification test times when compared to those of smaller ion thrusters planned for NSSK applications. The mass benefits, which depend on the spacecraft mass and mission duration, increase dramatically with arcjet specific impulse in the 500 to 600 s range, but are nearly constant for the derated ion thruster operated in the 2300 to 3000 s range. For a given mission, the mass benefits with an ion system are typically double those of the arcjet system; however, the total thrusting time with arcjets is less than one-third that with ion thrusters for the same thruster power. The mass benefits may permit increases in revenue producing payload or reduce launch costs by allowing a move to a smaller launch vehicle.

INTRODUCTION

Electric Propulsion Systems (EPS's) are attractive candidates to perform north-south stationkeeping (NSSK) functions on geosynchronous spacecraft (refs. 1 to 4). The primary benefit of using electric propulsion is a significant reduction of NSSK propellant from that presently required. Electric propulsion accomplishes this propellant mass reduction by increasing the propellant exhaust velocity, or specific impulse, over that of present chemical thrusters (ref. 4). This gross mass benefit, less the EPS mass, may then be traded for additional revenue producing payload, additional propellant to extend the mission, or a move to a smaller, less costly, launch vehicle.

Electric thrusters such as resistojets (refs. 5 and 6), arcjets (refs. 7 and 8), Hall-current devices (refs. 9 and 10), and ion thrusters (refs. 8 and 11 to 15) are under development. Propulsion systems with these different electric thrusters have widely varying performance, mass properties, and maturity levels; therefore, they can yield vastly different benefits and risks to spacecraft using them. As electric thrusters mature, their performance and mass parameters become better defined as does the ability to compare their relative benefits. This paper compares the benefits of applying two of the more mature EPS technologies as well as advanced chemical propulsion to the task of providing NSSK to communications satellites in geosynchronous orbit. A thorough mass breakdown of an entire specific communications satellite is beyond the scope of this paper; however, detailed descriptions of two EPS technologies are presented.

Systems employing hydrazine arcjets or xenon ion thrusters are selected for mass budget analyses performed herein. Hydrazine arcjets are selected for this comparison because they have been baselined for NSSK on General Electric's 7000 series communications satellite (refs. 7 and 16). Xenon ion thrusters are also selected for comparison because of the results of prior studies and world-wide interest (refs. 11 to 15).

In particular, a new low risk approach is in effect at the National Aeronautics and Space Administration's Lewis Research Center (NASA Lewis) to assist in the implementation of ion thrusters for NASA's space propulsion needs. For NSSK applications, this approach involves using a throttled or "derated" ion thruster system which greatly mitigates all known ion thruster life-limiting mechanisms. It is proposed that a large 30 cm diameter ion thruster, which has demonstrated operation beyond 5 kW, be used at a small fraction of this power (ref. 17). In addition to significant lifetime gains, some performance benefits accrue, allowing shorter total thrusting times than those of thrusters with diameters of 10 to 13 cm (refs. 13 to 16). The penalties of this derated thruster philosophy are a heavier and more voluminous EPS.

The mass benefits obtained by using EPS with various electric thrusters are computed by comparing the wet spacecraft mass injected into a geosynchronous transfer orbit (GTO) for an all chemically propelled spacecraft with those utilizing EPS for NSSK. This general methodology, which has been used before and most recently in reference 18, is presented in appendix A. All symbols are defined in appendix B. A listing of the computer program used is given in appendix C. Specific values for each variable are given in the PARAMETER VALUES section. Then, the results obtained from mission, spacecraft, and chemical propulsion parameter variations are examined.

PARAMETER VALUES

This section describes and quantifies the model inputs for Mission and Spacecraft, Apogee Propulsion, NSSK Chemical Propulsion and, NSSK Electric Propulsion parameters. State of the art values of parameters in these four groups are kept in separate files and designated as default values. When the program is run, a screen for each group appears with the default values. Any or all of the parameter default values may be changed before computations are conducted. Initial default values for each parameter are discussed below and quantified for an arcjet EPS. Then, those initial default values for an ion thruster EPS, which are different from those of the arcjet EPS, are described.

Mission and Spacecraft

The "mission" in this study is north-south stationkeeping (NSSK) of a three-axis stabilized communications satellite in geosynchronous orbit (GEO) with a chemical propulsion system or an electric propulsion system (EPS). Spacecraft such as the GE 7000 series and Intelsat VII are typical geosynchronous satellites of interest (refs. 16 and 18). Baseline and growth versions of these spacecraft have dry masses between 1400 and 2000 kg. Typical launch masses for these spacecraft lie between 3000 and 4000 kg; therefore, they could use members of the Atlas II or Ariane 4 families of launch vehicles. The launch sites chosen are the Eastern Test Range (ETR) in Florida, USA and Kourou, Fr. Guiana, which require velocity increments of approximately 1785 m/s (ref. 19) and 1514 m/s (ref. 4), respectively, for the apogee injection maneuver. An average annual velocity increment required for NSSK is 46 m/s (ref. 4). A baseline mission lifetime of 15 years is selected although it is varied from 10 to 20 years in the analysis. It is assumed that the energy to perform a NSSK maneuver with an EPS will always be provided by the on-board eclipse batteries (ref. 20), rather than the spacecraft solar arrays. To minimize the battery depth of discharge, any excess solar array power will probably be used to support the EPS maneuver (ref. 18).

The total energy of the eclipse batteries is assumed to be in the range of 4.8 to 7.2 kW-hr. Only a fraction of this energy will be available for an EPS maneuver. Table I summarizes the default values for the Mission and Spacecraft parameters.

Apogee Propulsion

Four hundred-fifty Newton (100 lbf) class, bipropellant, apogee kick motors integral to the spacecraft were assumed to provide the velocity increment required by the orbit transfer maneuver. Nitrogen tetroxide (NTO) is commonly used with either monomethyl-hydrazine (MMH) (ref. 4) or anhydrous hydrazine (refs. 21 and 22) in a variety of engine geometries. State of the art (SOA) default values of specific impulse for these two propellants are assumed to be 311 and 315 s, respectively. With material changes to allow hotter combustion chambers and increased area ratio nozzles (ref. 23), these values should increase. A propellant reserve fraction of 4.2 percent is believed to be typical for bipropellants and is assumed for the mass model.

Chemical NSSK Propulsion

Twenty-two Newton (5 lbf) class bipropellant rockets are assumed to be used to provide NSSK corrections. The initial specific impulse default value is 285 s (ref. 4). Again, the bipropellant reserve fraction is assumed to be 4.2 percent. For non-NSSK propulsion functions, such as east-west stationkeeping and attitude control, the non-NSSK propellant is assumed to be the sum of a time independent component and a component which varies at a rate of 4 kg/year of mission life. Values are estimated from prior spacecraft (ref. 4). The sum of the two components is then normalized to a spacecraft dry mass of 1640 kg, the example of reference 18. Table II lists the parameter default values used for bipropellant apogee, attitude control, and stationkeeping engines. Values for advanced technologies are discussed later.

Electric NSSK Propulsion-Arcjet

Recent governmental and industrial efforts have brought hydrazine arcjets to a state of flight readiness (refs. 7, 8, and 16).

Table III presents demonstrated (refs. 7, 8, and 24) and extrapolated (ref. 25) operating points for hydrazine and nitrogen-hydrogen gas mixtures simulating ideal hydrazine decomposition products. For the thruster input powers shown, the specific impulse and thrust values given are assumed to be mission averaged values. Variations on the order of ±5 percent from these average values would be expected as the propellant supply pressure decreases with propellant usage.

Mass models of electric propulsion systems were previously developed for high power, primary propulsion ion thruster systems (refs. 26 and 27). Where appropriate the methodology of reference 26 is followed in this study, as evidenced by the division of the EPS into a thrust module and an interface module as shown in figure 1. The thrust module includes the thrusters, gimbals, propellant distribution, and structure for these elements. The interface module contains the power processors, wiring, thermal control for the power processor and interface module waste heat, an EPS controller and housekeeping power supplies, propellant storage and control, and structure between the thrust module and interface module elements and the spacecraft bus. The division of components into modules is primarily for mass accounting purposes. For these missions, the mass model is, at this time, believed to be representative of fully integrated architectures.

Table IV lists mass model parameters used for arcjet systems and their default values. Four electric thrusters are employed to provide NSSK of three-axis stabilized spacecraft. One pair of thrusters are fired at a time such that the resultant thrust is parallel to the North or South axis and through the center of mass of the spacecraft. The other pair may be used or held for redundancy. The thruster mass (MTH) is 1.0 kg (ref. 7). In general, the need for thruster gimbals is determined by the placement of the thrusters and several strategies employed to compensate for induced disturbances. Thrusters may be placed on either the east and west or north and/or south faces of the spacecraft. Gimballed NSSK thrusters are typically vectored approximately ±10° about one or two axes. A final design will certainly be spacecraft configuration specific and based upon tradeoffs between propulsion module mass and attitude control complexity and/or propellant mass. When used, the gimbal mass for this study is assumed to be equal to 34 percent of the thruster mass (ref. 26). The total thruster and gimbal mass (MTGT) would then be 5.36 times that of a single thruster when gimbals are used or four times when no gimbals are used.

A propellant distribution mass (MPD) of 1 kg per thruster is included for the hydrazine arcjet system on spacecraft using MMH and NTO for non-NSSK functions. When anhydrous hydrazine (arcjet propellant) and NTO are available for non-NSSK functions, no mass penalty is assessed. Significant support structure is required between the thrust module elements and the spacecraft interface components and is assumed to be 31 percent of the thrust module component masses (ref. 26).

Each thruster requires a power processor to convert the spacecraft bus voltage to that required by the arcjet. Arcjet power processor mass has been modeled as that of a high power ion thruster discharge supply (ref. 26) or, more recently (ref. 28), as a unit scaled to a 10 kW level based on a 1.4 kW flight type power processor. Both equations yield power processor specific mass (ALPPU) values which decrease with input power. Reference 7 has demonstrated a 1.8 kW flight design power processor with a specific mass of 2.33 kg/kW which is less than that predicted by either of the prior methods; therefore, over the small arcjet power range of table III, the power processor specific mass is assumed to be constant at a value of 2.33 kg/kW. Arcjet power processor efficiencies (ETAPPU) of about 90 percent have been demonstrated (ref. 7) and are assumed in this study. The power cable assembly required for each low power arcjet has a mass (MCAB) of 0.8 kg (ref. 7). The arcjet EPS controller and housekeeping power supplies are estimated (ref. 26) to have a mass (MIFM) of 2.2 kg, require 50 W (PIFM), and be 90 percent efficient (ETAIFM). Heat pipe radiators are required for thermal control of heat generated by two operating power conditioners and the interface module. The thermal control specific mass (ALTC) is assumed to be 31 kg/kW, as suggested in reference 26. It is believed that this value is near that proposed for the next generation of communications satellites.

The mass of the hydrazine propellant tank and propellant control (MPTK) is assumed to be the product of a propellant tank fraction (0.07) and the NSSK propellant. An initial value of propellant mass is provided, as mentioned earlier, because the program iteratively computes the EPS propellant tankage mass which is a function of the effective specific impulse. The initial value of propellant mass is assumed to be 250 kg. An interface module structure mass (MIFS) is then computed as a fraction (0.04) of the sum of the propulsion module elements and a mass (MEPAD) to account for any elements required by the use of EPS. A value of 10 kg is assumed for this added mass for EPS which includes extra sensors, eclipse battery charge circuitry, and heaters to maintain thermal control of unused power processors during eclipse periods (ref. 18). A contingency mass (MCON), which is 20 percent of the sum of the propulsion module and interface module structure masses, is added to that sum to give the electric propulsion system dry mass (MEPD). This contingency mass fraction is the same for both arcjet and ion thruster systems.

Regardless on which spacecraft face electric NSSK thrusters are mounted, they are usually located on the spacecraft bus exterior and pointed primarily in the north and/or south directions. To minimize impingement of the arcjet exhaust plume on the solar array panels, which also extend in the north and south directions, the arcjets are canted away from the NS axis. Based on plume studies (refs. 29 and 30), a thruster cant angle (PH1) of 17° is assumed for arcjets. Thus, the thrust and specific impulse are reduced by the cosine of 17°. To calculate the total NSSK propellant mass, a 6 percent propellant reserve fraction (PRF) is assumed for hydrazine and used to increase the propellant mass value computed with the rocket equation. This factor combines the reserve and residual propellant in the propellant tank.

The calculation of thrusting time per firing (TTIME) requires the number of permissible thrusting days per year (D) and the number of firing(s) per day(s) (N). Values for the number of thrusting days per year used in this study are 365 or 275, which excludes the two 45 day eclipse periods. The thruster firing frequency may vary considerably from twice a day (around each nodal crossing), to once a day or once every few days depending on the parameter values in equation A4.

Electric NSSK Propulsion - Ion

Research and flight programs have demonstrated the flexibility of ion propulsion to perform primary and auxiliary propulsion functions (ref. 31). Auxiliary propulsion xenon ion thrusters are being developed worldwide. Table V lists a nominal operating condition for each of the most mature known xenon ion thrusters and the source of the information. These thrusters are briefly described below. Other ion thruster default values used in this study are presented in table VI. Only the values which are different from those used for arcjets (table IV) are discussed here. If no reference is given, then the value is an estimated one.

Derated Thruster. - The laboratory model derated thruster presented in reference 17 has a mass of 10.7 kg. Based on preliminary testing, it is anticipated that this mass can be reduced to 7.0 kg by thruster structural redesign. For this thruster, the propellant distribution includes one low pressure propellant line flowing to a pair of latching values at the thruster as shown in figure 2 (ref. 12). From there, it splits into three lines, each ending at a flow limiting impedance. The mass of propellant distribution hardware (MPD) is estimated to be 7.3 kg based on the component masses presented in reference 27.

Power processor specific mass (ALPPU), as a function of input power may be estimated from figure 3(a). At a low power level (0.71 kW), the data of reference 13 are used. At a power level of 1.4 kW, the data of reference 32 are plotted. For ion thrusters operating at the 5 kW level, the mass models of reference 28 are used to combine the technology demonstrated for a 5 kW arcjet power supply (ref. 33) with that of reference 32. Figure 3(a) shows that as the power processor input power is increased, the specific mass decreases asymptotically with increasing power. This trend has also been observed for power processors developed for mercury ion thrusters (ref. 31). Figure 3(a) and table VII also show the power processor specific masses assumed for the derated thruster. Figure 3(b) and table VII show the values of power processor efficiency used in this study. It is assumed that the power processor efficiency (ETAPPU) increases asymptotically with increasing power to a value of about 0.94. This tendency is based on both demonstrated (refs. 13, and 32) and estimated (ref. 31) results.

Table VII summarizes five operating conditions demonstrated by the derated xenon ion thruster (ref. 17) and used as mass model input parameter variations. Also shown are corresponding power processor specific mass and efficiency values assumed for the model.

The propellant tankage and control fraction (PTF) is estimated to be 20 percent of the propellant mass. It includes a 14 percent fraction of the propellant mass for the tankage (refs. 26 and 27) and approximately 3.4 kg for propellant control (ref. 27) including valves, regulators, and filters. For this mission, this latter mass may be approximated as 6 percent of the NSSK propellant mass. For the large systems of references 26 and 27, this mass would be insignificant compared to the tankage. The initial value of the propellant mass (MPSK) used for ion thruster calculations is 50 kg.

