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NUMERICAL FEASIBILITY DESIGN CODE OF ION THRUSTER

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Abstract

In order to develop the ion engine design and development assistance code (IEDDAC) for cost reduction, in this study, the code which can derive the optimized thruster performance of ion engine was developed and evaluated, as the first step. Using this code, the optimized mass distribution ratio of spacecraft and specific impulse of ion engine was evaluated. Moreover, an integration of this code, the analysis code of ion generation part and ion extraction/acceleration part which have been developed, was proposed and discussed.

1. Introduction

1.1 Background

Use of the space circumstance has been increasing day by day. Using of electric propulsion which has high specific impulse, the range of the missions which can be attained become large; probe of asteroid, station keeping with long durability. An ion engine is most high specific impulse among electric propulsion systems in practical use. The ion engines are set to propulsion system which goes up to a candidate ordinarily as a subsystem. Present ion engine has the following thruster performance: approximately 3,000 sec specific impulse and 2.0 N/m^2 thrust density.

In the present, propellant cannot be gained in the universe. An ion engine needs electric power. It is not realizable to transmit electric power in the space isolated from the earth. Therefore, the electric power which a spacecraft can consume is limited to the electric power which the spacecraft can generate itself. In order to generate electric power, power generator mass, such as a solar cell and a thermo-electronic cell, is required. That is, the missions which can be attained are limited because the mass is less than initial mass. Accordingly, the distribution of mass of power generator, propellant and propulsion system is important.

If present ion engine is evaluated from such a viewpoint, some questions come up-- "Are the abovementioned present thruster performance best?" "Is xenon, the typical propellant of ion engine, best choice?" Moreover, improvements of power supplies mass and propulsion system for lightweight have evolved. Then, "How much is the performance which ion engine should have to be in the future?" -- such a question also comes up.

It can imagine easily that use of space circumstance will be propagated and equipments and parts of spacecraft will be improved, then, the optimal performance of ion engine should be determined by the missions and technology level in the future.

In the near future, it is likely that ion engine will be not mass-produced like a passenger car, and be custom-made. The scaling law on account of ion engine has been not established. In order to carry out custom-made of ion engine, an improvement or combination of conventional ion engine or development from nothing will be required. Therefore, these required much time and cost. It is because many experiments with high cost must be carried out. If there is a computer analysis code which can derive an appropriate solution at high speed and low cost, it will be supportable to development and reduce the cost and time.
"What calculation analysis approach for supporting development is necessary?" In this study, this approach is called "IEDDAC" – Ion Engine Design and Development Assistance Code.

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It seems that the optimized thruster performance of ion engine which can take a conveyance to target should be calculated at first, and the design of ion engine which has the thruster performance will be necessary. The element of ion engine can be divided into an ion generation part, ion extraction/acceleration part, and neutralization part. The neutralization part does not have big influence on thruster performance. Thus, the analysis of an ion generation part and ion extraction/acceleration part will be important.

The author has already developed the analysis code of an ion generation part and ion extraction/acceleration part.[1-3] However, it is necessary to refine these codes and some idea in order to integrate the IEDDAC.

1.2 Objectives

The objectives of this paper are (1) to develop the code which can derive the optimized thruster performance of ion engine which can take a conveyance to destination target, hereafter this code is called “HIT,” (2) to evaluate the code, and (3) to evaluate and estimate the integration of the analysis of an ion generation part and ion extraction/acceleration part which have been developed to IEDDAC.

2. Analysis Model of “HIT”

2.1 Generating Electric Power

There are some means of electric power generating in the universe. In this study, electric power generating by the solar cell is adopted. Generating electric power is expressed with the following equation:

$$P_t = \alpha \left(\frac{r_0}{r} \right)^2 M_c \qquad (1)$$

where M_c, r_0, r and α is solar cell mass, the distance between the sun and earth, the distance between the sun and spacecraft and a proportional constant value, respectively. This proportional constant, α, expresses the electric power generating efficiency of solar cell. Figure 1 shows that historical change of the proportional constant α.[4-5] As shown in this figure, the proportional constant α is improving every year. In this study, the α in 2000-, 2010-, and 2020-year was set up with 50, 70, and 90 [W/kg], respectively.

