3 Spacecraft System

The spacecraft system design, including the orbit profiles and operation concept, are presented in this section. Section 3.1 summarizes the requirements to the spacecraft system derived from the discussion in the previous chapters. Obviously, the most important requirement of SOLAR-C Plan-A is to place the spacecraft at a high solar latitude. Working out the way to reach there dominates the mission design, which is discussed in Section 3.2. Several options are investigated, and an option using solar electric propulsion (SEP option) is presented as the primary option in this report. Based on the system requirements and the orbit design results, the spacecraft system for carrying out the mission is briefly investigated. The objective of the study is to confirm the feasibility of the mission mainly from the point of view of the resource budget, and its successful result is given in Section 3.3. The details of the subsystems are to be shown in Section 3.4, At this stage of interim report, however, the main focus is on those subsystems that require newly developed technology (communication system, light weight solar panel, and electric propulsion system), and the other subsystems are only briefly touched.

3.1 Spacecraft System Requirements

The major requirements to the spacecraft system (including orbit design) are,
- To reach the solar latitude of 40° by early 2020's.
- To support science instruments' mass of 130 kg (or larger).
- To transmit the scientific data continuously recorded in the rate of 100 kbps to the ground.

The third item is interpreted as the 300 kpbs down link rate under the assumption of 8 hours use of the ground station per day.

The following requirements are also recognized. However, they are regarded as non-dominating in the total system design and left for the further study in this report.
- Providing payloads a stable platform for imaging observations
- 150 W (TBD) payload power
- Regular notification of absolute time for regular-interval exposures required by the helioseismic instrument.

3.2 Orbit and Mission Profile

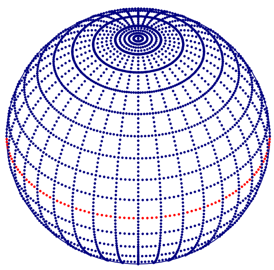
The SOLAR-C Plan-A is the first Japanese solar mission that operates in the interplanetary space. The following new issues in mission profile should be noticed when compared with the other Japanese (Earth orbiting) solar missions.
- It cruises in an interplanetary orbit, and takes much longer time to reach the final orbit.
- It only carries light payloads even if a heavy launcher (as H-IIA) is used.
- It is in severer radiation environments.

The following new issues from other Japanese interplanetary missions should be noticed as well.
- It requires much higher telemetry rate than ever achieved.

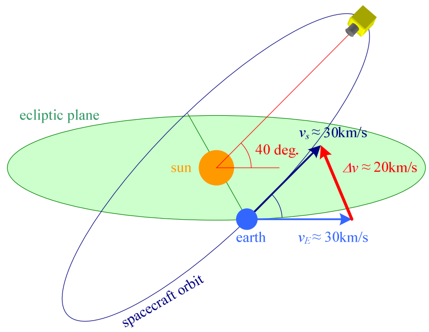
This item is not only reflected in the communication system design, but also considered in the orbit design to keep the distance from the Earth to the spacecraft short.

3.2.1 Trajectory Options for SOLAR-C Plan-A

The major mission requirement of SOLAR-C Plan-A is to observe the Sun from high latitudes. The target maximum latitude is tentatively specified as 40° (Figure 21a). To observe the Sun from the high latitudes, the space observatory (spacecraft) must be in an orbit largely inclined with respect to the ecliptic plane. It is not an easy task to inject the spacecraft into the orbit of this type. A rough estimate shows that the velocity increment required to inject the spacecraft into this largely inclined orbit is approximately 20 km/s (Figure 21b).



(a) View from lat. 40° of the sun



(b) Challenge to lat. 40°

Figure 21. Schematics of Trajectory Design Challenge in SOLAR-C Plan-A

In order to achieve this severe mission target, possible trajectory sequences are investigated considering the application of various trajectory manipulation techniques. The items considered are, the geometric relation (i.e. the tilt of the solar equatorial plane to the ecliptic plane), launcher capacity, planetary gravity assists, and the usage of a highly efficient propulsion system. As a result, four major trajectory options are listed up to achieve the mission objective (Figure 22).