For the more complex ion propulsion systems, the EPS controller and housekeeping power supply mass (MIFM) is estimated (ref. 26) to be 6.2 kg. The cant angle PH1 for the larger derated ion thruster is assumed to be 30°. The ion thruster cant angle is larger than that of the arcjet because particles emitted from the ion thruster typically have energies about 1000 times those from the arcjet and are capable of doing considerable sputtering damage. Energetic ion densities at angles greater than 30° are believed to be sufficiently low to preclude solar array damage.

XIPS thrusters. - Hughes Research Laboratories (HRL) developed and endurance tested a working model Xenon Ion Propulsion Subsystem (XIPS) for NSSK of large geosynchronous communications satellites (refs. 12 and 32). The ion thrusters for that subsystem are 25 cm in diameter. HRL has also designed a propulsion system with similar technology using a 13 cm diameter thruster (ref. 34). The operating characteristics of these two ion thrusters are presented in table V.

IES, IPS thruster. - Expanding on the ion thruster results from Japan's third Engineering Test Satellite (ETS-III) (ref. 35), their National Space Development Agency (NASDA) has developed a 12 cm diameter xenon Ion Engine System (IES) with the cooperation of Mitsubishi Electric Corporation and Toshiba (ref. 13). The IES is scheduled to perform the NSSK functions for ETS-VI (ref. 13). Properties of IES are also given in table V. This ion thruster is also proposed for the Ion Propulsion System (IPS) of reference 18.

RIT thrusters. - The European Space Agency (ESA) has sponsored electric propulsion development resulting in two different xenon ion propulsion systems (refs. 14 and 15). Both systems use ion thrusters which are 10 cm in diameter. One, known as RITA, for Radio Frequency Ion Thruster Assembly, has been selected as a flight experiment on the European Retrievable Carrier (EURECA-I) (ref. 36) and proposed as an operational system for ESA's advanced communications technology satellite (SAT-2) (ref. 14). The RIT-10 ion thruster is manufactured by Messerschmitt Bolkow Blohm and is unique in that it employs radio frequency power to ionize the xenon. Additional RIT-10 properties are given in table V.

UK-10 thruster. - ESA's other electric propulsion system is designated the UK-10 Ion Propulsion System (ref. 15) and has also been proposed for SAT-2 (ref. 37). Here, the xenon ions are generated conventionally with a direct current discharge. Some of the operational characteristics of this thruster are presented in table V.

RESULTS AND DISCUSSION

The mass model parameter values presented in the previous section were varied using the computer program listed in appendix C. An example of the mass model input and output values is shown in table VIII. The primary outputs of the model are the initial spacecraft masses in geosynchronous transfer orbit (GTO) with and without an electric propulsion system (EPS) option. The difference between these two masses quantifies the mass benefit of using EPS, and the magnitude of the spacecraft mass in GTO determines the launch vehicle requirements and subsequent differences in the launch costs. The results of input parameter variations from the default values and their impact on launch vehicle requirements are presented below.

Launch Vehicle Capability

Throughout the history of communication satellites, the configurational direction has been toward more massive spacecraft, with a wider range of payloads, increased total power levels, and longer on-orbit lifetimes (refs. 4 and 18). One consequence of these trends is a greater demand on the performance of today's midrange launch vehicles, such as those of the Atlas and Ariane families. Table IX lists the approximate present and near-term mass capabilities delivered to GTO for several launch vehicles (refs. 38 to 41). An assumed mass of 60 kg for a spacecraft adapter has been subtracted from the advertised launch vehicle performance values. Minor improvements in these values are continuously sought in order to capture payload growth opportunities and are usually incorporated in block changes yielding 100 to 200 kg increases in the mass delivered to GTO. Examples of this type of growth are shown in the differences between the capabilities of the Atlas II and IIA or Ariane 44P and 42L launch vehicles. Consequently, the manufacturers of satellites have options in sizing their payloads to fit a particular launch vehicle. These mass options are valued at approximately $30 000/kg (ref. 18) of mass in GTO and may be traded for anticipated revenue. The fact that different launch vehicle providers charge users differently (either a fixed price per vehicle or a price per kg) also complicates the spacecraft design process. When the model results are presented, an appropriate GTO mass capability for a given launch vehicle is also shown to indicate a possible move to a smaller launch vehicle.

Mass Model Test Cases

Test cases for the mass model were examined using estimated Ion Propulsion System (IPS) parameters shown in table V with the Intelsat VII growth version spacecraft (1640 kg dry) of reference 18. Table X compares the results of reference 18 with outputs from this model. Minor differences between the propellant budget results of reference 18 and this mass model are caused by several factors. First, the non-NSSK propellant in this model is a function of the spacecraft mass and mission duration. In addition, the annual NSSK velocity increment for reference 18 is 48.6 m/s (the worst case for a 15 year mission) whereas this model uses an average value of 46 m/s. Also, the chemical propulsion NSSK specific impulse used in reference 18 is believed to have been slightly higher than that which this model assumes. These input parameter differences lead to chemical and electric propulsion wet spacecraft masses at GEO which are -34 kg and +10 kg, respectively, different from those of the reference case. This 44 kg difference is compounded by an additional 30 kg difference in apogee injection propellant. Minor differences between the assumed values of apogee injection velocity increment, specific impulse, and propellant reserve fraction account for another 11 kg of difference between the two cases. The combined effect of these differences show up as an 85 kg variation in the mass benefit or about 2 percent of the mass in GTO. Table X also shows that when only the IPS thruster specific impulse, power, thrust, and mass and power processor specific mass and efficiency are used as model inputs, with the EPS model default values for the other inputs, the difference in the mass benefit with respect to the reference case, decreases to 58 kg. The EPS dry mass for this third case is 17 kg lighter than the reference case primarily because the EPS model assumes passive NSSK propellant flow control and a smaller gimbal mass fraction.

As discussed in reference 18 and shown herein, the use of an ion thruster EPS can allow a mass benefit of about 500 kg to be realized. This benefit may translate into a $15M savings in launch costs or possibly a move to a smaller launch vehicle.

Input Parameter Variations

In order to evaluate possible variations in spacecraft bus geometry and payload, the mission and spacecraft parameters were varied. The impacts of using advanced chemical propulsion technology and EPS with arcjets and ion thrusters at different operating conditions are also examined.

Years on orbit (Y). - While holding the dry spacecraft mass and the propulsion technologies fixed, the mission duration was varied from 10 to 20 years. The results are shown in figure 4. The mass in GTO increases nearly linearly with mission duration, as expected, but at a slower rate for NSSK propulsion with increased specific impulse. Thus, mass benefits also increase with mission duration and at a faster rate for the propulsion system with higher specific impulse. For both arcjet and ion thruster systems, the required total thrusting times increase slightly more than a factor of 2 from 600 and 2900 hr, respectively, as mission duration increases from 10 to 20 years. This faster than linear effect occurs because the non-NSSK on-orbit propellant also increases with mission duration, raising the initial mass in GEO and causing the thrusting time per firing to increase slightly with mission duration. A more refined mass model would account for a spacecraft mass which decreases with mission duration as on-orbit propellant is consumed.

Launch site. - Figure 4 is generated from cases where the launch site is Kourou and the velocity increment required for apogee injection is 1514 m/s. For launches from ETR, the required velocity increment is 1785 m/s due to the greater change in orbital inclination. Figure 5 shows the mass in GTO and mass benefits with EPS mission duration as functions of mission duration for the same conditions as figure 4 with the exception of the launch site. Mass differences result only from the added apogee injection propellant mass for ETR. Once GEO is achieved, the balance of the missions are identical. The net result is that the mass in GTO and the mass benefits are always about 10 percent greater for an ETR launch than one from Kourou, for fixed apogee and NSSK propulsion assumptions.

Spacecraft dry mass (MSC). - The spacecraft dry mass was varied from 1400 to 2000 kg for spacecraft using state-of-the-art (SOA) chemical, arcjet and ion propulsion systems. Figure 6 shows the mass in GTO and mass benefits for these three cases as functions of the dry spacecraft mass. The mass in GTO and the mass benefit both increase linearly with spacecraft dry mass. This means that any of the results presented herein for one spacecraft dry mass may be scaled to other spacecraft dry mass values. The spacecraft mass in GTO decreases as the NSSK propulsion system specific impulse increases mainly due to the reduction in NSSK propellant. This trend is also reflected in the mass benefits, which are always greater for the higher specific impulse ion thruster. Additional technological improvements in launch vehicle capability or spacecraft propulsion would permit an Atlas IIAS to capture a greater fraction of the payloads shown for an ETR launch. The total thrusting time required by EPS increases nearly directly with increases in the dry spacecraft mass for given thruster conditions.

Battery energy rating (kW-hr). - When the eclipse battery energy rating is increased from 4.8 to 6.0 to 7.2 kW-hr for a fixed dry spacecraft mass, only the maximum battery depth of discharge (MAXDOD) changes. The maximum depth of discharge is inversely proportional to the battery energy rating. For this model, it is assumed that the NSSK maneuver with EPS would always be accomplished with battery power alone. However, excess solar array capability could be used to reduce the use of the eclipse batteries (ref. 18).

The case of increased battery energy for a constant dry spacecraft mass may occur if the communications technology is improved to yield a reduced payload mass. Then, additional communications capability, solar array, and batteries could be added and the depth of discharge for EPS maneuvers would decrease. As an option, increases in battery energy could permit less frequent EPS usage and allow a decrease in the number of battery cycles rather than a decrease in the battery depth of discharge.

The upper limits of allowable battery depth of discharge and number of battery cycles have been increasing with time (refs. 4 and 18) as extended testing of nickel-hydrogen storage battery cells continues at depth-of-discharge values up to 0.80 for more cycles than are required by an ion thruster EPS performing a 20 year NSSK mission (refs. 42 to 44). The more demanding battery requirements of low Earth-orbit missions has spurred battery research to produce technological advances which easily satisfy the needs of EPS propelled communications satellites at GEO.

Thrusting days per year (D). - This study assumes that EPS thrusting would be allowed every day of the year, including the two 45 day eclipse periods. This may not be realistic because there could be days when spacecraft functions, such as orbit determination, may require undisturbed periods in excess of 24 hr. In the past, certain spacecraft configurations also precluded propulsion power loads on the storage batteries during eclipse periods (ref. 4). Thus, the allowable number of thrusting days per year may be less than 365 and may be on the order of 275 days. The thrusting time and battery depth of discharge per firing were found to be nearly inversely proportional to the number of allowable thrusting days per year. There were small additional increases due to the off-nodal thrusting inefficiency mentioned earlier. The number of battery cycles varied proportionally with the allowable number of thrusting days per year while all other outputs remained unchanged.

Firing(s) per day(s) (N). - In this study, the baseline values of firing frequency for arcjet and ion thrusters are once per week and daily, respectively. This typically leads to values of thrusting time of about 1 hr and battery depth of discharge values of 0.3 to 0.6. However, under certain spacecraft and/or thruster operating conditions, thrusting times may be less than half or more than double this value. When these events occur, the firing frequency may be, respectively, lowered or raised accordingly to optimize the thrusting time per firing without exceeding some desired maximum battery depth of discharge. The battery depth of discharge was found to vary inversely with the firing frequency. The mass in GTO and mass benefits were unaffected by changes in firing frequency.

Apogee engine specific impulse (ISPAI). - A new class of storable bipropellant engines for spacecraft propulsion have demonstrated performance levels which are significantly greater than SOA (ref. 23). One technology enabling the improvements is the use of oxidation resistant materials. An iridium-lined rhenium combustion chamber is used, without conventional film cooling, to produce an engine which operates with margin at increased temperatures and in a radiation cooled mode. The fuel coolant film is forced to leave the wall and burn, thereby improving combustion efficiency. The specific impulses expected with NTO oxidizer and MMH or anhydrous hydrazine (N_2H_4) fuels, respectively, are 321 s (ref. 23) and 326 s.

The EPS mass model was used to evaluate the impact of variations in apogee engine specific impulse. Figure 7 presents the mass in GTO as functions of apogee engine specific impulse. As the apogee engine specific impulse is increased from 311 to 325 s, mass in GTO decreases linearly about 100 kg. This reduction of GTO mass is in agreement with that predicted in reference 23 for a 1450 kg Intelsat VII spacecraft. The mass benefit of using EPS remains nearly constant, as expected, because all missions benefit from increased apogee engine specific impulse. A default value of 321 s for apogee engine specific impulse is selected for future cases using MMH and NTO propellant.

NSSK specific impulse (ISPSKC). - The bipropellant engine advances described above have also been demonstrated with lower thrust, attitude control and stationkeeping engines in steady-state and pulsed mode operation (ref. 23). A stationkeeping engine specific impulse of 310 s is expected (ref. 23).

The impact of a variation in NSSK specific impulse was examined with the EPS mass model. Figure 8 shows that the GTO mass decreases linearly by about 110 kg as the NSSK specific impulse is increased from 285 to 320 s. Because the specific impulses of the EPS are not varied, the EPS GTO mass values do not change. Thus, the mass benefits also decrease about 110 kg for the NSSK specific impulse range shown. A default value of 310 or 315 s for NSSK specific impulse is assumed for future cases using either MMH or hydrazine, respectively, and NTO propellants.

Arcjet propulsion systems. - The five arcjet operating points of table III are used as inputs to the mass model. A 15 year mission with an ETR launch of a 1500 kg dry spacecraft is assumed. A battery energy rating of 7.2 kW-hr is chosen. In general, the spacecraft mass in GTO decreases, as expected, with increases in arcjet specific impulse as shown in figure 9 and table XI. The estimated low power datum point at 510 s shows the benefit of advanced technology over that which has been demonstrated at 375 and 530 s. The 2.0 kW points at 550 and 600 s are demonstrated and extrapolated, respectively, and yield increased mass benefits compared to the chemical NSSK case. However, the required thruster lifetime, which is only 723 hr at the 550 s point, has not yet been demonstrated at these higher power conditions (ref. 24). Present and future technology levels yield mass benefits of about 100 and 150 kg, respectively, for this mission.

The major advantages of EPS using arcjets over those using ion thrusters are the simplicity and relatively low mass of arcjet systems and their high thrust to power ratio. The arcjet EPS system mass, as shown in table XI, varies from about 72 kg for the low power-510 s case to about 100 kg for the high power-high specific impulse points. Most of the increase is due to increases in power processor and thermal control masses as thruster power is increased. For this reason, the low power high-specific impulse case yields a 23 kg greater mass benefit than that for the high power-550 s point. A specific impulse of 600 s at high power is required to gain a 22 kg mass benefit advantage over the low power-510 s case. At 600 s, the NSSK propellant savings more than offsets the increased EPS mass.

Table XI also shows that for the SOA arcjet design point (530 s, 1.62 kW) the thrusting time per firing is about 1.1 hr, at a frequency of once per week and with a maximum battery depth of discharge of about 0.56. A duty cycle of one firing every four days would reduce the depth of discharge to 0.32 if the higher value were a concern.

When the spacecraft bipropellant fuel is anhydrous hydrazine, the same as that used in the arcjet, the propellant distribution mass is set equal to zero because it is assumed that the existing propellant distribution for the chemical thrusters would be adequate. In this case, the propellant required for the all chemical portion is reduced and the mass in GTO decreases by 46 kg. With EPS, the 4 kg decrease in the propellant distribution mass grows to a 6.6 kg decrease in EPS mass and coupled with reduced apogee injection propellant, to a 45 kg decrease in GTO mass such that the mass benefit does not change. Thus, the impact on the mass benefit may be small, but the mass in GTO will decrease even with spacecraft parameters different from those assumed here.