2.2 Weight/Power Ratio

A consumption power of ion engine is related to the generating power, the thrust is related to the beam diameter, the diameter is related to the volume of discharge chamber, and the volume is related to the mass of ion engine. Therefore, there must be a certain relation to consumption power and the mass of ion engine. The figure which plotted consumption power to the mass of practical ion engine is shown in Fig. 2. [6-8] As shown in this figure, the consumption power is approximately proportional to the mass of ion engine. This proportional constant is set to β as expressed with the following equation:

$$M_e = \beta \cdot P_t \qquad (2)$$

Yamaki *et.al.* reported[9] that they manufactured a few lightweight ion engines, and predicted the mass of ion engine in the near future. In this study, their prediction mass was referred and the β in 2000-, 2010-, and 2020-year was set up with 0.040, 0.035, and 0.030 [kg/W], respectively.

2.3 Throttling Range

As shown by above-mentioned Eq. (1), the more spacecraft is distant from the sun, the more generating electric power decreases. If consumption power decreases, ion engine cannot be operated with 100% thruster performance. It is difficult to reduce the specific impulse during the cruise. Therefore, intermittent operation of ion engine with a battery is carried out during the reduced generating power. The minimum of the operating ratio of ion engine is made into 5%, in this research. Moreover, in the situation that the generating electric power exceeds the electric power to operate ion engine with 100% thruster performance, the surplus electric power is not used. Accordingly, the operating ratio of ion engine is 5% to 100%. In addition, the charge density of storage battery is improving every year. Thus, the storage battery mass required to carry out the intermittent operation is enough smaller than the mass of ion engine, and the installation of the battery does not affect the mass distribution ratio.

2.4 Thruster Performance

The thrust F, specific impulse I_{sp}, electric power for beam extraction P_b, flow rate $\dot{m}$ and ion production cost C_i are expressed with the following equations:

$$F = \dot{m}v = I_b\sqrt{\frac{2m_x V}{e}} \tag{3}$$

$$I_{sp} = \frac{F}{\dot{m}g} = \eta\frac{F}{\dot{m}_i g} \tag{4}$$

$$\dot{m}_i = I_b\frac{m_x}{e} = \eta\cdot\dot{m} \tag{5}$$

$$P_b = P_t - C_i I_b = V_b I_b \tag{6}$$

$$C_i = \frac{V_d I_d}{I_b} \tag{7}$$

where η is propellant utilization ratio.

By rearranging and solving these equations on account of flow rate, the following equation is obtained:

$$\dot{m} = \frac{1}{\eta}\frac{2P_t}{\left[\left(\frac{I_{sp}g}{\eta}\right)^2 + \frac{2eC_i}{m_x}\right]} \tag{8}$$

Thus, the beam current I_b and net acceleration voltage V_b is calculated in the following equation:

$$I_b = \eta\frac{\dot{m}}{m_x}e = \frac{\dot{m}_i}{m_x}e \tag{9}$$

$$I_{sp} = \frac{\eta}{g}\sqrt{\frac{2eV_b}{m_x}} \tag{10}$$

2.5 Orbit Transfer

Furthermore, the propellant mass m_p consumed during time step Δt and the spacecraft mass M_t after a time step Δt are expressed with the following equations:

$$\Delta m_p = \dot{m}\Delta t \tag{11}$$

$$M_t' = M_t - \Delta m_p \tag{12}$$

Movement of spacecraft in the solar gravity can be expressed with the following equation in 2-dimensional polar coordinates ($r - \theta$):

$$F_r = m\left[\frac{d^2r}{dt} - r\left(\frac{d\theta}{dt}\right)^2 + \frac{GM}{r^2}\right] \tag{13}$$