The first option is called “SEP option”, which is characterized by the usage of the solar electric propulsion (SEP) combined with the Earth gravity assists (EGAs). In this option, SEP is used to increase the relative velocity (v_∞) at the Earth encounter and the direction of $\mathbf{v}_\infty$ is changed by EGA to contribute to inclination increase. As a result of the sequential use of this process, latitude 40° can be achieved in five years from the launch. The other options are characterized by the usage of the planetary gravity assists only, without using SEP. The second option is called “Jupiter option”. In this option the operation sequence begins by using the Jupiter gravity assist (JGA) to highly incline the orbit plane from the ecliptic plane. What is unique to this option is the usage of EGAs in order to reduce the orbit period after JGA, which enables the frequent observation of the solar polar region. The remaining two options are characterized by the usage of EGAs and the Venus gravity assists (VGAs). In both options the operation sequence begins with the sequence of EGAs/VGAs to increase v_∞ to the planets. Then, the sequential planetary gravity assists are used to change the direction of $\mathbf{v}_\infty$ so as to contribute to incline the orbit. “Venus-1 Option” uses the Venus for the plane change gravity assists, whereas “Venus-2 Option” uses the Earth for this purpose.

Among the four options, the first "SEP option" has a major advantage in complying with the most of the mission requirements (the detail shown in 3.1.2). The option has frequent launch windows (every half year), which should be also counted as a merit from the programmatic point of view. On the other hand, it must be noted that this option requires an advanced technology, which is not required in the other options. This point will be discussed in 3.3. In view of the major advantage mentioned above, in the following part of Section 3, the case of SEP option is focused on. Alternative Jupiter option with the same final orbit is shown in Appendix B.