As mentioned earlier, the EPS model assumes the use of two-axis gimbals and a gimbal mass equal to 34 percent of the thruster mass. With lightweight arcjets, this only amounts to about 5 kg of GTO mass for the case of table XI. The arcjet system of reference 45 does not use gimbals, therefore, the gimbal mass fraction is set to zero for future arcjet cases.

Ion thruster propulsion systems. - The five demonstrated "derated" ion thruster operating points of table VII were utilized as mass model inputs. The results are shown in figure 10 and table XII. The baseline mission is an ETR launch of a 1640 kg dry mass spacecraft with 6.0 kW-hr of battery energy and a 15 year on-orbit requirement. Figure 10 shows that the mass in GTO with ion EPS is relatively insensitive to ion thruster specific impulse. Consequently, mass benefit values realized are about 290 kg or 2.6 times that of the SOA arcjet operating on the same spacecraft. Table XII summarizes the pertinent mass model outputs for the derated thruster and those for the IPS of reference 18 and table X. With the smaller, lower powered IPS, the EPS mass is 153 kg while those with the derated thruster vary with specific impulse by no more than 13 kg from an average value of 220 kg. As the ion thruster specific impulse increases for the derated thruster, the power processor mass also increases to nearly offset the NSSK propellant reduction. The 70 kg difference in EPS mass with the IPS or derated thrusters is compounded to a 140 kg difference in GTO mass and mass benefit. But, as explained in reference 17, the smaller thrusters may have significant lifetime issues.

Table XII also shows that the SOA derated thruster is required to operate for less than half the time of the IPS thruster and also has a smaller maximum battery depth of discharge. With the exception of the 25 cm diameter XIPS thruster, all of the low power thrusters listed in table V give total thrusting times comparable to or greater than that shown for IPS. They are also expected to have similar lifetime issues and would require very long qualification test times (ref. 17). The derated thruster also provides the option of firing once every other day if a battery depth of discharge of 0.66 is allowed. In addition, if the higher powered derated thrusters are used on lighter more powerful spacecraft (like that of table XI), thruster lifetimes of less than 2500 hr would be required with a battery depth of discharge about 0.5 and a thruster firing frequency of every other day. These conditions would reduce ion thruster qualification test times to about 7 months.

Derated thruster mass (MTH). - The sensitivity of mass model outputs to the derated thruster mass was examined for the mission of table XII. Figure 11 shows that if the thruster mass were decreased from its present value of 10.7 to 5 kg, the spacecraft mass in GTO would decrease by nearly 100 kg. This mass reduction factor of more than 17 occurs because there are four thrusters with gimbals which are assumed to be 34 percent of the thruster mass. This reduced hardware mass translates into mass reductions for structure, contingency, and propellant for all functions. Reference 17 describes how the derated thruster mass will probably be reduced to 7 kg; therefore, a thruster mass of 7 kg is used for future cases.

Figure 11 also shows the effect of reducing the gimbal mass fraction to zero with a thruster mass of 7 kg. If gimbals are not used and the thrust vectors from each thruster are neither equal nor through the spacecraft center of mass, then a disturbance force will exist. This unwanted action must be corrected by other auxiliary propulsion systems, requiring additional propellant from lower specific impulse engines. Based on the fact that neither of the two-axis gimbal systems on the Space Electric Rocket Test II (SERT II) spacecraft were required for thrust misalignment corrections (ref. 31), we believe that ion engine thrust vector alignment can be made with sufficient accuracy to take advantage of the 30 kg savings. However, that will require detailed design analyses which are beyond the scope of this study. Reference 18 selected single axis gimballing combined with thrust throttling as one possible solution.

Power processor specific mass (ALPPU). - Because of the ion thruster power processor's relative complexity, its SOA specific mass is about four times that of the arcjet's. For the baseline ion mission, the mass benefit increases about 14 kg for every 1 kg/kW decrease in power processor specific mass. The reasons for this strong sensitivity are the same as those for thruster mass. The sensitivities of mass benefits to power processor efficiency and thermal control specific mass were found to be small. Therefore, efforts to reduce the number of power supply outputs and to simplify thruster control, as demonstrated in reference 17, should reap substantial mass benefits.

Advanced technologies combined. - The combined effects of chemical and electric propulsion technological advances on the spacecraft mass in GTO and the mass benefit of using EPS are shown in figure 12 and partially in table XIII. A 1500 kg spacecraft with 7.2 kW-hr of battery storage is launched from ETR to GEO for a 15 year mission. The specific impulse values for apogee injection and attitude control-stationkeeping functions are increased from SOA values of 311 and 285 s, respectively, to 321 and 310 s by incorporating the hot rocket technology of reference 23. The mass in GTO decreases significantly by about 143 kg and nearly enables the use of a smaller launch vehicle. Then, anhydrous hydrazine replaces the monomethyl-hydrazine for another 5 s increase in specific impulse and 46 kg decrease in GTO mass to a value below the capability limit of an Atlas IIAS. This becomes the baseline case for the addition of EPS. First, low power (0.62 kW) and high power (2.0 kW) arcjets without gimbals and a propellant distribution penalty are assumed for EPS. This yields additional GTO mass reductions of as much as 154 kg which provides margin for the move to the Atlas IIAS launch vehicle. Next, a 7 kg derated xenon ion thruster is employed without gimbals and with power processor simplifications which reduce the specific masses by an estimated 2 kg/kW from those shown in figure 3. The resulting GTO mass reduction, from that of the advanced chemical case, for any ion thruster operating condition is more than 320 kg as shown in table XIII. While the changes in mass benefits with specific impulse are small for ion thruster systems, the total thrusting time decreases from 6700 hr for low powered ion thruster to 2320 hr for the 2 kW operating point which should ease qualification test time requirements to tractable values.

The maximum mass benefit of combining all of these technology advances is about 530 kg. This may have a launch cost value of about $16M or permit the use of an Atlas IIAS with 370 kg margin for payload growth.

CONCLUSIONS

A model was developed and exercised to allow comparisons between the wet masses of three-axis stabilized communications satellites, delivered to a geosynchronous transfer orbit (GTO), which utilize chemical or electric propulsion systems to provide north-south stationkeeping. The model consists of a set of simple equations which starts with the dry spacecraft mass, adds the mass of the electric propulsion system, and then calculates the propellant required for on-orbit operations and apogee injection for each propulsion system. The model is extremely flexible because the user can easily vary any or all of four mission and spacecraft parameters, six chemical propulsion parameters, and twenty-three electric propulsion parameters for either hydrazine arcjets or xenon ion thrusters. The results obtained with the model agree reasonably well with those presented for a detailed point design.

The model input parameters were varied and it was shown that the benefits of using EPS increase linearly with spacecraft mass and mission duration. For a growth version of an Intelsat VII size (1640 kg) spacecraft launched from the Eastern Test Range (ETR) with a mission duration of 15 years and SOA chemical and arcjet electric propulsion systems, the mass benefit of using the 530 s specific impulse arcjet is about 210 kg. For the same mission, the mass benefit for a 30 cm diameter derated ion thruster propulsion system operated at a specific impulse of nearly 2500 s is about 390 kg.

Advances in low thrust chemical rocket technology, such as hot rockets which use iridium-lined rhenium combustion chambers and allow increases in the apogee and on-orbit engine specific impulse values, should lead to a 143 kg reduction in spacecraft wet mass placed in GTO. Nearly half of the reduction would be due to reduced apogee propellant which applies to all missions and does not significantly affect the mass benefit of using EPS. Reductions in stationkeeping propellant reduce the mass in GTO and the mass benefits of using EPS. The use of anhydrous hydrazine instead of monomethylhydrazine (MMH) fuel may allow an additional reduction in propellant mass of about 46 kg.

Advances from SOA hydrazine arcjet operating conditions (530 s and 1.62 kW), such as to 510 s specific impulse at a thruster power of 0.62 kW or to 600 s specific impulse at a thruster power of 2 kW, would be expected to increase the mass benefits of using arcjets. For a 15 year spacecraft, with a dry mass of 1500 kg and advanced hot rocket chemical propulsion with MMH, these mass benefit increases would be 26 and 48 kg, respectively, above the 102 kg benefit obtained with the SOA arcjet.

For the other EPS, a 30 cm diameter derated ion thruster has been selected over smaller thrusters because it greatly mitigates all known ion thruster life-limiting mechanisms. For a 15 year spacecraft with a dry mass of 1640 kg and advanced chemical propulsion, the mass benefits of using this thruster are about 300 kg. The mass in GTO and the mass benefits of using ion EPS were nearly unchanged by variations in the ion thruster specific impulse from 2300 to 3000 s. However, the total thrusting time decreased, with increasing specific impulse, from 7500 to 2600 hr. The 220 kg mass of the derated xenon ion thruster propulsion system, which is more than double that of the arcjet system, was found to benefit greatly from proposed thruster mass reductions. The mass in GTO decreases 17 kg per kilogram of thruster mass reduction. With power processor simplifications and the elimination of thruster gimbals, a total propulsion system mass reduction of about 60 kg could be expected. This would be compounded through structure, contingency and propellant to yield a GTO mass reduction of about 120 kg.

Combining all of these spacecraft propulsion technology advances for chemical and electric thrusters allows GTO mass reductions of nearly 530 kg for a SOA Intelsat VII size spacecraft with a 15 year mission launched from ETR. The launch cost savings for the mass benefit could be about $16M, or the mass benefit could be used to move a larger spacecraft onto a smaller launch vehicle.

APPENDIX A

To calculate the separation mass at GTO for an all chemical system, the following steps are taken. First, a dry spacecraft mass (MSC) is selected. Then, the non-NSSK on-orbit propellant mass (MPOC) is estimated by:

$$MPOC = \frac{MSC}{1640}\left[MPOF + MPOV\,(Y)\right] \tag{A1}$$

where MPOF and MPOV, respectively, are fixed and time dependent components.

The sum of the dry spacecraft mass and the non-NSSK propellant mass is conservatively used as the final mass in the rocket equation to calculate the mass of the chemical NSSK propellant (MPSKC), including reserve propellant as:

$$MPSKC = (MSC + MPOC)\left[e^{\left[\frac{(DVNS)(Y)}{(ISPSK)(ACCG)}\right]} - 1\right](1 + PRFC). \tag{A2}$$

The initial mass (MGEOC) in geosynchronous orbit (GEO) is the sum of the dry spacecraft mass, the non-NSSK propellant mass, and the chemical NSSK propellant mass. This sum is also the final mass at the end of the apogee injection maneuver. It is used in the rocket equation to estimate the apogee injection propellant mass (MPAIC), including reserve, from:

$$MPAIC = MGEOC\left[e^{\left[\frac{DVAI}{(ISPAI)(ACCG)}\right]} - 1\right](1 + PRFAI). \tag{A3}$$

The separation mass at GTO (MGTOC) then becomes the sum of the initial mass in GEO and the apogee injection propellant and is the reference mass against which spacecraft using EPS are compared.

For the case where an EPS is utilized, the dry mass of the electric propulsion system (MEPD) is estimated as the sum of the component masses which includes thrusters, gimbals, power processors, an EPS controller and housekeeping power supplies, wiring, thermal control, propellant tankage, structure, and contingency. The propellant tankage mass (MPTK) varies with NSSK propellant mass (MPSK) for which a value is assumed at this time. (Later, values for the NSSK propellant mass and an effective specific impulse are iteratively computed and used to give final propellant tankage and dry electric propulsion system masses.) The initial value of specific impulse used is that of the EP thruster reduced by the cosine of the thruster cant angle with respect to the NS axis of the spacecraft. Next, the EPS mass is added to dry spacecraft mass to obtain a dry spacecraft mass with EPS. Using this mass, the non-NSSK propellant (MPO) is calculated (as in eq. A1) as a function of the mission duration (Y). Then, non-NSSK propellant mass is added to the dry spacecraft mass with EPS and the sum is used in the rocket equation (as in eq. A2) to calculate the NSSK propellant mass and give an initial mass (MGEO) at synchronous altitude. The electric propulsion thrusting time per firing (TTIME) is then calculated from:

$$TTIME = \frac{24}{\pi} \sin^{-1}\left[\frac{(\pi)(MGEO)(DVNS)}{172800(TTH)(COS\frac{\pi(PHI)}{180})(D)(N)}\right] \tag{A4}$$

Typical thrusting times for an EPS are 1 to 3 hr or more which result in an off-node thrusting inefficiency which increases with thrusting time (ref. 4). The consequence of this inefficiency is to reduce the specific impulse of the EP thruster. If this new value of effective specific impulse (ISPSKE) is more than 1 s smaller than the initial value used to calculate the NSSK propellant mass, then a new EPS mass and subsequent parameters are computed and compared. When the effective specific impulse changes become insignificant, the initial mass at GEO is used in the rocket equation to calculate (as in eq. A3) an apogee injection propellant mass with EPS which is added to the initial mass at GEO to give a separation mass at GTO for the EPS case (MGTOEP). The mass benefit of using an EPS is the difference between the initial masses at GTO for the chemical case and the EPS case. The GEO mass of the spacecraft with EPS and the mass benefit of using EPS are felt to be conservative because only chemical NSSK propellant is replaced by the EPS. No allowance is made for any chemical propulsion hardware which may not be used. The program then calculates some additional EPS parameters such as the total thrusting time, EPS power per firing, the number of battery cycles and the maximum battery depth of discharge.