$$F_\theta = m\left[r\left(\frac{d^2\theta}{dt^2}\right) + 2\left(\frac{dr}{dt}\right)\left(\frac{d\theta}{dt}\right)\right] \tag{14}$$

where m, G and M are the spacecraft mass, universal gravitation and mass of the sun, respectively. It is assumed that the thrust of ion engine shall be given only in the direction of a circumference. Moreover, By introducing of the following non-dimensional values (equations), the Eq. (13) and (14) can be expressed with the following equation:

$$\rho = \frac{r}{r_0}, \qquad \tau = \sqrt{\frac{GM}{r_0^3}}t, \qquad \nu = \frac{F_\theta r_0^2}{GMm} \tag{15}$$

$$\frac{d}{d\tau}\left(\rho^3 \frac{d^2\rho}{d\tau^2}+\rho\right)^{\frac{1}{2}}=\nu\rho \qquad (16)$$

$$\frac{d}{d\tau}\left(\rho^2 \frac{d\theta}{d\tau}\right)=\nu\rho \qquad (17)$$

In this calculation analysis, the differential equations of Eq. (16) and (17) were rearranged with careful attention to the time. This is for prevention and suppression of calculation error. The speed of spacecraft after a time step Δt is calculated using these differential equations.

2.6 Evaluation of Energy Error

As mentioned above, polar coordinates are used in this calculation analysis. Moreover, it is possible that an accumulation of the error by repetition calculation brings about an incorrect result. In order to obtain the correct result, the energy error calculated by following equation is evaluated each time steps. In large error case, the radius and velocity is to be regulated. However, in negligible error case, the regulation is not carried out.

2.7 Flow Chart

The schematic of this "HIT" code is shown in Fig. 3. In this study, the starting point is set to escape radius of Earth gravity, and the destination is set to the revolution orbit of Jupiter or Mercury. The payload ratio (conveyance mass / initial mass ratio) is set, at first. The ion engine and its power supply have only the role which makes the orbit transfer. Therefore, the payload contains all systems required after destination arrival. A year model is selected in 2000-, 2010- and 2020-year model, and the α and β are set as represented in Table 1(a), respectively. The propellant was selected in xenon, krypton and argon. The ion production cost and propellant utilization ratio are set as represented in Table 1(b). The temporary mass distribution ratio of spacecraft and specific impulse are set. The orbit transfer analysis is repeated until the spacecraft arrives at the destination. The ion engine is not operated when the operating ratio is less than the minimum (5%) or all propellant is consumed, that is, the spacecraft is cruising without thrust. An energy error is evaluated each time steps. The mass distribution ratio and specific impulse of spacecraft is rearranged and this analysis is repeated with simplified Newton method. In this analysis, this simple optimization method is adopted because of a small number of optimizing values; mass distribution ratio and specific impulse only. This repetition derives the finest mass distribution ratio and specific impulse to transport the conveyance to the destination with minimum cruise time.

3. Results and Discussion on "HIT"

3.1 Mission to Jupiter

Figure 4 shows a typical result on the orbit from escape radius of Earth gravity to revolution orbit of Jupiter. As shown in this figure, it is confirmed that the ion engine can carry out the mission to planet Jupiter, without swing-by method. Figure 5 shows a relationship between cruise time and payload ratio in Jupiter mission case. As shown in this figure, it is clear that the maximum payload ratio is 70%, there is a slight increase of cruise time in from 10% to 40% payload ratio range, and a steep increase of cruise time in over 40% payload ratio range. Moreover, it is also clear that the difference of cruise time each year models is significantly larger as the payload ratio is larger. Figure 6 shows a relationship between the optimized specific impulse and the payload ratio. As shown in this figure, it is clear that the optimized specific impulse is 3,000 sec., 5,100 sec. and 10,000 sec. for 30%, 50% and 70% of payload ratio in 2000-year model case.