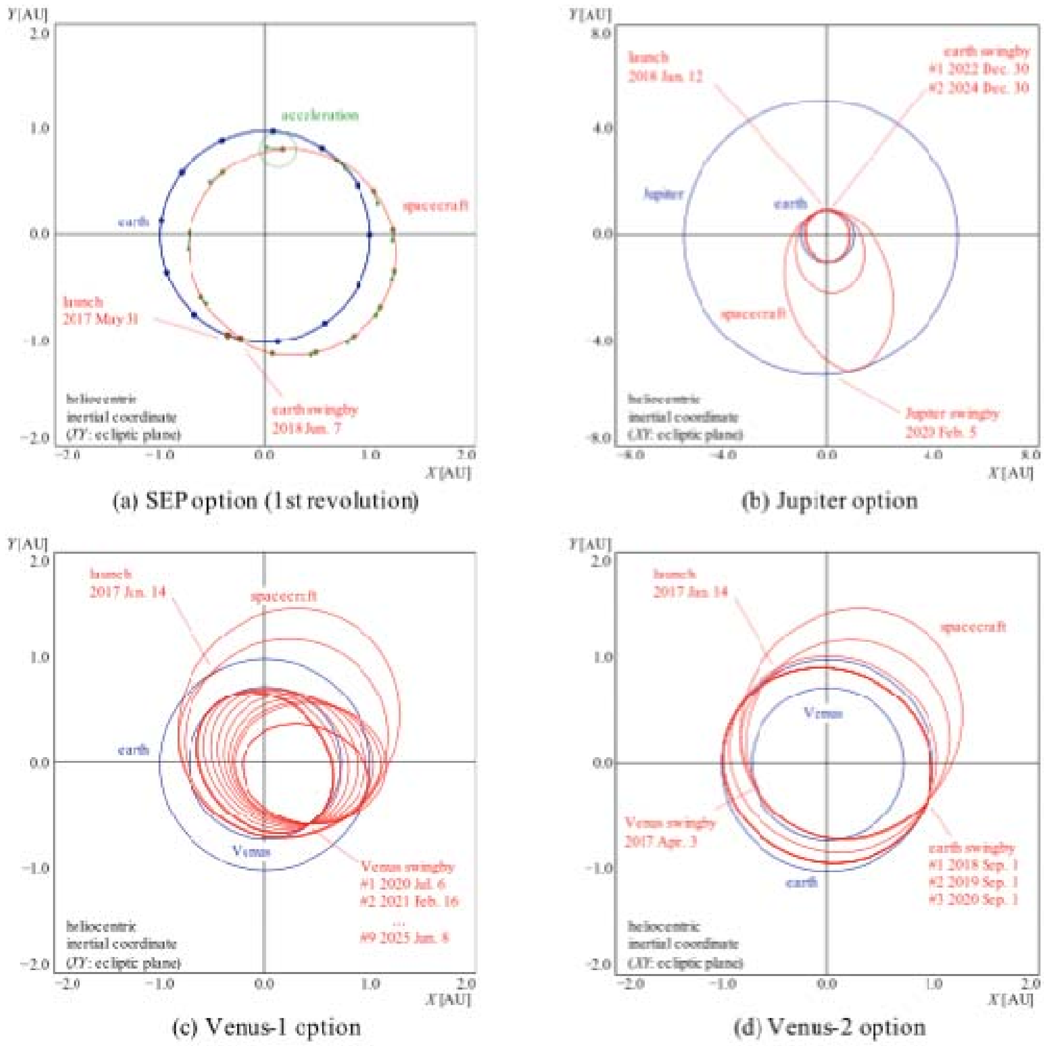


Figure 22. Trajectory Options for SOLAR-C Plan-A

3.2.2 Orbit and Mission Profile of SEP Option

Before showing the baseline sequence of the SEP option, a couple of its variants are introduced briefly. The first method, which is called "Direct Inclining Method (DIM)", is a simple one which uses SEP directly to increase the inclination. The second method uses SEP combined with EGA, which is called Electric Propulsion Delta-V Earth Gravity Assist (EDVEGA). The method uses

EDVEGA repetitively, and is called “Sequential EDVEGA Method (SEM)”. Both methods were investigated quantitatively, and it was concluded that, DIM is infeasible from the points of view of the spacecraft’s mass budget and the operation time of the ion engine system (IES), whereas SEM is feasible in these aspects. A slight variation of SEM is also investigated, which applies an additional Venus gravity assist (VGA) prior to the sequential EDVEGA. This method has an advantage in reducing the launch energy drastically compared to the original SEM. However, the use of Venus/Earth gravity assists limits the launch opportunity, which makes it difficult to take advantage of the geometrical relation (i.e. the tilt of the solar equatorial plane to the ecliptic plane). From this point, the usage of VGA is regarded only as a backup option.

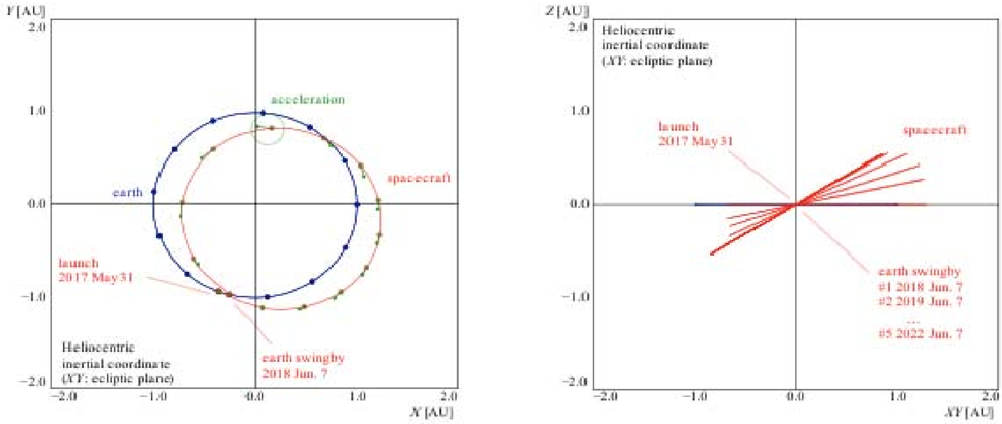
As a result of the discussion above, SEM is adopted and used to construct the baseline sequence of the SEP option. The basic procedure of SEM is described as follows.