APPENDIX B

ACCG	acceleration due to gravity, 9.8066 m/s^2
ALPPU	power processor specific mass, kg/kW
ALTC	thermal control specific mass, kg/kW
BACYC	number of battery discharge cycles by EPS
D	number of firing days per year allowed for EPS
DVAI	apogee injection delta-v, m/s
DVNS	annual NSSK delta-v, 46 m/s/yr
ETAIFM	interface module power efficiency
ETAPPU	power processor power efficiency
ETR	eastern test range
EPS	electric propulsion system
GEO	geosynchronous Earth orbit
GMF	gimbal mass fraction
GTO	geosynchronous transfer orbit
IES	ion engine system
IFMSF	interface module structure fraction
IPS	ion propulsion system
ISPAI	apogee injection engine specific impulse, s
ISPSK	PHI corrected, thruster specific impulse, s
ISPSKC	chemical propulsion NSSK specific impulse, s
ISPSKE	effective EPS specific impulse, s
ISPTH	electric thruster specific impulse, s
KWHR	spacecraft battery energy rating kW-hr
MAXDOD	maximum battery depth of discharge
MBEN	mass benefit in GTO of using EPS, kg
MCAB	mass of individual thruster wiring to a power processor, kg
MCABT	total mass of all MCAB, kg
MCF	contingency mass fraction
MCON	contingency mass, kg
MEPAD	additional spacecraft mass required for EPS use, kg
MEPD	electric propulsion system dry mass, kg
MGEO	spacecraft mass with EPS delivered to GEO, kg

MGEOC	spacecraft mass with chemical propulsion delivered to GEO, kg
MGTO	spacecraft mass delivered to GTO, kg
MGTOC	spacecraft mass with chemical propulsion delivered to GTO, kg
MGTOEP	spacecraft mass with EPS delivered to GTO, kg
MIFM	EPS controller and housekeeping power supply mass, kg
MIFS	interface module structure mass, kg
MMH	monomethyl-hydrazine fuel
MPAI	apogee injection propellant mass for EPS NSSK cases, kg
MPAIC	apogee injection propellant mass for chemical NSSK cases, kg
MPD	EPS propellant distribution mass, kg
MPM	EPS propulsion module mass, kg
MPO	on-orbit propellant mass for non-NSSK, EPS cases, kg
MPOC	on-orbit propellant mass for non-NSSK, chemical cases, kg
MPOF	fixed portion of MPO and MPOC, kg
MPOV	annual rate for variable portion of MPO and MPOC, kg/yr
MPPT	total power processor mass, kg
MPSK	EPS NSSK propellant mass, kg
MPSKC	chemical propulsion NSSK propellant mass, kg
MPTK	EPS propellant tank mass, kg
MSC	baseline spacecraft dry mass, excluding MEPS, kg
MSCEP	spacecraft dry mass with EPS, kg
MTC	thermal control mass for EPS, kg
MTH	EPS NSSK thruster mass, kg
MTMS	thrust module structure mass, kg
MTGT	total thruster and gimbal mass, kg
N,n	number of firing(s) per day(s) N = 2, 1, 1/2, 1/3 . . .
NFW	number of firings per week
NSSK	north-south stationkeeping
NTO	nitrogen tetroxide oxidizer
PEPS	total power required for EPS NSSK maneuver, kW
PHI	EPS thruster cant angle, deg
PI	the constant 3.1415927
PIFM	EPS controller and housekeeping power, kW
PRF	EPS NSSK propellant reserve fraction

PRFAI	apogee injection propellant reserve fraction
PRFC	chemical NSSK propellant reserve fraction
PTF	EPS propellant tankage fraction
PTH	EPS individual thruster input power, kW
RITA	Radiofrequency Ion Thruster Assembly
TMSF	thrust module structure fraction
TOTIME	total EPS thruster thrusting time, hr
TTH	EPS individual thruster thrust, N
TTIME	EPS thrusting time per firing, hr
VALSK	argument of the $\sin^{-1}$ function in TTIME calculation
Y	on-orbit mission duration, years

APPENDIX C

```
'****************************************************************************
'                              SKMODEL.BAS
'                           Gregory A. Majcher
'                        Cleveland State University
'                              May 13, 1991
'****************************************************************************
'
DECLARE SUB default (index1%, index2%, DefltFileName$, head$, setnum%)
DECLARE SUB FirstToLast (NumOfParams%, NumOfScreens%, ScreenNum%, first%, last%)
DECLARE SUB header (head$)
DECLARE SUB LoadData ()
DECLARE SUB scroll (n%)

DIM LAUNCHSITE AS STRING
DIM SHARED display(60) AS STRING
DIM SHARED PARAM(60)

CONST PI = 3.1415927#
CONST ACCG = 9.8066

KEY 15, CHR$(0) + CHR$(1)           'This series of statements traps for the
KEY 16, CHR$(32) + CHR$(1)          'Esc key anywhere in the program. Upon
KEY 17, CHR$(64) + CHR$(1)          'pressing this key the program will end.
ON KEY(15) GOSUB GetOuttaHere
ON KEY(16) GOSUB GetOuttaHere
ON KEY(17) GOSUB GetOuttaHere
KEY(15) ON
KEY(16) ON
KEY(17) ON
ON ERROR GOTO ErrorHandler

'----------------------------------------------------------------------------
'                              MAIN PROGRAM
'----------------------------------------------------------------------------

CLS
CALL LoadData
z% = 0
Thrtstype$ = "Ion Thruster"
Prplnttype$ = "Xenon"
LAUNCHSITE = "Eastern test range"

DVAI = PARAM(1)
DVNS = PARAM(2)
Y = PARAM(3)
MSC = PARAM(4)
KWHR = PARAM(5)
ISPAI = PARAM(6)
PRFAI = PARAM(7)
ISPSKC = PARAM(8)
PRFC = PARAM(9)
MPOF = PARAM(10)
MPOV = PARAM(11)
ISPTH = PARAM(12 + z%)
PTH = PARAM(13 + z%)
TTH = PARAM(14 + z%)
PHI = PARAM(15 + z%)
MPSK = PARAM(16 + z%)
PTF = PARAM(17 + z%)
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```
PRF = PARAM(18 + z%)
D = PARAM(19 + z%)
n = PARAM(20 + z%)
MTH = PARAM(21 + z%)
GMF = PARAM(22 + z%)
MPD = PARAM(23 + z%)
TMSF = PARAM(24 + z%)
ALPPU = PARAM(25 + z%)
ETAPPU = PARAM(26 + z%)
MIFM = PARAM(27 + z%)
PIFM = PARAM(28 + z%)
ETAIFM = PARAM(29 + z%)
MCAB = PARAM(30 + z%)
ALTC = PARAM(31 + z%)
IFMSF = PARAM(32 + z%)
MEPAD = PARAM(33 + z%)
MCF = PARAM(34 + z%)

DO

 DO

  CLS
  LOCATE 10, 15: COLOR 14: PRINT " (1) "; : COLOR 7: PRINT " View or Change Mission
  LOCATE 11, 15: COLOR 14: PRINT " (2) "; : COLOR 7: PRINT " View or Change Propuls
  LOCATE 12, 15: COLOR 14: PRINT " (3) "; : COLOR 7: PRINT " Calculate model with c
  LOCATE 13, 15: COLOR 14: PRINT "(Esc)"; : COLOR 7: PRINT " Exit"
  DO
    menu$ = INKEY$
  LOOP WHILE menu$ = ""

  SELECT CASE menu$

      CASE "1"

    head$ = "MISSION AND SPACECRAFT PARAMETERS"
    CALL header(head$)
    LOCATE 10, 10, 0
    PRINT "Please enter the launch site:"
    COLOR 14: LOCATE 12, 15: PRINT "1"; : COLOR 7: PRINT ") Eastern test range"
    COLOR 14: LOCATE 13, 15: PRINT "2"; : COLOR 7: PRINT ") Kourou"
    DO
      keys% = VAL(INKEY$)
    LOOP WHILE keys% <> 1 AND keys% <> 2
    IF keys% = 1 THEN
      LAUNCHSITE = "Eastern test range"
    ELSE
      LAUNCHSITE = "Kourou"
    END IF

    index1% = 1
    index2% = 5
    DefltFileName$ = "mission.def"
    CALL default(index1%, index2%, DefltFileName$, head$, 1)
    DVAI = PARAM(1)
    DVNS = PARAM(2)
    Y = PARAM(3)
    MSC = PARAM(4)
    KWHR = PARAM(5)
```

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  CASE "2"

index1% = 6
index2% = 7
DefltFileName$ = "propapog.def"
head$ = "PROPULSION PARAMETERS (Apogee Engine)"
CALL default(index1%, index2%, DefltFileName$, head$, 2)
ISPAI = PARAM(6)
PRFAI = PARAM(7)

index1% = 8
index2% = 11
DefltFileName$ = "propchem.def"
head$ = "PROPULSION PARAMETERS (NSSK Chemical Engine)"
CALL default(index1%, index2%, DefltFileName$, head$, 3)
ISPSKC = PARAM(8)
PRFC = PARAM(9)
MPOF = PARAM(10)
MPOV = PARAM(11)

CLS
head$ = "NSSK Electric Propulsion"
CALL header(head$)
LOCATE 10, 10
PRINT "Please enter the type of electric propulsion:"
LOCATE 12, 15: COLOR 14: PRINT "     1"; : COLOR 7: PRINT ") Ion Thruster"
LOCATE 13, 15: COLOR 14: PRINT "     2"; : COLOR 7: PRINT ") Arcjet"
DO
  keys% = VAL(INKEY$)
LOOP WHILE keys% <> 1 AND keys% <> 2
IF keys% = 1 THEN
  DefltFileName$ = "ionthrst.def"
  head$ = "ION THRUSTER PARAMETERS"
  index1% = 12
  index2% = 34
  CALL default(index1%, index2%, DefltFileName$, head$, 4)
  Thrtstype$ = "Ion Thruster"
  Prplnttype$ = "Xenon"
  z% = 0
ELSE
  DefltFileName$ = "arcjet.def"
  head$ = "ARCJET PARAMETERS"
  index1% = 35
  index2% = 57
  CALL default(index1%, index2%, DefltFileName$, head$, 4)
  Thrtstype$ = "Arcjet"
  Prplnttype$ = "Hydrazine"
  z% = 23
END IF
ISPTH = PARAM(12 + z%)
PTH = PARAM(13 + z%)
TTH = PARAM(14 + z%)
PHI = PARAM(15 + z%)
MPSK = PARAM(16 + z%)
PTF = PARAM(17 + z%)
PRF = PARAM(18 + z%)
D = PARAM(19 + z%)
n = PARAM(20 + z%)
MTH = PARAM(21 + z%)
GMF = PARAM(22 + z%)
```

```
    MPD = PARAM(23 + z%)
    TMSF = PARAM(24 + z%)
    ALPPU = PARAM(25 + z%)
    ETAPPU = PARAM(26 + z%)
    MIFM = PARAM(27 + z%)
    PIFM = PARAM(28 + z%)
    ETAIFM = PARAM(29 + z%)
    MCAB = PARAM(30 + z%)
    ALTC = PARAM(31 + z%)
    IFMSF = PARAM(32 + z%)
    MEPAD = PARAM(33 + z%)
    MCF = PARAM(34 + z%)

      CASE ELSE

    END SELECT

  LOOP WHILE menu$ <> "3"
'----------------------------------------------------------------------------
'                              CALCULATIONS
'----------------------------------------------------------------------------
' I - Chemical NSSK

   MPOC = (MPOF + MPOV * Y) * (MSC / 1640)
   MPSKC = (MSC + MPOC) * (EXP((DVNS * Y) / (ISPSKC * ACCG)) - 1) * (1 + PRFC
   MGEOC = MSC + MPOC + MPSKC
   MPAIC = MGEOC * (EXP(DVAI / (ISPAI * ACCG)) - 1) * (1 + PRFAI)
   MGTOC = MGEOC + MPAIC

' II - Electric NSSK

1  MTGT = 4 * (1 + GMF) * MTH
2  MTMS = TMSF * (MTGT + MPD)
3  MPPT = (4 * PTH * ALPPU) / ETAPPU
4  MCABT = 4 * MCAB
5  MTC = ALTC * (2 * PTH * ((1 - ETAPPU) / ETAPPU) + PIFM * ((1 - ETAIFM) / ETAIFM)
6  ISPSK = ISPTH * COS(PHI * (PI / 180))

   DO
     DO

7   MPTK = PTF * MPSK
8   MPM = MTGT + MPD + MTMS + MPPT + MIFM + MCABT + MTC + MPTK + MEPAD
9   MIFS = IFMSF * MPM
10  MCON = MCF * (MPM + MIFS)
11  MEPD = MPM + MIFS + MCON
12  MSCEP = MSC + MEPD
13  MPO = (MPOF + MPOV * Y) * (MSCEP / 1640)
    OLDMPSK = MPSK
14  MPSK = (MSCEP + MPO) * (EXP((DVNS * Y) / (ISPSK * ACCG)) - 1) * (1 + PRF)

     LOOP WHILE ABS(MPSK - OLDMPSK) > .0001

15  MGEO = MSCEP + MPO + MPSK
16  VALSK = (PI * MGEO * DVNS) / (172800 * TTH * COS(PHI * (PI / 180)) * D * n)
17  TTIME = (24 / PI) * ATN(VALSK / (SQR(1 - VALSK * VALSK)))
    OLDISPSKE = ISPSKE
18  ISPSKE = (ISPSK * 24 * SIN((TTIME * PI) / 24)) / (TTIME * PI)
```


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```
   LOOP WHILE ABS(ISPSKE - OLDISPSKE) > 1

20 MPAI = MGEO * (EXP(DVAI / (ISPAI * ACCG)) - 1) * (1 + PRPAI)
21 MGTOEP = MGEO + MPAI
22 MBEN = MGTOC - MGTOEP
23 NEW = 7 * n
24 TOTIME = TTIME * D * Y * n
25 BACYC = TOTIME / TTIME
26 PEPS = (2 * (PTH / ETAPPU) + PIPM / ETAIPM)
27 MAXDOD = (PEPS * TTIME) / KWHR

'-------------------------------------------------------------------------------
'                                    OUTPUT
'-------------------------------------------------------------------------------
' Print out the model inputs

   casenum% = 1
   DO

    SELECT CASE casenum%

     CASE 1

     CLS
     LOCATE 1, 1: COLOR 11: PRINT "MODEL INPUTS"
     COLOR 9: PRINT "Mission and Spacecraft": COLOR 7
     PRINT SPC(5); "Launchsite = "; LTRIM$(LAUNCHSITE)
     FOR i% = 1 TO 5
        outputs$ = LTRIM$(MID$(display(i%), 4))
        PRINT SPC(5); outputs$; PARAM(i%)
     NEXT
     COLOR 10: PRINT "Propulsion"
     COLOR 12: PRINT "Apogee": COLOR 7
     FOR i% = 6 TO 7
        outputs$ = LTRIM$(MID$(display(i%), 4))
        PRINT SPC(5); outputs$; PARAM(i%)
     NEXT
     COLOR 13: PRINT "NSSK-CHEM": COLOR 7
     FOR i% = 8 TO 11
        outputs$ = LTRIM$(MID$(display(i%), 4))
        PRINT SPC(5); outputs$; PARAM(i%)
     NEXT
     CALL scroll(casenum%)

     CASE 2

     CLS
     COLOR 10: PRINT "Propulsion inputs continued"
     COLOR 4: PRINT "NSSK-EP": COLOR 7
     PRINT SPC(5); "Thruster Type = "; LTRIM$(Thrtstype$)
     PRINT SPC(5); "Propellant Type = "; LTRIM$(Prplnttype$)
     FOR i% = 12 + z% TO 25 + z%
        outputs$ = LTRIM$(MID$(display(i%), 4))
        PRINT SPC(5); outputs$; PARAM(i%)
     NEXT
     CALL scroll(casenum%)

     CASE 3

     CLS
```


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```
   COLOR 4: PRINT "NSSK-EP continued": COLOR 7
   FOR i% = 26 + z% TO 34 + z%
      outputs$ = LTRIM$(MID$(display(i%), 4))
      PRINT SPC(5); outputs$; PARAM(i%)
   NEXT
   CALL scroll(casenum%)

' Print out the model outputs.
   CASE 4

   CLS : COLOR 11: PRINT "MODEL OUTPUTS"
   COLOR 12: PRINT "Chemical": COLOR 7
   PRINT SPC(5); "Other Propellant Mass (kg) = "; MPOC
   PRINT SPC(5); "SK Propellant Mass (kg) = "; MPSKC
   PRINT SPC(5); "Initial Mass at GEO (kg) = "; MGEOC
   PRINT SPC(5); "Apogee Injection Propellant Mass (kg) = "; MPAIC
   PRINT SPC(5); "Separation Mass at GTO (kg) = "; MGTOC
   COLOR 13: PRINT "Electric": COLOR 7
   PRINT SPC(5); "Total Thruster/Gimbal Mass (kg) = "; MTGT
   PRINT SPC(5); "Thrust Module Structure Mass (kg) = "; MTMS
   PRINT SPC(5); "Total Power Processor Mass (kg) = "; MPPT
   PRINT SPC(5); "Total Thruster Cable Mass (kg) = "; MCABT
   PRINT SPC(5); "Thermal Control Mass (kg) = "; MTC
   PRINT SPC(5); "Propellant Tank Mass (kg) = "; MPTK
   PRINT SPC(5); "Propulsion Module Mass (kg) = "; MPM
   PRINT SPC(5); "Interface Module Structure Mass (kg) = "; MIFS
   CALL scroll(casenum%)