3.2 Mission to Mercury

Figure 7 shows a typical result on the orbit from escape radius of Earth gravity to revolution orbit of Mercury. As shown in this figure, it is confirmed that the ion engine can carry out the mission to planet Mercury, without swing-by method. Moreover, as is the case with the mission to Jupiter, it was obvious that the maximum payload ratio is 70%, and that the optimized specific impulse is 3,000 sec., 5,800 sec. and 10,000 sec. for 20%, 50% and 70% of payload ratio in 2000-year model case.

3.3 Mass Distribution Ratio

Figure 8 shows the mass distribution ratio in Jupiter mission case. As shown in this figure, it is clear that the required (installed) propellant mass is almost same in all cases. Moreover, Fig. 9, the temporary change of throttling ratio in typical Jupiter mission case, represents that the ion engine is operated at 100% in spite of the high generating electric power in first 1.8 year, the engine is throttled because of the power reduction in from 1.8 to 3.2 year, and the spacecraft is coasted until it arrives at Jupiter. The unused generating power in first 1.8 year is due to the optimized mass distribution ratio and the distance from the sun; refer to Eq. (1).

3.4 Propellant

Table 2 summarizes the optimized specific impulse and cruise time for xenon, krypton and argon propellant. As indicated in this table, it is clear that the optimized specific impulse is higher as the molecular weight of propellant is smaller. However, since specific impulse is almost proportional to the square root of net acceleration voltage, it is also clear that the applied net acceleration voltage is lower as the molecular weight of propellant is smaller. This low applied voltage gives birth to an advantage; prevention of sputtering phenomena on grid system due to the reduction of back flow rate of charge exchange ion generated in grid system downstream region. It was confirmed that the required propellant mass is almost same in each year model. It is well known that the cost per mass of xenon is more expensive than that of krypton; in present, that of xenon is about 4 times as high as that of krypton. Therefore, it is likely that the using of krypton as the propellant brings about the cost reduction of transportation, though the pressure of propellant storage tank is higher. In addition, as shown in this table, it is clear that the cruise time propelled by krypton ion engine is approximately 1.1 times as long as that propelled by xenon ion engine. In 2010- and 2020-year, an advantage that uses krypton as ion engine propellant may be emphasized.

3.5 Specific Impulse

At present, typical specific impulse of ion engine is 3,000 sec. The above figures and tables indicate that the maximum payload ratio is limited to 30% in 3,000 sec. specific impulse case, and the higher specific impulse ion engine is necessary to carry out the mission with over 30% payload ratio. In large conveyance mass case, an advantage of high specific impulse ion engine may be emphasized.

3.6 Future Work

In this paper, it is assumed that only ion engine operates for the orbit transfer from escape radius of Earth gravity to planet revolution orbit, and the other propulsion system, *e.g.* chemical propulsion, is to be used to propel spacecraft from Earth ground to escape radius of Earth gravity and the payload may install the other propulsion system as its reaction control system. The use of other propulsion system has not been discussed in this study. It is safe to say that the combination between ion engine and the other propulsion system should be evaluated and discussed for more efficiency transportation mission.

Moreover, it seems reasonable to suppose that the evaluation of ion engine durability is to be necessary because the confirmed maximum life time of practical ion engine is 18,000 hours (approximately 2.0 years).[10]

Besides, it was confirmed that the energy error of each time steps was negligible small (under 1%), and the regulation of radius and velocity was not carried out. This is because the adequate time steps were adopted; 10,000 s for Jupiter mission and 2,000 s for Mercury mission.