1. The spacecraft is injected into the Earth synchronous orbit to re-encounter the Earth after one year cruise.
2. During the cruise, SEP is used to maximize the spacecraft’s relative velocity to the Earth (v_∞) at the next Earth encounter. Note that the thrust does not necessarily increase the inclination by itself. To enhance the efficiency to increase v_∞, an elliptic orbit is used for the cruise orbit.
3. By EGA, the direction of $\mathbf{v}_\infty$ is changed to contribute to the inclination increase.
4. By the repetitive use of the steps 2 and 3, the inclination is increased step by step.

Before presenting the constructed trajectory sequence, major assumptions used in the trajectory design should be noted.

The first assumption is related to the launch condition. The assumed launcher is the Japanese H-II A heavy launch vehicle equipped with a solid motor upper stage. The initial mass of the spacecraft is assumed to be 1200 kg. Assuming the practical settings of the launch site and the launch direction, the launcher is capable of injecting the spacecraft into the escape orbit with the excessive velocity (v_∞) of 7.3 km/s. The launch date is selected to take advantage of the not negligible tilt (7°.25) of the solar equatorial plane to the ecliptic plane, and they are June 7 or December 8. In the following discussions, June 7 is used as the launch date.

The second assumption is related to the ion engine system (IES). The specific impulse (Isp) of IES is assumed to be 3800 s, and the maximum thrust of IES is assumed to be 120mN in total. In the trajectory design, the actual thrust available for the maneuver is constrained by the available power, which is assumed to decrease as the inverse square of the distance from the Sun.



(a) Projected on the ecliptic plane (1st revolution) (b) Side view

Figure 23. Trajectory profile of SEP option

The third assumption is related to the eccentricity (e) of the cruise orbit. To efficiently increase v_∞, the eccentricity e should be large. However, considering the difficulties in the thermal design of the spacecraft, the eccentricity e is constrained to be less than 0.3 in the trajectory design.

Figure 23 shows the trajectory profile of the SEP option. In Figure 23a, the trajectory of the first revolution is projected on the ecliptic plane. The figure shows that the spacecraft's orbit has eccentricity, and intersects with the ecliptic approximately at the Earth's position at the launch. The acceleration vectors indicate that the thrust is used to decelerate the spacecraft at aphelion and to accelerate the spacecraft at perihelion, which result in the increase of v_∞ at the next Earth encounter. Figure 23b shows the trajectory profile through the sequence. The trajectory is projected on the plane perpendicular to the ascending node direction so that the gradual change of the orbit plane can be easily seen. It is observed that the first four cycles have asymmetry resulted from the orbit eccentricity. However, the orbit is finally circularized by EGAs, which results in the symmetry observed in the final orbit (the orbit whose inclination is the highest).

Table 7. Sequence of events of SEP option

Date	Event	v_∞	i_{SEQ}
2017/05/31	Launch	7.3 km/s	7.2°
2018/06/07	EGA #1	9.8 km/s	7.3°
			18.8°
2019/06/07	EGA #2	12.3 km/s	20.2°
			25.8°
2020/06/07	EGA #3	14.6 km/s	27.2°
			33.0°
2021/06/07	EGA #4	15.7 km/s	34.9°
			38.1°
2022/06/07	EGA #5	16.6 km/s	39.8°
			40.0°

The sequence of events of the SEP option is summarized in Table 7. In the table, v_∞ denotes the relative velocity to the Earth at EGA, and i_{SEQ} denotes the inclination to the solar equatorial plane. The spacecraft reaches i_{SEQ} of 30° after the 3rd EGA (3 years from the launch), and finally reaches i_{SEQ} of 40° after the 5th EGA (5 years from the launch).

Figure 24 shows the profiles of some important parameters of the mission. The top chart indicates the points of events and basic operation concept. The following three charts respectively show the profile of the spacecraft's distance from the Sun and the Earth, its instantaneous solar latitude, and the expected down link rate of the scientific data.