   CASE 5

   CLS
   COLOR 13: PRINT "Electric continued": COLOR 7
   PRINT SPC(5); "Contingency Mass (kg) = "; MCON
   PRINT SPC(5); "Electric Propulsion Dry Mass (kg) = "; MEPD
   PRINT SPC(5); "Spacecraft Mass For EP (kg) = "; MSCEP
   PRINT SPC(5); "Other Propellant Mass (kg) = "; MPO
   PRINT SPC(5); "NSSK Propellant Mass (kg) = "; MPSK
   PRINT SPC(5); "Initial Mass at GEO (kg) = "; MGEO
   PRINT SPC(5); "Thrusting Time Per Firing (hr) = "; TTIME
   PRINT SPC(5); "Effective SK Specific Impulse (s) = "; ISPSKE
   PRINT SPC(5); "Apogee Injection Propellant Mass (kg) = "; MPAI
   PRINT SPC(5); "Separation Mass at GTO (kg) = "; MGTOEP
   PRINT SPC(5); "Mass Benefit With EP (kg) = "; MBEN
   PRINT SPC(5); "Firings Per Week = "; NFW
   PRINT SPC(5); "Total Thrusting Time (hr) = "; TOTIME
   PRINT SPC(5); "Number of Battery Cycles = "; BACYC
   PRINT SPC(5); "Power for EPS (kw) = "; PEPS
   PRINT SPC(5); "Maximum (EOL) Battery Depth of Discharge = "; MAXDOD
   CALL scroll(casenum%)

   CASE ELSE
  END SELECT

LOOP WHILE casenum% <> 99

CLS
LOCATE 10, 10: PRINT "Do you want a hardcopy? [Y/N]"
hardcopy$ = ""
DO
  hardcopy$ = INKEY$
```


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```
  hardcopy$ = UCASE$(hardcopy$)
LOOP WHILE hardcopy$ <> "Y" AND hardcopy$ <> "N"

IF hardcopy$ = "Y" THEN
Print out the model inputs
 LPRINT : LPRINT : LPRINT : LPRINT "MODEL INPUTS"
 LPRINT "~~~~~~~~~~~~~~": LPRINT : LPRINT
 LPRINT "Mission and Spacecraft": LPRINT "~~~~~~~~~~~~~~~~~~~~~~~"
 LPRINT SPC(5); "Launchsite = "; LTRIM$(LAUNCHSITE)
 FOR i% = 1 TO 5
    outputs$ = LTRIM$(MID$(display(i%), 4))
    LPRINT SPC(5); outputs$; PARAM(i%)
 NEXT
 LPRINT : LPRINT "Propulsion": LPRINT "~~~~~~~~~~~"
 LPRINT "Apogee": LPRINT "~~~~~~"
 FOR i% = 6 TO 7
    outputs$ = LTRIM$(MID$ [illegible]
    LPRINT SPC(5); outputs$; PARAM(i%)
 NEXT
 LPRINT : LPRINT "NSSK-CHEM": LPRINT "~~~~~~~~~~"
 FOR i% = 8 TO 11
    outputs$ = LTRIM$(MID$(display(i%), 4))
    LPRINT SPC(5); outputs$; PARAM(i%)
 NEXT
 LPRINT : LPRINT "NSSK-EP": LPRINT "~~~~~~~~"
 LPRINT SPC(5); "Thruster Type = "; LTRIM$(Thrtstype$)
 LPRINT SPC(5); "Propellant Type = "; LTRIM$(Prplnttype$)
 FOR i% = 12 + z% TO 34 + z%
    outputs$ = LTRIM$(MID$(display(i%), 4))
    LPRINT SPC(5); outputs$; PARAM(i%)
 NEXT
 LPRINT CHR$(27): LPRINT CHR$(12)

Print out the model outputs.
 LPRINT : LPRINT : LPRINT : LPRINT "MODEL OUTPUTS"
 LPRINT "~~~~~~~~~~~~~~": LPRINT : LPRINT
 LPRINT "Chemical": LPRINT "~~~~~~~~~~": LPRINT
 LPRINT SPC(5); "Other Propellant Mass (kg) = "; MPOC
 LPRINT "SK Propellant Mass (kg) = "; MPSKC
 LPRINT SPC(5); "Initial Mass at GEO (kg) = "; MGEOC
 LPRINT SPC(5); "Apogee Injection Propellant Mass (kg) = "; MPAIC
 LPRINT "Separation Mass at GTO (kg) = "; MGTOC
 LPRINT : LPRINT "Electric": LPRINT "~~~~~~~~~~": LPRINT
 LPRINT SPC(5); "Total Thruster/Gimbal Mass (kg) = "; MTGT
 LPRINT SPC(5); "Thrust Module Structure Mass (kg) = "; MTMS
 LPRINT SPC(5); "Total Power Processor Mass (kg) = "; MPPT
 LPRINT SPC(5); "Total Thruster Cable Mass (kg) = "; MCABT
 LPRINT SPC(5); "Thermal Control Mass (kg) = "; MTC
 LPRINT SPC(5); "Propellant Tank Mass (kg) = "; MPTK
 LPRINT SPC(5); "Propulsion Module Mass (kg) = "; MPM
 LPRINT SPC(5); "Interface Module Structure Mass (kg) = "; MIFS
 LPRINT SPC(5); "Contingency Mass (kg) = "; MCON
 LPRINT "Electric Propulsion Dry Mass (kg) = "; MEPD
 LPRINT SPC(5); "Spacecraft Mass For EP (kg) = "; MSCEP
 LPRINT SPC(5); "Other Propellant Mass (kg) = "; MPO
 LPRINT "NSSK Propellant Mass (kg) = "; MPSK
 LPRINT SPC(5); "Initial Mass at GEO (kg) = "; MGEO
 LPRINT "Thrusting Time Per Firing (hr) = "; TTIME
 LPRINT "Effective SK Specific Impulse (s) = "; ISPSKE
 LPRINT SPC(5); "Apogee Injection Propellant Mass (kg) = "; MPAI
```

```
     LPRINT "Separation Mass at GTO (kg) = "; MGTOEP
     LPRINT "Mass Benefit With EP (kg) = "; MBEN
     LPRINT "Firings Per Week = "; NFW
     LPRINT "Total Thrusting Time (hr) = "; TOTIME
     LPRINT "Number of Battery Cycles = "; BACYC
     LPRINT "Power for EPS (kw) = "; PEPS
     LPRINT "Maximum (EOL) Battery Depth of Discharge = "; MAXDOD
    END IF

    CLS
    DO
      LOCATE 10, 10: PRINT "Do you want to run another case ?  [Y/N]"
      goagain$ = INKEY$
      goagain$ = UCASE$(goagain$)
    LOOP UNTIL goagain$ = "Y" OR goagain$ = "N"

LOOP WHILE goagain$ = "Y"
END

'****************************************************************************


GetOuttaHere:
     CLS
     END


ErrorHandler:
   ErrorNumber% = ERR
'
   SELECT CASE ErrorNumber%
'
       CASE 25
         PRINT " Check Printer!"
'
       CASE ELSE
         CLS
         PRINT "Error Number = ", ErrorNumber%
'
       END SELECT
       PRINT "Press Esc to quit or any other key to continue."
       DO
         fixed$ = INKEY$
       LOOP WHILE fixed$ = ""
       IF fixed$ = CHR$(27) THEN END
       RESUME

'---------------------------------------------------------------------------
'
' This subroutine allows the user to see default parameters for inputted
' data. The user may use the defaults or override them by writing over them
' when prompted. The subroutine requires the filename containing the default
' parameters and the number of paramters to be read. To change the defaults
' the user must change the ASCII file. Sub Default passes the values to the
' main program through the array param(). They must be assigned to thier
' respective variables in the main.
'
SUB default (index1%, index2%, DefltFileName$, head$, setnum%)

    DIM statements(60) AS STRING
```


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```
    NumOfParams% = (index2% - index1%) + 1
    NumOfScreens% = INT(NumOfParams% / 15) + 1
    ScreenNum% = 1
    CALL FirstToLast(NumOfParams%, NumOfScreens%, ScreenNum%, first%, last%)
    loop1% = first% + index1% - 1
    loop2% = last% + index1% - 1

'***** VARIABLE MENU *****
    DO
      CLS
      CALL header(head$)
      count% = 3
      FOR i% = loop1% TO loop2%
         LOCATE count%, 10: PRINT display(i%) + STR$(PARAM(i%))
         count% = count% + 1
      NEXT i%
      LOCATE 20, 15: COLOR 14: PRINT " (1) "; : COLOR 7: PRINT " Change a value"
      LOCATE 21, 15: COLOR 14: PRINT " (2) "; : COLOR 7: PRINT " See more values"
      LOCATE 22, 15: COLOR 14: PRINT " (3) "; : COLOR 7: PRINT " Reset all values to
      LOCATE 23, 15: COLOR 14: PRINT " (4) "; : COLOR 7: PRINT " Save these values"
      DO
        menu$ = INKEY$
      LOOP WHILE menu$ = ""

      SELECT CASE menu$

       CASE "1"

         LOCATE 20, 15: PRINT SPC(22); : LOCATE 21, 15: PRINT SPC(22);
         LOCATE 22, 15: PRINT SPC(22); : LOCATE 23, 15: PRINT SPC(23);
         DO
           LOCATE 22, 65: PRINT SPACE$(9)
           LOCATE 22, 10
           INPUT "Enter the number of the value that you wish to change > ", pn%
         LOOP WHILE pn% < first% OR pn% > last%
         equals% = INSTR(display(pn% + index1% - 1), "=")
         IF pn% <= 15 THEN
           LineNum% = pn% + 2
         ELSE
           LineNum% = (pn% MOD 15) + 2
         END IF
         LOCATE LineNum%, equals% + 11, 1
         DO
           key$ = INKEY$
         LOOP WHILE key$ = ""
         IF key$ <> CHR$(13) THEN
           LOCATE LineNum%, equals% + 11, 0: PRINT SPACE$(30)
           LOCATE LineNum%, equals% + 11
           COLOR 15
           IF VAL(key$) = 0 AND key$ <> "-" THEN
             INPUT "", value$
           ELSE
             PRINT key$;
             INPUT "", value$
             value$ = key$ + value$
           END IF
           COLOR 7
           PARAM(pn% + index1% - 1) = VAL(value$)
         ELSE
           LOCATE 1, 1, 0
```


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```
      END IF

    CASE "2"

      CALL FirstToLast(NumOfParams%, NumOfScreens%, ScreenNum%, first%, last%)
      loop1% = first% + index1% - 1
      loop2% = last% + index1% - 1

    CASE "3"
      DFNnum% = FREEFILE
      OPEN DefltFileName$ FOR INPUT AS #DFNnum%
      FOR i% = index1% TO index2%
         INPUT #DFNnum%, statements(i%)
         equals% = INSTR(statements(i%), "=")
         NumLength% = LEN(statements(i%)) - equals%
         PARAM(i%) = VAL(RIGHT$(statements(i%), NumLength%))
      NEXT i%
      CLOSE #DFNnum%

    CASE ELSE

   END SELECT
  LOOP WHILE menu$ <> "4"

END SUB

'------------------------------------------------------------------------------
'
SUB FirstToLast (NumOfParams%, NumOfScreens%, ScreenNum%, first%, last%)

  IF NumOfScreens% = 1 THEN
    first% = 1
    last% = NumOfParams%
  ELSE
    SELECT CASE ScreenNum%
      CASE NumOfScreens%
        first% = (ScreenNum% - 1) * 15 + 1
        last% = NumOfParams%
        ScreenNum% = 1
      CASE ELSE
        first% = (ScreenNum% - 1) * 15 + 1
        last% = first% + 14
        ScreenNum% = ScreenNum% + 1
    END SELECT
  END IF

END SUB

'------------------------------------------------------------------------------
'
SUB header (head$)

  HeadLength% = LEN(head$)
  start% = (80 - (20 + HeadLength%)) / 2
  CLS : LOCATE 1, 1: COLOR 14, 7: PRINT SPACE$(80)
  LOCATE 1, start%: PRINT "* * * * * "; : COLOR 4: PRINT head$;
  COLOR 14, 7: PRINT " * * * * *": COLOR 7, 0

END SUB
```

```
' ---------------------------------------------------------------------------

SUB LoadData

  OPEN "mission.def" FOR INPUT AS #1
  FOR i% = 1 TO 5
    LINE INPUT #1, display(i%)
    equals% = INSTR(display(i%), "=")
    NumLength% = LEN(display(i%)) - equals%
    PARAM(i%) = VAL(RIGHT$(display(i%), NumLength%))
    display(i%) = LEFT$(display(i%), equals%)
  NEXT
  CLOSE #1

  OPEN "propapog.def" FOR INPUT AS #1
  FOR i% = 6 TO 7
    LINE INPUT #1, display(i%)
    equals% = INSTR(display(i%), "=")
    NumLength% = LEN(display(i%)) - equals%
    PARAM(i%) = VAL(RIGHT$(display(i%), NumLength%))
    display(i%) = LEFT$(display(i%), equals%)
  NEXT
  CLOSE #1

  OPEN "propchem.def" FOR INPUT AS #1
  FOR i% = 8 TO 11
    LINE INPUT #1, display(i%)
    equals% = INSTR(display(i%), "=")
    NumLength% = LEN(display(i%)) - equals%
    PARAM(i%) = VAL(RIGHT$(display(i%), NumLength%))
    display(i%) = LEFT$(display(i%), equals%)
  NEXT
  CLOSE #1

  OPEN "ionthrst.def" FOR INPUT AS #1
  FOR i% = 12 TO 34
    LINE INPUT #1, display(i%)
    equals% = INSTR(display(i%), "=")
    NumLength% = LEN(display(i%)) - equals%
    PARAM(i%) = VAL(RIGHT$(display(i%), NumLength%))
    display(i%) = LEFT$(display(i%), equals%)
  NEXT
  CLOSE #1

  OPEN "arcjet.def" FOR INPUT AS #1
  FOR i% = 35 TO 57
    LINE INPUT #1, display(i%)
    equals% = INSTR(display(i%), "=")
    NumLength% = LEN(display(i%)) - equals%
    PARAM(i%) = VAL(RIGHT$(display(i%), NumLength%))
    display(i%) = LEFT$(display(i%), equals%)
  NEXT
  CLOSE #1

END SUB

SUB scroll (n%)

   LOCATE 23, 15: COLOR 14
   PRINT "<P> Previous  <N> Next  <Q> Quit": COLOR 7
```

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```
DO
  wait$ = INKEY$
  wait$ = UCASE$(wait$)
LOOP UNTIL wait$ = "P" OR wait$ = "Q" OR wait$ = "N"

 SELECT CASE wait$

  CASE "N"

   IF n% = 5 THEN
     n% = 5
   ELSE
     n% = n% + 1
   END IF

  CASE "P"