4. Discussion for IEDDAC

As a springboard of IEDDAC, the following approach is proposed. The schematic of IEDDAC approach is shown in Fig. 10. This approach is composed of three analysis codes as follows:

Code A:

for derivation of the optimized mass distribution ratio of spacecraft and thruster performance of ion engine

step (1) Set of transportation target (destination and conveyance)

step (2) Set of initial mass of spacecraft and generating power efficiency of solar cell

step (3) Set of mass per consumption power, ion production cost and propellant utilization ratio of ion engine

step (4) Temporary set of mass distribution ratio and specific impulse of spacecraft

step (5) Analysis on orbit transfer

step (6) Calculation of cruise time
step (7) Repetition of the sequence from step (4) to step (6) with an optimization method
step (8) Derivation of the optimized mass distribution ratio of spacecraft and thruster performance of ion engine

Code B:
for derivation of the optimized design parameters of grid system -- ion extraction/acceleration part
step (1) Set of the values derived from Code A results
step (2) Calculation of net acceleration voltage and pitch of each grid holes
step (3) Temporary set of each grid hole diameters, gap between grids and each applied voltages
step (4) Ion beam trajectory calculation
step (5) Calculation of thrust, each grid impinged currents and beam divergence angle
step (6) Repetition of the sequence from step (3) to step (5) with an optimization method
step (7) Derivation of the optimized design of grid system

Code C:
for derivation of discharge chamber design parameters and thruster performance
step (1) Set of the values derived from results on both Code A and B
step (2) Set of discharge chamber volume and consumption power
step (3) Derivation of discharge chamber design parameters
step (4) Derivation of propellant utilization ratio and ion production cost

Integration of Code A, B and C:
for total design of ion engine
by repetition of the sequence from Code A to Code C until the value of ion production cost and propellant utilization ratio derived from Code C is to be same that temporary set at Code A.

Code A has been developed and discussed in chapter 2 and 3 of this paper as “HIT” code. Moreover, mostly Code B and C have also been developed and evaluated already.[1-3] These codes are called GAIGX and TPEC, respectively. The GAIGX is composed of IGX, which is analysis code of ion beam optics, and an optimization method with genetic algorithm (GA). The TPEC is a thruster performance evaluation code through the primary electron path is defined by the principle of minimum dissipation and an optimization method with genetic algorithm. In previous works, it has been confirmed that the results derived these codes are good agreement with experimental data.

In order to integrate these codes, it seems that the following discussion and improvements are necessary.
(1) Matching of each in/out interfaces
Since the input/output parameters of these codes do not coincide with each other, these codes should be improved to match their interfaces.
(2) Precisely definition of optimization target and criterion
The optimization target and criterion are different each missions. A mission's priority may be cruise time, another may be conveyance mass, and the other may be cost. The balance of these priorities significantly influences the results. It seems reasonable to suppose that this precisely definition actually make up the majority of IEDDAC integration.

5. Concluding Remarks

(1) The code which can derive the optimized thruster performance of ion, “HIT” code, was developed.
(2) In over 30% payload case, the ion engine which has higher specific impulse (over 4,000 sec.) is well suited to deep space mission.
(3) In 2010- and 2020-year, an advantage that uses krypton as ion engine propellant may be emphasized because of small difference from cruise time propelled by xenon ion engine, and because of small sputtering phenomena (long durability).
(4) In order to develop the ion engine design and development assistance code (IEDDAC), it was proposed the integration of three analysis codes, “HIT,” “GAIGX” and “TPEC” code, with matching of in/out interfaces and precisely definition of optimization target and criterion.

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Table 1　Setting parameters (left:(a), right:(b))

	Model		
	2000 yr	2010 yr	2020 yr
α [w/kg]	50	70	90
β [kg/w]	0.040	0.035	0.030

	Xe 131.3 a.m.u.	Kr 83.8 a.m.u.	Ar 39.9 a.m.u.
C_t [W/A]	200	231	260
η [-]	0.9	0.9	0.9

Table 2　Optimized specific impulse and cruise time (Jupiter)

Payload ratio 40%

I_{sp} [sec.]	Model		
	2000-Y	2010-Y	2020-Y
Xe	3900	3850	3800
Kr	4300	4250	4200
Ar	5200	5350	5200

C.Time [year]	Model		
	2000-Y	2010-Y	2020-Y
Xe	7.9	5.7	4.3
Kr	9.1	6.4	4.7
Ar	11.3	8.4	6.4