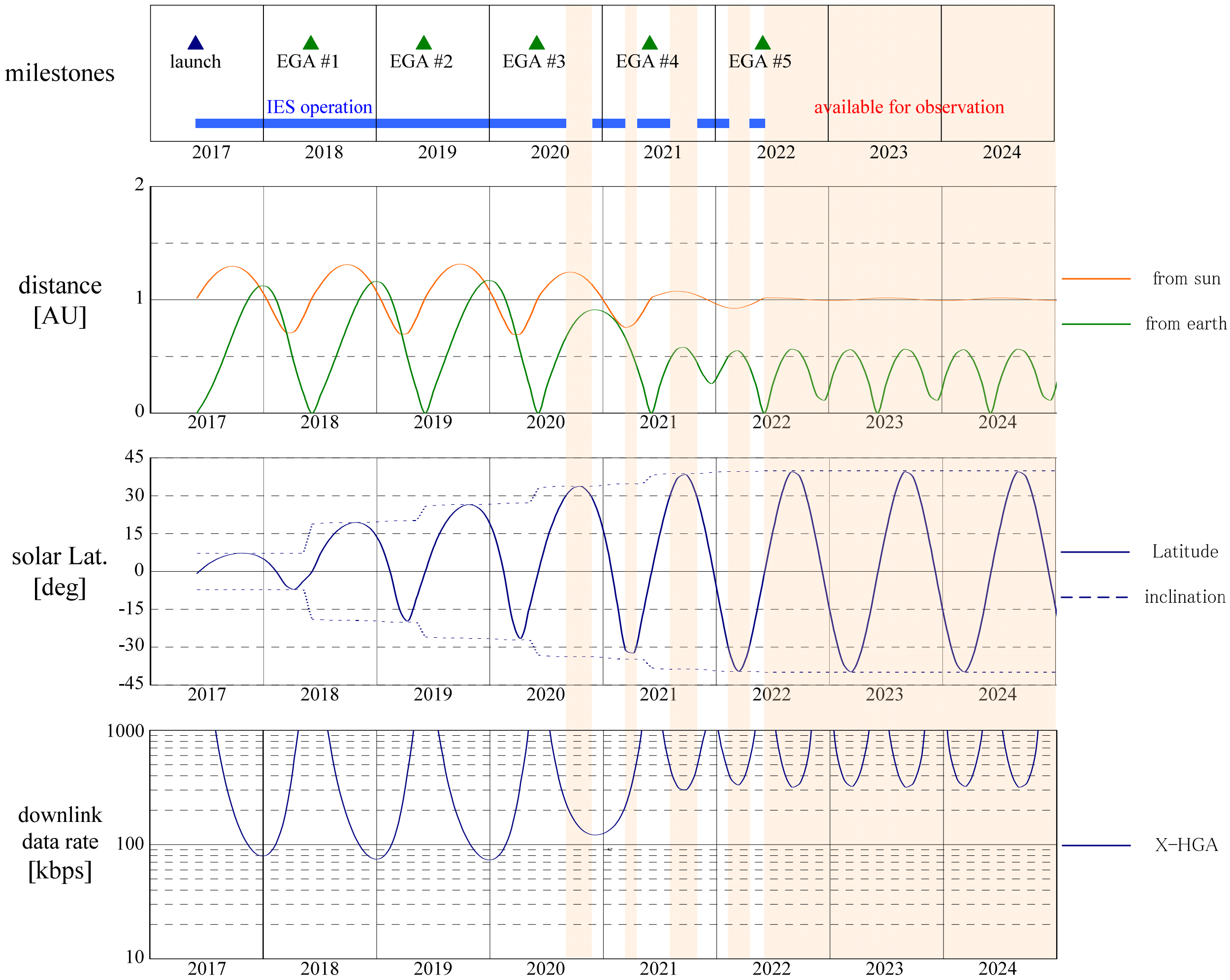


Figure 24. Mission profile of SEP option

In the first three years from the launch, before the spacecraft reaches i_{SEQ} of 30°, the spacecraft operation is devoted to increase the inclination. IES is operating most of the time, and silent condition for the scientific observation is basically not guaranteed in this phase. Even in this phase, intermittent suspension of IES is planned for accurate orbit determination, which may be used as occasions for observations. In the fourth year, after EGA #3, i_{SEQ} exceeds 30°. From this year onwards, the period during which the spacecraft is at latitudes higher than 30° is allocated as "the observation phase (orange area in Figure 24)", and the IES operation is intentionally suspended. Even after EGA #3, IES operation is continued while the spacecraft is in the lower solar latitude. This IES operation and the following two EGAs contribute to the further increase of the inclination and the circularization of the orbit. Finally, as a result of EGA #5, the spacecraft is injected in to the final observing orbit, the circular one-year orbit with i_{SEQ} of 40°.

3.3 Spacecraft System Design

The SOLAR-C Plan-A is the first Japanese solar mission that operates in the interplanetary space. The following new issues in spacecraft system design should be noticed when compared with the other Japanese (Earth orbiting) solar missions (a part of issues results from the mission design).
- It requires a much longer mission life.
- It is in severer radiation environments.
- It carries light payloads only.
- It requires large solar array panels
- It requires fuel to keep three-axis attitude control.

The following new issue from other Japanese interplanetary missions should be noticed as well.
- It requires much higher telemetry rate than ever achieved.

This item is already considered in the orbit design, and is also reflected on the communication system design. In addition, it should be noted that, as a result of using a liquid fuel rocket (H-IIA), the mechanical environment at the launch is relaxed from that in case of using solid rocket as previous solar missions.

3.3.1 Spacecraft Configuration

The mission design provides basic information of the spacecraft configuration. The required thrust determines the configuration of IES, and the configuration of IES determines the required power which finally determines the size of solar array panels (SAP). The orbit design provides the geometrical relation around the spacecraft during the mission, which determines the basic location of the key components. As a result, the basic configuration the spacecraft is determined as shown in Figure 25.