   IF n% = 1 THEN
     n% = 1
   ELSE
     n% = n% - 1
   END IF

  CASE "Q"
     n% = 99

 END SELECT

END SUB
```

REFERENCES

1. Schreib, R.: Planning for Ion Propulsion on Communications Satellites. 17th International Electric Propulsion Conference, Japan Society for Aeronautical and Space Sciences, 1984, pp. 299-305.

2. Schreib, R.: Utility of Xenon Ion Stationkeeping. AIAA Paper 86-1849, June 1986.

3. Schreib, R.: Readiness Appraisal: Ion Propulsion for Communication Satellites. AIAA Paper 88-0777, 1988.

4. Agrawal, B.N.: Design of Geosynchronous Spacecraft. Prentice-Hall, Inc., NJ, 1986.

5. Sackheim, R.L.; and Howell, G.W.: Trends in Propulsion Systems for Geosynchronous Satellites. 14th International Symposium on Space Technology and Science, M. Nagatomo, Ed., AGNE Publishing, Tokyo, Japan, 1984, pp. 245-254.

6. Stone, J.R.: Recent Advances in Low Thrust Propulsion Technology. AIAA Paper 88-3283, July 1988. (Also, NASA TM-100959).

7. Smith, R.D., et al.: Development and Demonstration of a 1.8 kW Hydrazine Arcjet Thruster. AIAA Paper 90-2547, July 1990.

8. Stone, J.R.; and Bennett, G.L.: The NASA Low Thrust Propulsion Program. AIAA Paper 89-2492, July 1989. (Also, NASA TM-102065).

9. Kaufman, H.R., et al.: End-Hall Thrusters. AIAA Paper 90-2595, July 1990.

10. Komurasaki, K.; and Arakawa, Y.: Hall Current Ion Thruster Performance. AIAA Paper 90-2594, July 1990.

11. Rawlin, V.K.: Operation of the J-Series Thruster Using Inert Gas. AIAA Paper 82-1929, Nov. 1982. (Also, NASA TM-82977).

12. Beattie, J.R.; Matossian, J.N.; and Robson, R.R.: Status of Xenon Ion Propulsion Technology. J. Propulsion Power, vol. 6, no. 2, Mar-Apr. 1990, pp. 145-150.

13. Shimada, S., et al.: Ion Engine Systems Development of ETS-VI. AIAA Paper 89-2267, July 1989.

14. Bassner, H., et al.: The Design of RITA Electric Propulsion System for SAT 2. AIAA Paper 90-2539, July 1990.

15. Smith, P.: Design and Development of the UK-10 Ion Propulsion Subsystem. 20th International Electric Propulsion Conference, DGLR, 1988, pp. 196-202. (IEPC Paper 88-033, Oct. 1988; AIAA Paper 88-2911).

16. Beattie, J.R.: Electric Propulsion. Aerospace America, vol. 28, no. 12, Dec. 1990, p. 50.

17. Patterson, M.J.; and Foster, J.: Performance and Optimization of a "Derated" Ion Thruster for NSSK. AIAA Paper 91-2350, June 1991.

18. Day, M.L., et al.: INTELSAT VII Ion Propulsion Subsystem Implementation Study. AIAA Paper 90-2550, July 1990.

19. Palaszewski, B.A.: Advanced Launch Vehicle Upper Stages Using Liquid Propulsion and Metallized Propellants. NASA TM-103622, 1990.

20. Free, B.; and Dunlop, J.: Battery Powered Electric Propulsion for North-South Stationkeeping. Comsat Tech. Rev., vol. 3, Spring 1973, pp. 211-214.

21. Stechman, R.C.; and Smith, J.A.: Monomethyl-hydrazine Versus Hydrazine Fuels: Test Results Using Flight Qualified 100 LBF and 5 LBF Bipropellant Engine Configurations. AIAA Paper 83-1257, June 1983.

22. Meagher, G.M.; and McPeak, A.W.: Dual Mode Propulsion System Engine Testing 1984 JANNAF Propulsion Meeting, vol. 1, K.L. Strange, ed., CPIA-PUBL-390-VOL-1, CPIA, 1984, pp. 55-64.

23. Rosenberg, S.D.; and Schoenman, L.: A New Generation of High Performance Engines for Spacecraft Propulsion. AIAA Paper 91-2039, June 1991.

24. Morren, W.E.; and Curran, F.: Performance and Life Evaluations of a 2-kW Class Arcjet. AIAA Paper 91-2228, June 1991.

25. Morren, W.E., Private communication, NASA Lewis Research Center, Cleveland, OH.

26. Byers, D.C.; Terdan, F.F.; and Myers, I.T.: Primary Electric Propulsion for Future Space Missions. AIAA Paper 79-0881, May 1979. (Also, NASA TM-79141).

27. Brophy, J.R.; and Aston, G.: A Detailed Model of Ion Propulsion Systems. AIAA Paper 89-2268, July 1989.

28. Patterson, M.J.; and Curran, F.M.: Electric Propulsion Options for 10 kW Class Earth Space Missions. NASA TM-102337, 1989.

29. Zana, L.M.: Langmuir Probe Surveys of an Arcjet Exhaust. AIAA Paper 87-1950, June 1987. (Also, NASA TM-89924).

30. Carney, L.M.: An Experimental Investigation of an Arcjet Thruster Exhaust Using Langmuir Probes. NASA TM-100258, Apr. 1988.

31. Rawlin, V.K.; Patterson, M.J.; and Gruber, R.P.: Xenon Ion Propulsion for Orbit Transfer. AIAA Paper 90-2527, July 1990. (Also, NASA TM-103193).

32. Beattie, J.R., et al.: Xenon Ion-Propulsion System. AIAA Paper 85-2012, Sept. 1985.

33. Gruber, R.; Gott, R.; and Haag, T.: 5 kW Arcjet Power Electronics. AIAA Paper 89-2725, July 1989. (Also, NASA TM-102108).

34. Beattie, J.R.; and Penn, J.P.: Electric Propulsion - A National Capability. AIAA Paper 89-2490, July 1989.

35. Kitamura, S., et al.: ETS-III Ion Engine Flight Operations in the Extended Mission Period. J. Propulsion Power, vol. 2, no. 6, Nov.-Dec. 1986, pp. 513-520.

36. Bassner, H.; Bartoli, C.; and Trippi, A.: Status of the RITA-Experiment on EURECA. 20th International Electric Propulsion Conference, DGLR; 1988, pp. 180-185.

37. Fearn, D.G.: The Proposed Demonstration of the UK-10 Ion Propulsion System on ESA's SAT-2 Spacecraft. 20th International Electric Propulsion Conference, DGLR, 1988, pp. 186-195.

38. U.S. Launch Vehicle Specifications. Aviat. Week Space Technol., vol. 132, no. 12, Mar. 19, 1990, p. 163.

39. General Dynamics Commercial Launch Services, Inc., Mission Planners Guide for the Atlas Launch Vehicle Family. Rev. 2, July 10, 1990.

40. Corcoran, E.; and Beardsley, T.: The New Space Race. Sci. Am., vol. 263, no. 1, July 1990, pp. 72-85.

41. Ariane 4 Users Manual. Issue 1, Rev. 3, Apr. 1988.

42. Smithrick, J.J.; and Hall, S.W.: Effect of KOH Concentration on LEO Cycle Life of IPV Nickel-Hydrogen Flight Battery Cells. Proceedings of the 25th IECEC, Vol. 3, AIChE, 1990, pp. 16-21.

43. Lowery, J.E., et al.: Ongoing Nickel-Hydrogen Energy Storage Device Testing at George C. Marshall Space Flight Center. Proceedings of the 25th IECEC, Vol. 3, AIChE 1990, pp. 28-32.

44. Schiffer S.F.; Maurer, D.M.; and Markowitz, C.A.: NiH2 Battery Cell Life Tests for Low Earth Orbit Applications. Proceedings of the 25th IECEC, Vol. 3, AIChE, 1990, pp. 55-60.

45. Knowles, S.C.; Yano, S.E.; and Aadland, R.: Qualification and Life Testing of a Flight Design Hydrazine Arcjet System. AIAA Paper 90-2576, July 1990.

TABLE I. - MISSION AND SPACECRAFT PARAMETER DEFAULT VALUES

Launch Site	ETR	KOUROU
Apogee injection delta-v, m/s	1785	1514
Annual NSSK delta-v, m/s/yr	46	
Mission duration, yr	15	
Dry spacecraft mass, kg	1640	
Battery energy rating, kW-hr	6.0	

TABLE II. - CHEMICAL PROPULSION PARAMETER DEFAULT VALUES

Propellant	MMH + NTO		N_2H_4 + NTO
Apogee Injection:	SOA	Advanced	Advanced
Specific impulse, s	311	321	326
Propellant reserve fraction	0.042	0.042	0.042
NSSK:			
Specific impulse, s	285	310	315
Propellant reserve fraction	0.042	0.042	0.042
Non-NSSK Propellant Mass[a]	Time-independent		30 kg
	Variable		4 kg/yr

[a]Normalized to a spacecraft dry mass of 1640 kg.