Payload ratio 50%

I_{sp} [sec.]	Model		
	2000-Y	2010-Y	2020-Y
Xe	4850	5100	4950
Kr	5400	5300	5200
Ar	6150	6350	6200

C.Time [year]	Model		
	2000-Y	2010-Y	2020-Y
Xe	12.8	9.4	6.9
Kr	13.7	10.4	7.6
Ar	16.0	11.9	9.4

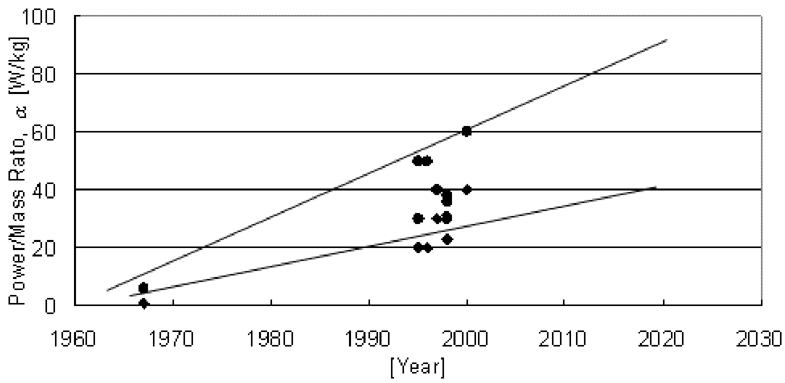

Figure 1　Electric power generating efficiency of solar cell (array)