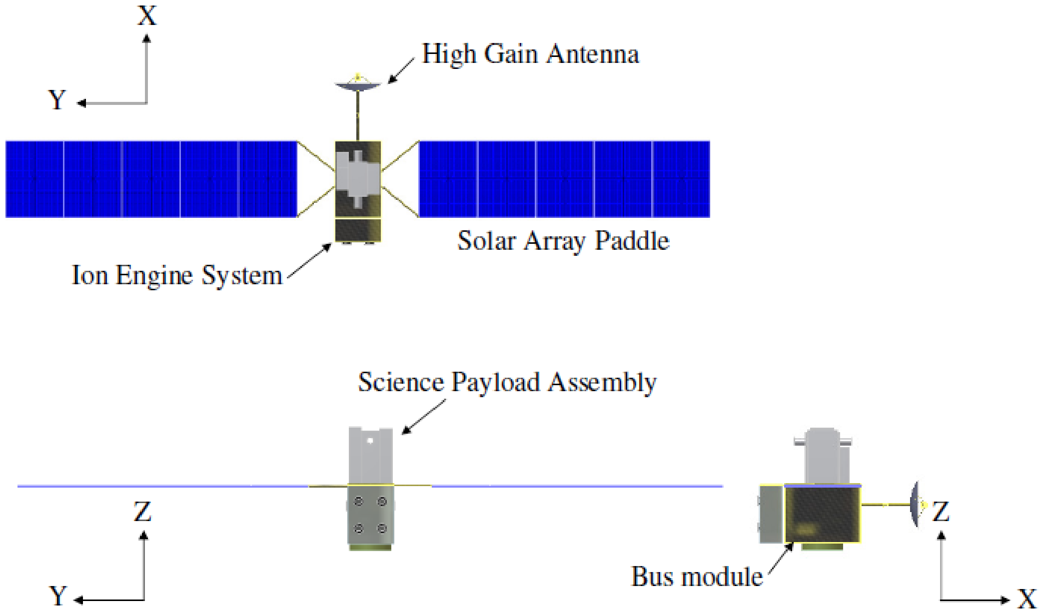


Figure 25. Spacecraft configuration of SEP option

The spacecraft's +Z-direction is always pointing to the Sun. The scientific payload assembly is mounted on the +Z panel of the bus module and pointing to the Sun. SAP are deployed from the bus module to the ±Y-direction and fixed to the body, facing the +Z-direction. In the trajectory design process, IES thrust direction is constrained to be approximately normal to the Sun direction, which enables IES module to be mounted on the –X panel. The panels not facing to the Sun direction (±Y, –X, -Z) are used as radiator panels to dissipate the large heat generated by IES components. The geometrical relation among the spacecraft, the Sun and the Earth varies as a function of time. Therefore, a high-gain antenna for the high rate data transmission must be maneuverable, and mounted on the +X-panel.

3.3.2 Mass Budget

In order to confirm the feasibility of the mission, the spacecraft mass budget is estimated. In the investigation, the total mass of the spacecraft is assumed to be 1,200 kg (which complies with the assumption of the trajectory design), and the difference with the sum of subsystem masses is counted as margin.

For most of the subsystems, the composition of the components is considered, and the mass of components is summed up to the subsystem mass, while the mass of subsystems such as structure, harness, etc. is estimated parametrically based on the previous spacecrafts. The propellant mass for IES is derived from the trajectory design result.

The resulting mass budget is summarized in Table 8. The positive margin of 106 kg indicates the feasibility of the mission from the point of the spacecraft's mass budget.

Table 8. Spacecraft mass budget of SEP option

Item	Mass (kg)
Total mass	1200.0
Dry mass	828.4
Mission payloads	130.0
Data handling system	6.7
Communication system	94.6
Power supply system	84.1
Attitude control system	79.9
Chemical propulsion system	36.5
Ion engine system	161.9
Structure / mechanical Int.	166.9
Thermal control system / Thermal Int.	14.8
Electric Int.	53.0
Propellant	265.9
for chemical propulsion	30.0
for ion engine system	235.9
Margin	105.7

3.3.3 Power Budget

In order to estimate the size and mass of SAP, the required power for the spacecraft is estimated. In the investigation, two cases of power budget (IES operating / non-operating) are estimated. The composition of the components is considered, and the powers for the components are summed up to the subsystem power. The resulting power budget is summarized in Table 9.

Table 9. Spacecraft power budget of SEP option
(Heater powers for payload not included.)

Item	Power (W)	
	IES op.	IES non op.
Total power	6065.7	1553.6
Mission payloads	0.0	150.0
Data handling system	12.3	12.3
Communication system	128.8	128.8
Power supply system	97.7	97.7
Attitude control system	165.0	165.0
Ion engine system	5637.0	0.0
Thermal control system	24.9	999.8

The mass of SAP is estimated on the basis of supplying the power for IES operating case at 1 AU. The use of newly developed light weight SAP is assumed, and the specific value of 100W/kg is used in the estimation. The resulting mass of SAP is included in the power supply system in the mass budget.

3.3.4 Communication Link Budget

In order to confirm the feasibility of the high rate data link, the communication link budget is estimated under the assumed communication system configuration. In the investigation, two cases of link budget (X-band / Ka-band) are estimated. The maximum distance of 0.56 AU in the final orbit is used, and NASA/DSN 34m antenna is assumed as the ground station. The resulting link budget is summarized in Table 10.

Table 10. Communication link budget of SEP option

Item	X-band	Ka-band	unit
Frequency	8400	32300	MHz
Transmitter power	40.0	20.0	W
Transmit antenna gain	36.5	48.2	dBi
Communication distance	0.56	0.56	AU
Data rate	300k	1M	bps
Link margin	1.0	2.0	dB
Received C/N0	59.3	64.6	dBHz
Required C/N0	58.4	62.6	dBHz

The positive margin of 1 dB for 300 kbps downlink indicates the feasibility of satisfying the system requirement in the final orbit. It should be noted that this downlink rate is achieved by more reliable X-band link. In addition, 1 Mbps downlink by Ka-band link is expected with positive margin of 2 dB.