TABLE III. - HYDRAZINE ARCJET OPERATING POINTS

Thruster specific impulse, s	Thruster input power, kW	Thruster thrust, N	Reference
375	0.62	0.130	8
510	0.62	.097	25
530 (SOA)	1.62	.225	7
550	2.0	.270	24
600	2.0	.235	25

TABLE IV. - HYDRAZINE ARCJET PROPULSION SYSTEM PARAMETER DEFAULT VALUES

Parameter	Default value	Reference
Cant angle, deg	17	
Initial propellant mass, kg	250	
Propellant tankage fraction	0.07	
Propellant reserve fraction	0.06	
Allowable firing days per year, days	365	18
Firing frequency, day^{-1}	1/7	
Thruster mass, kg	1.0	7
Gimbal mass fraction	0.34	26
Propellant distribution mass, kg	4.0	7
Thrust module structure fraction	0.31	27
Power processor specific mass, kg/kW	2.33	7
Power processor efficiency	0.9	7
Controller and housekeeping power supply mass, kg	2.2	26
Controller and housekeeping power supply power, kW	0.05	26
Controller and housekeeping power supply efficiency	0.9	26
Thruster wiring, kg	0.8	7
Thermal control specific mass, kg/kW	31	26
Interface module structure fraction	0.04	26
Mass added for EPS, kg	10	18
Contingency mass fraction	0.20	18

TABLE V. - NOMINAL OPERATING POINTS OF XENON ION THRUSTERS PROPOSED FOR NSSK

Thruster name, diameter, cm		Thruster specific impulse, s	Thruster power, kW	Thruster thrust, N	Reference
DERATED	30	2467	1.055	0.050	17
XIPS	25	2800	1.336	.064	12, 32
XIPS	13	2720	0.427	.018	34
IES, IPS	12	2906	0.61	.023	13, 18
RIT	10	3060	0.62	.015	14
UK	10	3486	0.64	.025	15

TABLE VI. - XENON ION THRUSTER PROPULSION SYSTEM PARAMETER DEFAULT VALUES

Thruster	Derated-30		IPS-12	
Parameter	Default value	Reference	Default value	Reference
Cant angle, deg	30		30	
Initial propellant mass, kg	50		50	
Propellant tankage fraction	0.2	26, 27	0.2	18
Firing frequency, day^{-1}	1		1	18
Thruster mass, kg	10.7	17	3	18
Gimbal mass fraction	0.34	26	0.73	
Propellant distribution mass, kg	7.3	27	9.0	
Power processor specific mass, kg/kW	9.0		15.1	18
Power processor efficiency	0.89		0.86	18
Controller and housekeeping power supply mass, kg	6.2	26	10.0	
Controller and housekeeping power supply power, kW	0.05	26	0.021	18
Thermal control specific mass, kg/kW	31.0	26	29.9	18

TABLE VII. - DERATED XENON ION THRUSTER (REF.17) AND POWER PROCESSOR OPERATING POINTS

Thruster specific impulse, s	Thruster power, kW	Thruster thrust, N	Power processor specific mass, kg/kW	Power processor efficiency
2285	0.640	0.030	15.1	0.86
2467	1.055	.050	9.0	.89
2649	1.370	.062	8.1	.91
2814	1.640	.073	7.5	.92
3031	1.991	.086	7.2	.93

TABLE VIII. - EXAMPLE OF EPS MASS MODEL INPUT AND OUTPUT VALUES

Model inputs	Model outputs
Mission and spacecraft Launchsite = eastern test range Apogee injection (m/s) = 1785 Annual nssk (m/s) = 46 Years on orbit (yrs) = 15 Spacecraft dry mass (kg) = 1640 Battery energy rating (kW-hr) = 6 Propulsion Apogee Specific impulse (s) = 321 Propellant reserve fraction = 0.042 NSSK-CHEM Thruster specific impulse (s) = 310 Propellant reserve fraction = 0.042 Annual other-propellant rate (kg/yr) = 4 NSSK-EP Thruster type = ion thruster Propellant type - xenon Thruster specific impulse (s) = 2467 Thruster power (kW) = 1.055 Thruster thrust (n) = 0.05 Thruster cant angle (deg) = 30 Initial nssk propellant mass (kg) = 50 Propellant tankage fraction = 0.2 Propellant reserve fraction = 0.06 Thrusting days/year (days) = 365 Firing(s)/day(s) (no./day) = 1 Thruster mass (kg) = 10.7 Gimbal mass fraction = 0.34 Propellant distribution mass (kg) = 7.3 Thrust module structure mass fraction = 0.31 Power processor specific mass (kg/kw) = 9 Power processor efficiency = 0.89 EPS controller mass (kg) = 6.2 Interface module power (kW) = 0.05 Interface module efficiency = 0.9 Thruster cable mass (kg) = 0.8 Thermal control specific mass (kg/kw) = 31 Interface module structure mass fraction = 0.04 Additional mass required for ep (kg) = 10 Mass contingency fraction = 0.2	Chemical Other propellant mass (kg) = 90 SK propellant mass (kg) = 459.3043 Initial mass at GEO (kg) = 2189.304 Apogee injection propellant mass (kg) = 1740.696 Separation mass at GTO (kg) = 3930 Electric Total thruster/gimbal mass (kg) = 57.352 Thrust module structure mass (kg) = 20.04212 Total power processor mass (kg) = 42.67416 Total thruster cable mass (kg) = 3.2 Thermal control mass (kg) = 8.256605 Propellant tank mass (kg) = 13.85771 Propulsion module mass (kg) = 168.8826 Interface module structure mass (kg) = 6.755303 Contingency mass (kg) = 35.12758 Electric propulsion dry mass (kg) = 210.7655 Spacecraft mass for EP (kg) = 1850.766 Other propellant mass (kg) = 101.5664 NSSK propellant mass (kg) = 69.28853 Initial mass at GEO (kg) = 2021.62 Thrusting time per firing (hr) = .8187724 Effective SK specific impulse (s) = 2132.397 Apogee injection propellant mass (kg) = 1607.372 Separation mass at GEO (kg) = 3628.993 Mass benefit with EP (kg) = 301.0073 Firings per week = 7 Total thrusting time (hr) = 4482.779 Number of battery cycles = 5475 Power for EPS (kW) = 2.426342 Maximum (EOL) battery depth of discharge = 0.3311037

TABLE IX. - LAUNCH VEHICLE GTO MASS CAPABILITIES

Launch Vehicle	Mass in GTO, kg	Reference
Delta 7925	1740	38
Atlas I	2275	39
Atlas II	2700	39
Atlas IIA	2840	39
Atlas IIAS	3570	39
Commercial Titan	5560	40
Ariane 40	1840	41
Ariane 42P	2540	41
Ariane 44P	2940	41
Ariane 42L	3140	41
Ariane 44LP	3640	41
Ariane 44L	4140	41

TABLE X. - EPS MASS MODEL TEST CASES

Parameter	Reference case (ref.18)		Mass model		
NSSK propulsion	Chemical	IPS	Chemical	IPS[a]	EPS[b]
Launch site	Kourou		Kourou		
Mission duration, years	15		15		
Dry spacecraft mass, kg	1640	1640	1640	1640	1640
Battery energy rating, kW-hr	N/A	N/A	----	6.0	6.0
Apogee injection specific impulse, s	N/A	N/A	311	311	311
Chemical, NSSK specific impulse, s	N/A	----	285	----	----
Electric thruster specific impulse, s	---	2906	----	2906	2906
EPS mass, kg	0	155	0	153	138
Dry spacecraft + EPS mass, kg	1640	1795	1640	1793	1778
Non-NSSK propellant mass, kg	90	90	90	98	98
NSSK propellant mass, kg	539	53	505	57	56
Mass in GEO, kg	2269	1938	2235	1948	1932
Apogee injection propellant mass, kg	1522	1289	1497	1305	1294
Mass in GTO, kg	3791	3227	3732	3253	3226
Mass benefit of EPS, kg	----	564	----	479	506
Thrusting time per firing, hr	----	1.74	----	1.73	1.71
Total thrusting time, hr	----		----	9452	9373
Power for EPS, kW	----		----	1.442	1.474
Maximum battery depth of discharge	----		----	0.41	0.42

[a]All input values estimated from reference 18.
[b]Only IPS thruster and power processor parameters from reference 18 are used.

TABLE XI. - MASS MODEL RESULTS WITH ARCJET EPS

Parameter						
NSSK Propulsion	Chemical	Arcjet				
Launch site Mission duration, years	ETR 15	ETR 15				
Dry spacecraft mass, kg	1500	1500	1500	1500	1500	1500
Battery energy rating, kW-hr	7.2	7.2	7.2	7.2	7.2	7.2
Apogee injection specific impulse, s	321	321	321	321	321	321
Chemical, NSSK specific impulse, s	310	----	----	----	----	----
Electric thruster, specific impulse, s	----	375	510	530	550	600