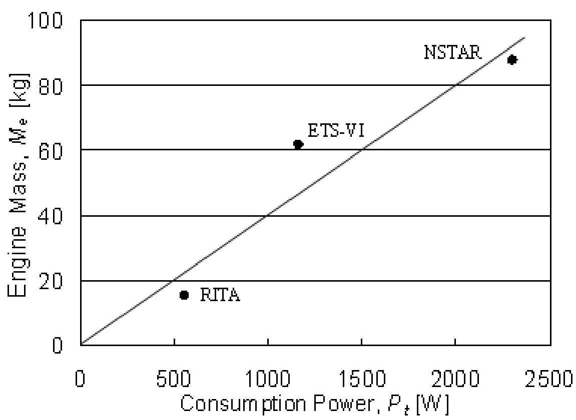

Figure 2　Engine mass / consumption power ratio

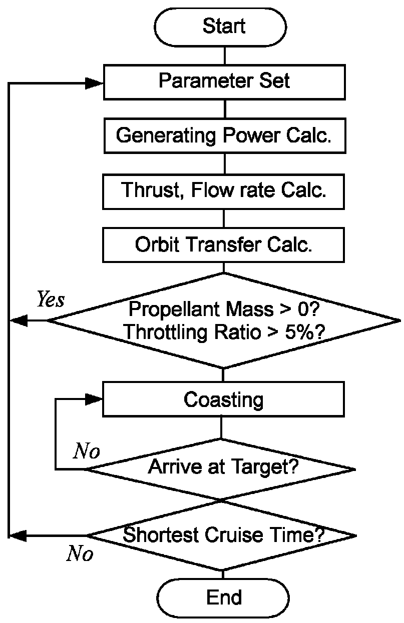

Figure 3　Flow chart of “HIT” code

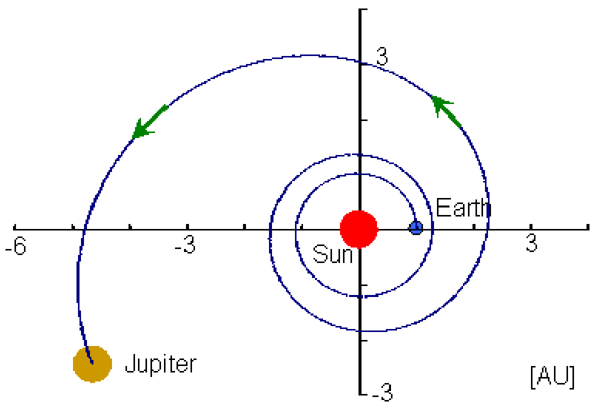

Figure 4　Typical orbit trajectory of mission to Jupiter

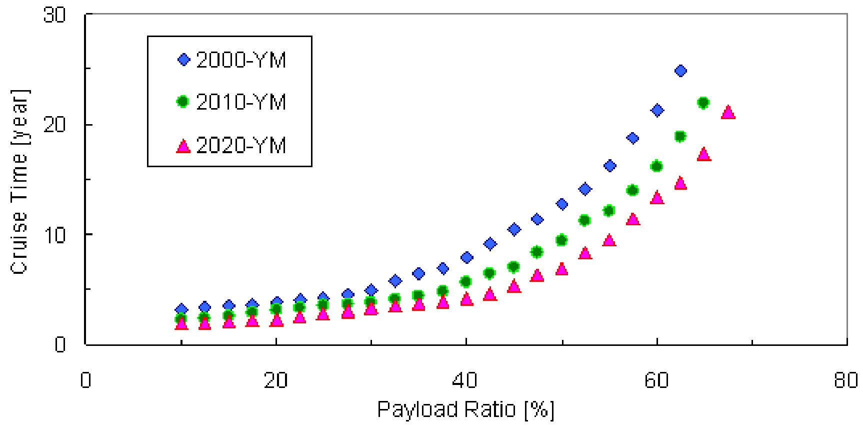

Figure 5　Cruise time vs. payload ratio (Jupiter)

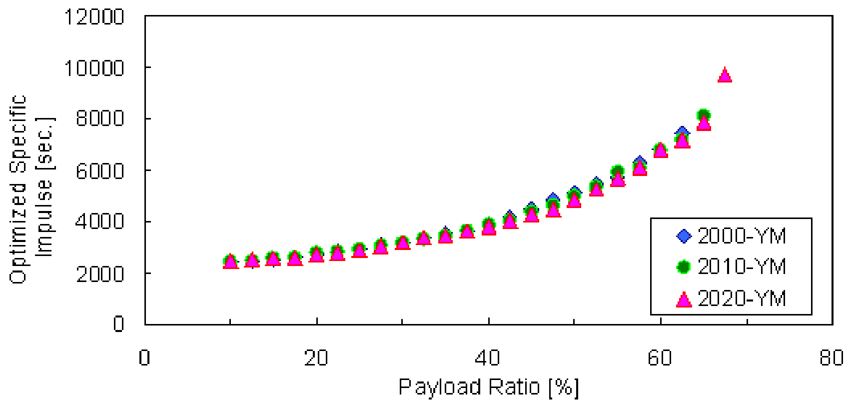

Figure 6　Optimized specific impulse vs. payload ratio (Jupiter)

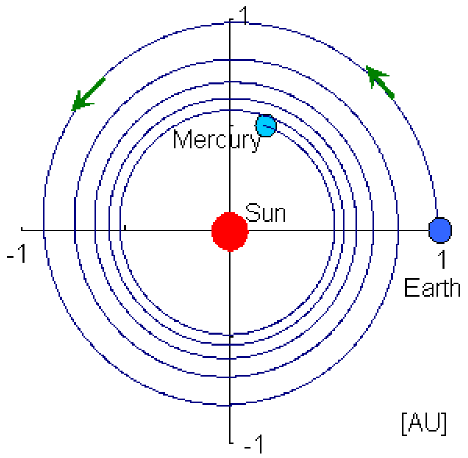

Figure 7　Typical orbit trajectory of mission to Jupiter

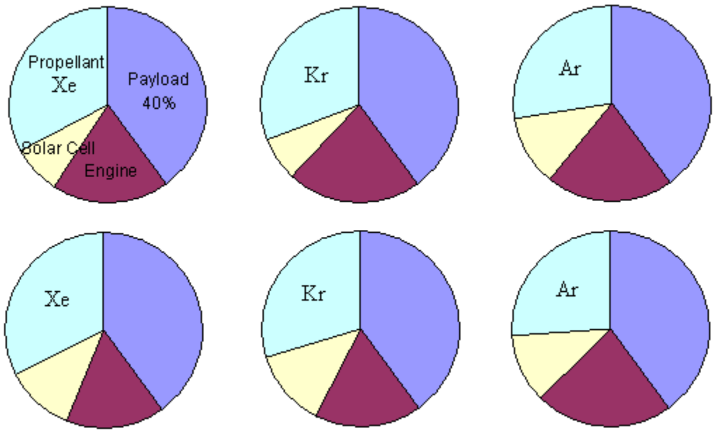

Figure 8 Mass distribution ratio (Jupiter, top:2000-YM, bottom:2020-YM)

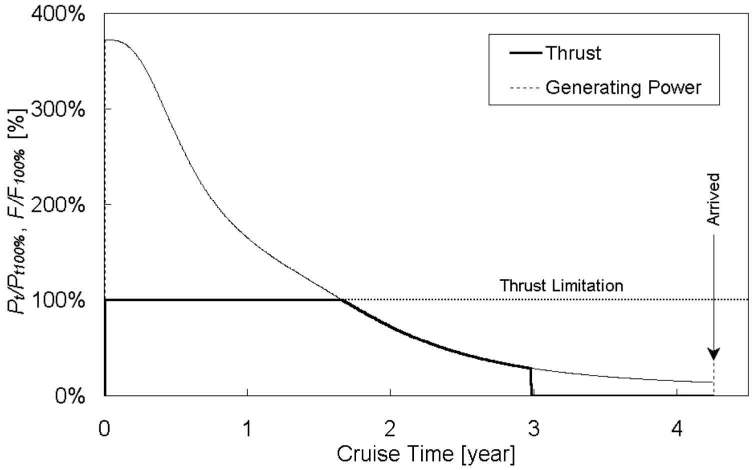

Figure 9 Typical throttling ratio (Jupiter)

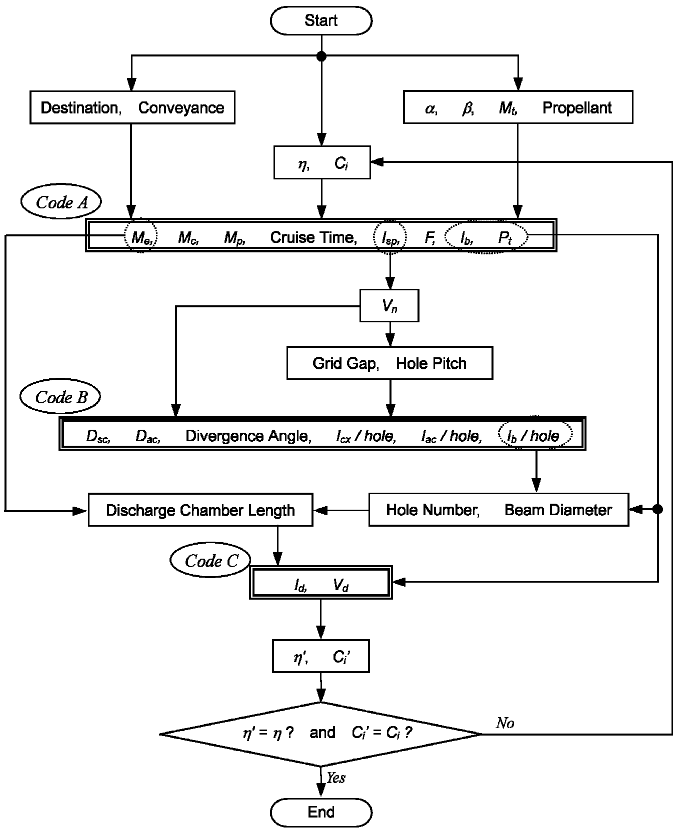

Figure 10 Flow chart of IEDDAC