EPS mass, kg	0	82	72	93	100	98
Dry spacecraft + EPS mass, kg	1640	1582	1572	1593	1600	1598
Non-NSSK propellant mass, kg	82	87	86	87	88	88
NSSK propellant mass, kg	420	383	273	265	256	233
Mass in GEO, kg	2002	2052	1931	1945	1944	1919
Apogee injection propellant mass, kg	1592	1631	1535	1547	1546	1526
Mass in GTO, kg	3595	3683	3466	3492	3490	3445
Mass benefit of EPS, kg	----	-89	128	102	105	150
Thrusting time per firing, hr	----	2.05	2.60	1.11	0.92	1.05
Total thrusting time, hr	----	1601	2034	869	723	820
Power for EPS, kW	----	1.433	1.433	3.656	4.500	4.500
Maximum battery depth of discharge	----	0.41	0.52	0.56	0.58	0.66

TABLE XII.- MASS MODEL RESULTS WITH ION THRUSTER EPS

Parameter							
NSSK propulsion	Chemical	IPS[a]	Derated 30 cm thruster				
Launch site Mission duration, years	ETR 15	ETR 15	ETR 15				
Dry spacecraft mass, kg	1640	1640	1640	1640	1640	1640	1640
Battery energy rating, kW-hr	6.0	6.0	6.0	6.0	6.0	6.0	6.0
Apogee injection specific impulse, s	321	321	321	321	321	321	321
Chemical, NSSK specific impulse, s	310	----	----	----	----	----	----
Electric thruster specific impulse, s	----	2906	2285	2467	2649	2814	3031
EPS mass, kg	----	153	213	211	218	223	233
Dry spacecraft + EPS mass, kg	1640	1793	1853	1851	1858	1863	1873
Non-NSSK propellant mass, kg	90	98	102	102	102	102	103
NSSK propellant mass, kg	459	57	75	69	65	61	57
Mass in GEO, kg	2189	1948	2030	2022	2024	2026	2033
Apogee injection propellant mass, kg	1741	1549	1614	1607	1609	1611	1616
Mass in GTO, kg	3930	3497	3643	3629	3634	3638	3649
Mass benefit of EPS, kg	----	433	287	301	296	292	281
Thrusting time per firing, hr	----	2.73	1.37	0.82	0.66	0.56	0.48
Total thrusting time, hr	----	9452	7527	4483	3617	3075	2617
Power for EPS, kW	----	1.442	1.544	2.426	3.067	3.621	4.337
Maximum battery depth of discharge	----	0.41	0.35	0.33	0.34	0.34	0.35

[a]All input values estimated from reference 18.

TABLE XIII. - MASS MODEL RESULTS WITH ADVANCED TECHNOLOGIES

Parameter							
NSSK propulsion	Chemical			Arcjet		Ion	
Launch site	ETR			ETR		ETR	
Mission duration, years	15			15		15	
Dry spacecraft mass, kg	1500			1500		1500	
Battery energy rating, kW-hr	7.2			7.2		7.2	
Apogee injection specific impulse, s	311	321	326	326	326	326	326
Chemical, NSSK specific impulse, s	285	310	315	315	315	315	315
Electric thruster specific impulse, s	---	---	---	530	600	2467	3031
EPS mass, kg	----	----	----	84	89	149	162
Dry spacecraft + EPS mass, kg	1500	1500	1500	1584	1589	1649	1662
Non-NSSK propellant mass, kg	82	82	82	87	87	90	91
NSSK propellant mass, kg	462	420	413	264	232	62	50
Mass in GEO, kg	2044	2002	1995	1934	1908	1801	1804
Apogee injection propellant mass, kg	1694	1592	1554	1507	1487	1404	1405
Mass in GTO, kg	3738	3595	3549	3442	3395	3205	3209
Mass benefit of EPS, kg	----	----	----	108	154	345	340
Thrusting time per firing, hr	----	----	----	1.11	1.04	0.73	0.42
Total thrusting time, hr	----	----	----	865	816	3993	2322
Power for EPS, kW	----	----	----	3.66	4.5	2.43	4.34
Maximum battery depth of discharge	----	----	----	0.56	0.65	0.25	0.26

EPS = Propulsion module	
Interface module	Thrust module
Power processors	Thrusters
Wiring	
Thermal control	Gimbals
EPS controller and housekeeping supplies	
Propellant storage and control	Propellant distribution
Structure	Structure

Figure 1.—Electric propulsion system components.

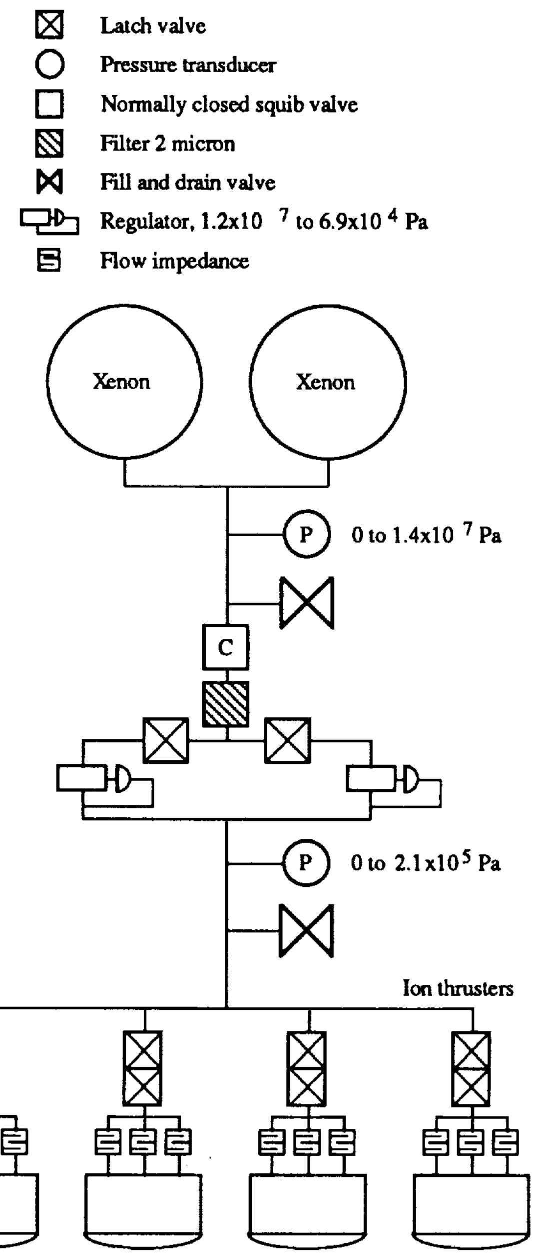


Figure 2.—Propellant tankage and distribution system (ref. 12).

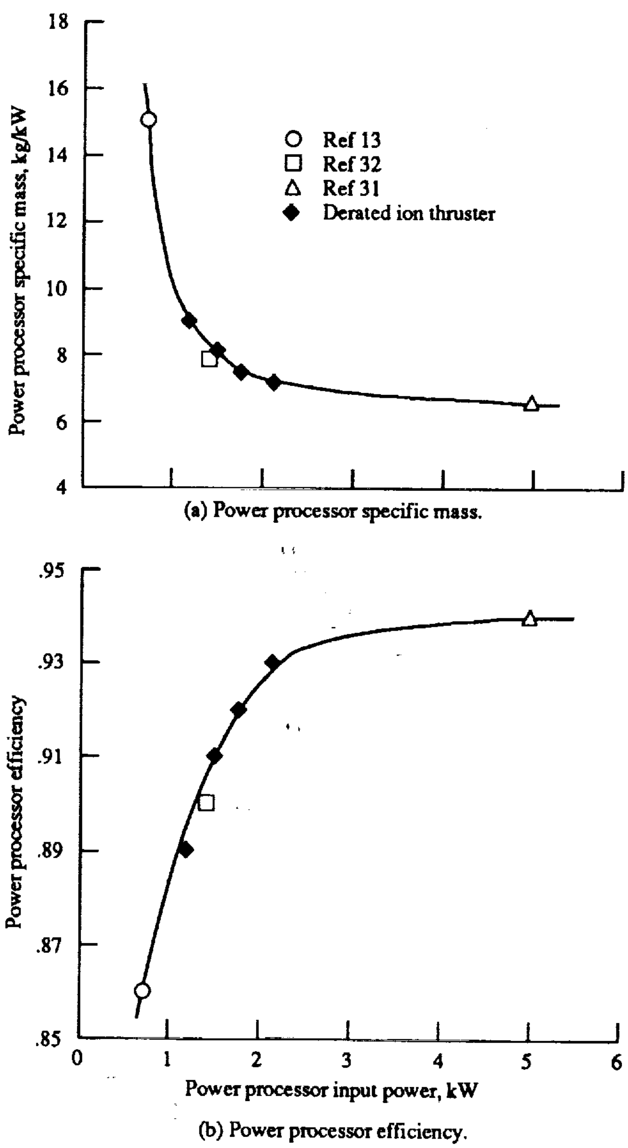


Figure 3.—Power processor specific mass and efficiency as functions of power processor input power.

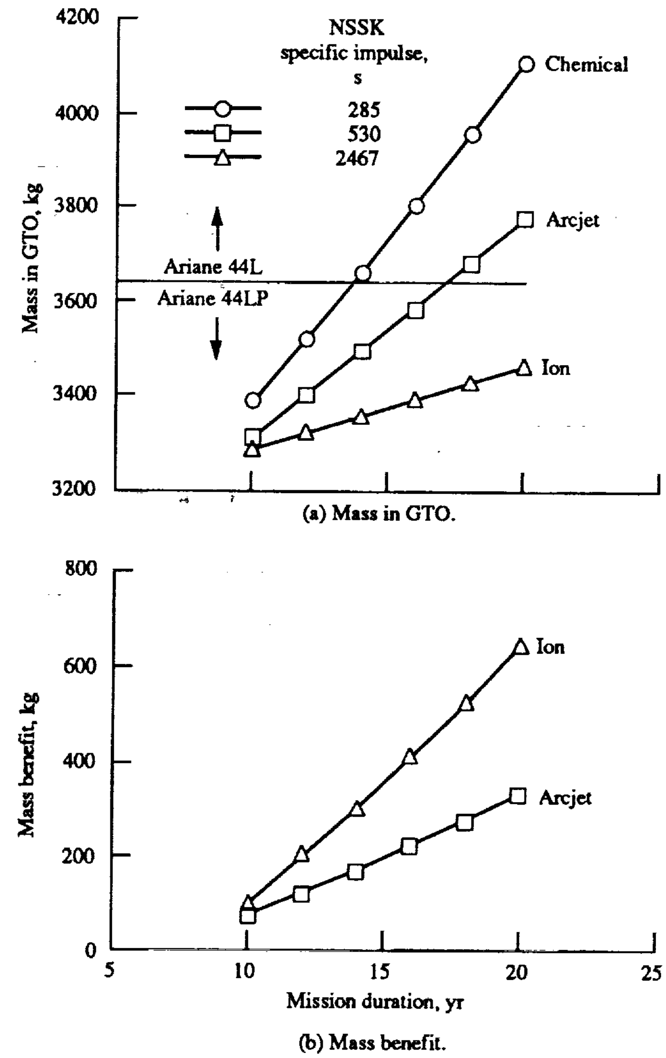


Figure 4.—Mass in GTO and mass benefit as functions of mission duration for Kourou launch, dry spacecraft mass of 1640 kg, apogee specific impulse of 311 s.

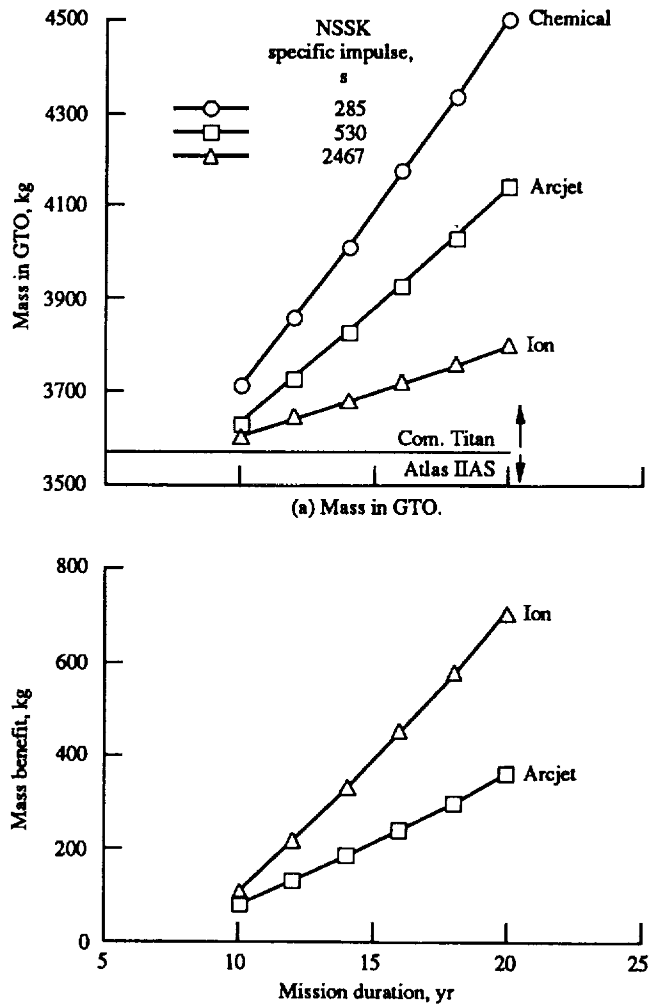


Figure 5.—Mass in GTO and mass benefit as functions of mission duration for ETR launch, dry spacecraft mass of 1640 kg, apogee specific impulse of 311 s.

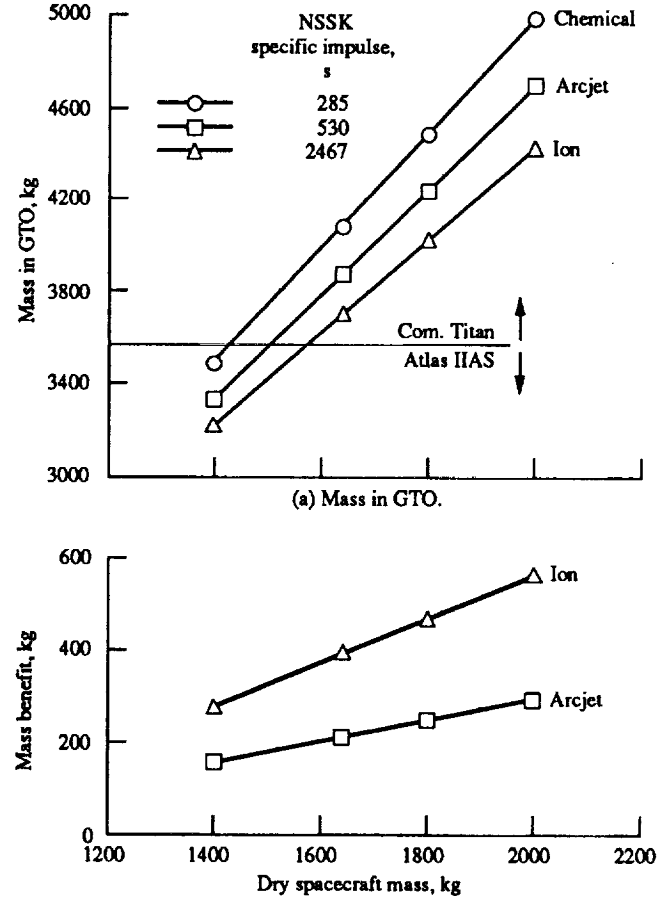


Figure 6.—Mass in GTO and mass benefit as functions of dry spacecraft mass for ETR launch, mission duration of 15 years, apogee specific impulse of 311 s.

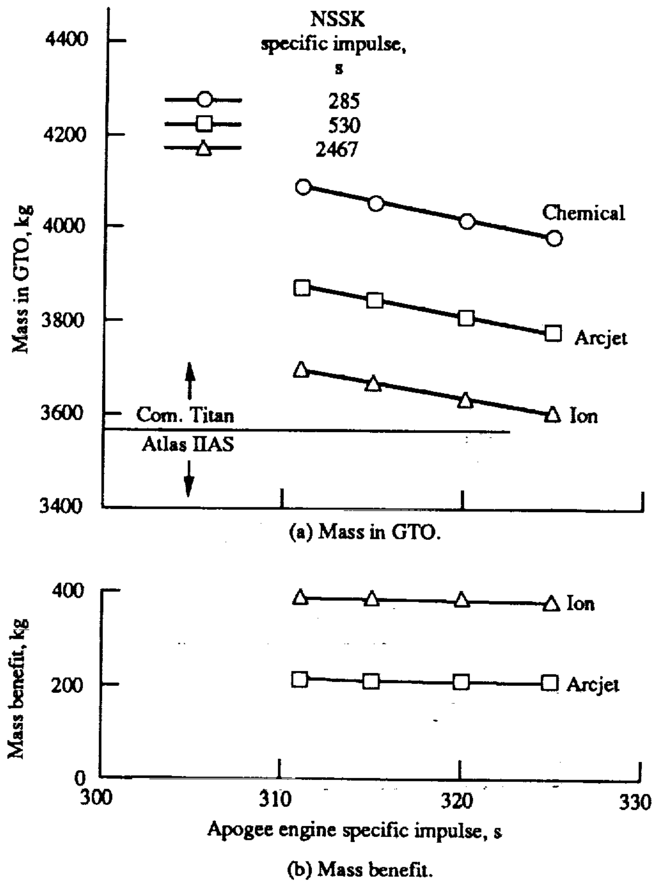


Figure 7.—Mass in GTO and mass benefit as functions of apogee specific impulse for ETR launch, dry spacecraft mass of 1640 kg, mission duration of 15 years.

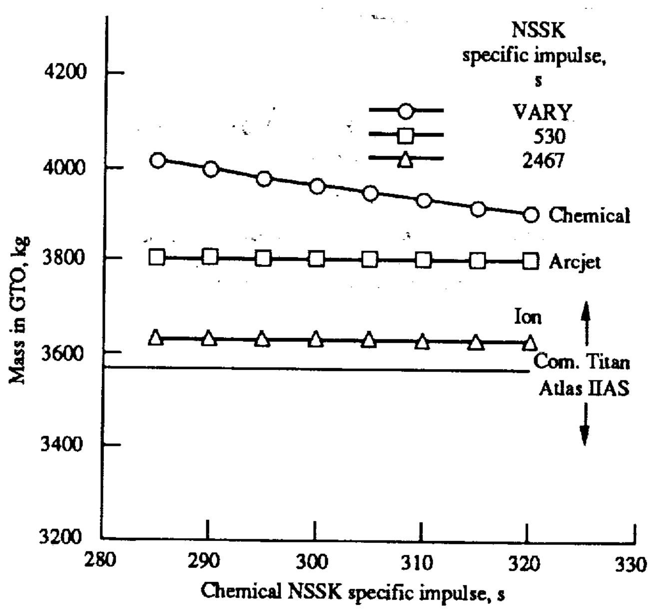


Figure 8.—Mass in GTO and as functions of NSSK chemical specific impulse for ETR launch, dry spacecraft mass of 1640 kg, mission duration of 15 years, and apogee specific impulse of 321 s.

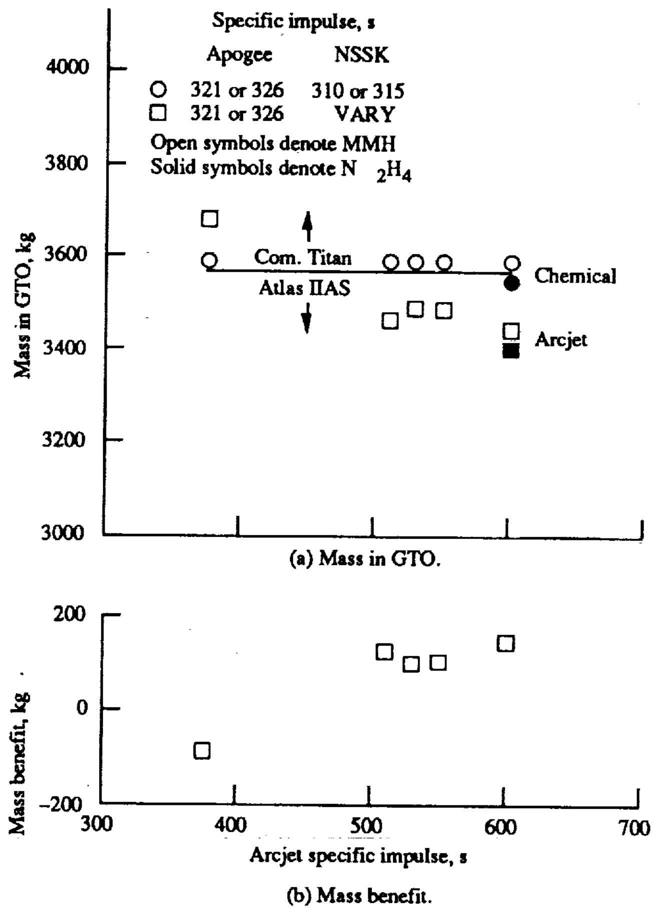
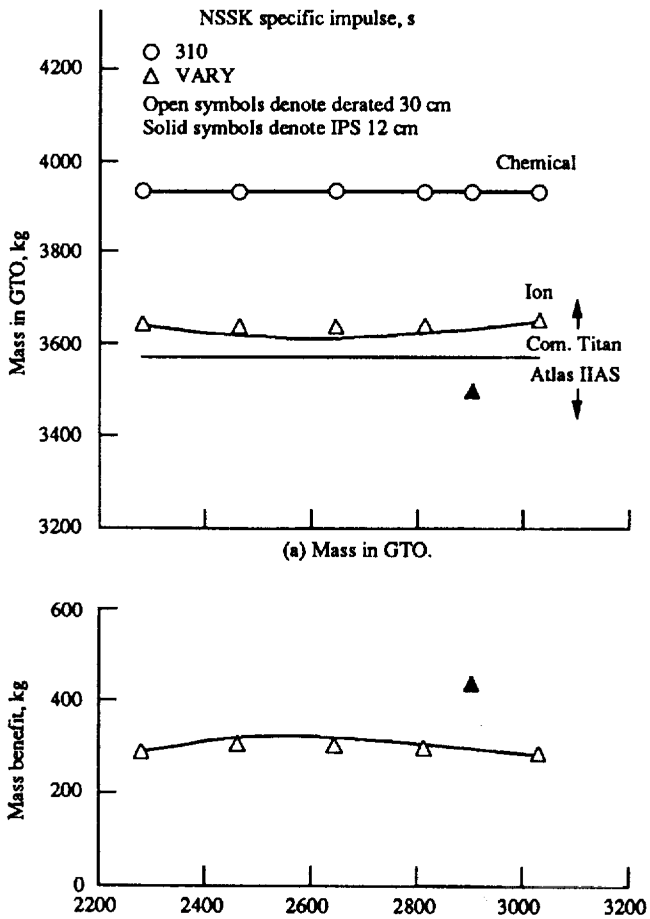


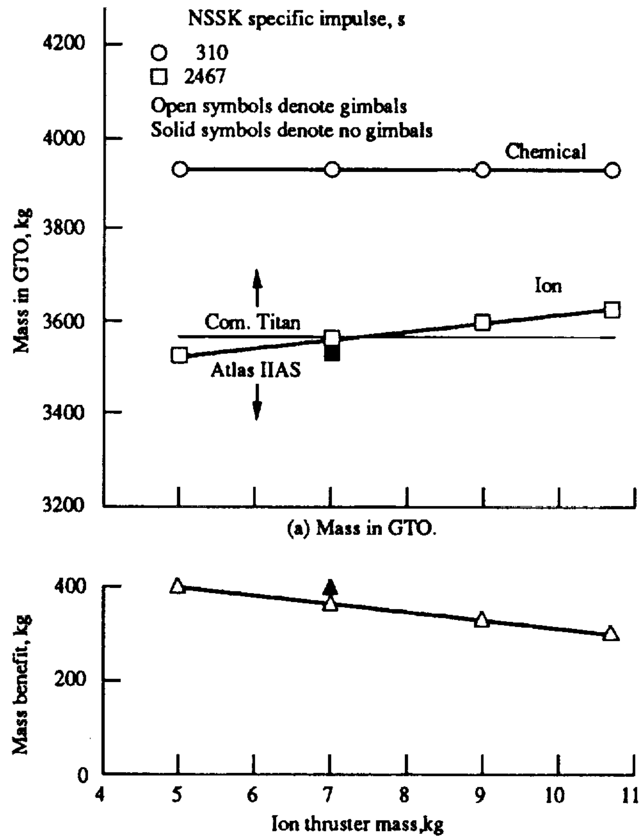
Figure 9.—Mass in GTO and mass benefit as functions of arcjet specific impulse for ETR launch, dry spacecraft mass of 1500 kg, mission duration of 15 years.



(a) Mass in GTO.

(b) Mass benefit.

Figure 10.—Mass in GTO and mass benefit as functions of ion thruster specific impulse for ETR launch, dry spacecraft mass of 1640 kg, mission duration of 15 years, apogee specific impulse of 321 s.



(a) Mass in GTO.

(b) Mass benefit.

Figure 11.—Mass in GTO and mass benefit as functions of ion thruster mass for ETR launch, dry spacecraft mass of 1640 kg, mission duration of 15 years, apogee specific impulse of 321 s.

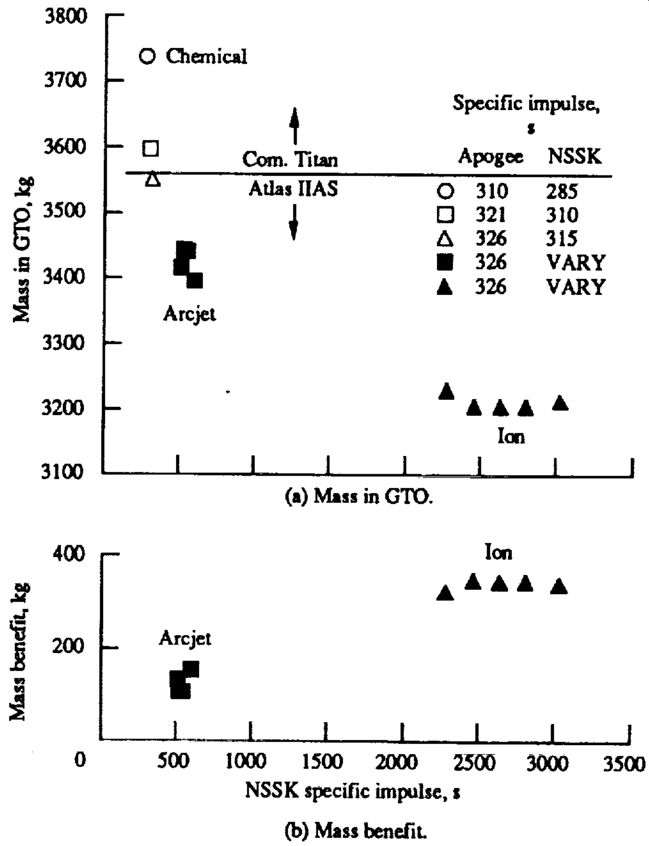


Figure 12.—Mass in GTO and mass benefit as functions of NSSK specific impulse for ETR launch, dry spacecraft mass of 1500 kg, mission duration of 15 years.

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13. ABSTRACT ***(Maximum 200 words)***

A model was developed and exercised to allow wet mass comparisons of three-axis stabilized communications satellites delivered to geosynchronous transfer orbit. The mass benefits of using advanced chemical propulsion for apogee injection and north-south stationkeeping (NSSK) functions or electric propulsion (hydrazine arcjets and xenon ion thrusters) for NSSK functions are documented. A large derated ion thruster is proposed which minimizes thruster lifetime concerns and qualification test times when compared to those of smaller ion thrusters planned for NSSK applications. The mass benefits, which depend on the spacecraft mass and mission duration, increase dramatically with arcjet specific impulse in the 500 to 600 s range, but are nearly constant for the derated ion thruster operated in the 2300 to 3000 s range. For a given mission, the mass benefits with an ion system are typically double those of the arcjet system; however, the total thrusting time with arcjets is less than one-third that with ion thrusters for the same thruster power. The mass benefits may permit increases in revenue producing payload or reduce launch costs by allowing a move to a smaller launch vehicle.

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