3.3.5 System Thermal Design

In the cruise phase of SEP option, the minimum solar distance (i.e. perihelion distance) is to be 0.7 AU. It is assumed that three μ20 ion engines are simultaneously used even at the perihelion passage. The amount of heat dissipated from IES is estimated to be approximately 1,800 W in total, and it is not easy to radiate this amount of heat to space. In order to confirm the feasibility of the thermal design, system thermal analysis is conducted. In the analysis, the simultaneous use of the three μ20 ion engines at 0.7 AU is assumed as the worst case. The Sun-light is assumed to input from the direction 5° apart from $+Z$ direction, which causes the heat input from the Sun into the side panels of spacecraft.

The thermal mathematical model consisting of the spacecraft bus module and ion engine module is constructed, and a steady state thermal analysis is conducted. The result shows that we can still have a feasible solution under this worst condition. The solution requires that concentrated heat dissipation from the ion-engine related components to be averaged at each structure panel by heat pipes and thermal doublers.

3.4 Spacecraft Subsystems

The spacecraft subsystems are discussed in this section. At this stage of the interim report, the investigation gives priority to the subsystems that require new key technologies to achieve this mission. They are Ka/X communication system, ultra-lightweight solar panels, and μ20 ion engine system. The details of these subsystems are given in Sections 3.4.2, 3.4.3 and 3.4.6 respectively. As for the other subsystems, the functions required to them are supposed to be achieved with proven technologies. Their compositions are tentatively assumed based on the previous missions, and they are only briefly touched in this report.

3.4.1 Data Handling System

The data handling subsystem are supposed to be composed based on the heritage of the previous interplanetary spacecrafts, *Hayabusa* or *Akatsuki*. It is composed of DHU (data handling unit) and TCIM (telemetry command interface module). DHU manages the automatic/autonomous function of the spacecraft bus system, collects and records the HK (house keeping) data, and manages DH network as the host. TCIM interfaces telemetry and command between DHU and the communication system. It generates the transfer frame and multiplexes the telemetry packets, and output to the transmitter. On the other hand, it modulates the uplink signal input from responder, and output command packets to DHU.

3.4.2 Communication System

Since the spacecraft is in the interplanetary space for the out-of-ecliptic mission, a high-speed telemetry is mandatory for imaging observations of the Sun. We have tentatively set a straw-man data recording rate of 100 kbps in X-band considering the continuous helioseismic observations. This level has been achieved in the NASA STEREO mission in the distance of 0.5 AU from the Earth. On the other hand, the Japanese interplanetary missions that have so far been flown only achieved the telemetry rate of less than 10 kbps using 64m-aperture downlink station at Usuda deep space center. When the downlink duration is limited to 8 hours per day for example, 300 kbps telemetry rate is required for 100 kbps recording rate. If this should be achieved by a domestic effort, a development of a high-power transponder will be required in addition to reducing any transmission losses.

Based on the consideration above, the following design guideline is derived.
- Achieve 300 kpbs downlink rate assuming the downlink duration of 8 hours per day.
- Use X-band system heritage in interplanetary communication for stable and assured downlink.
- Use Ka-band system for a higher-telemetry capability under good atmospheric conditions.

- Adopt domestic supplied components which we can have by around year 2012.
- Adopt a high gain antenna of 1.6 m diameter.
- Adopt a high-power (40W) X-band transmitter as a natural extension from the JAXA heritage.
- Reduce feeder loss compared with the previous JAXA interplanetary missions.
- Improve the coding gain by the modification of coding method.

The block diagram of the communication system (with the heritages) is shown in Figure 26.

In order to have a higher data transmission rate or a shorter downlink duration, the use of Ka-band system is supposed in addition to the X-band system. The downlink of Ka-band telemetry is possible at the NASA DSN stations to track the STEREO spacecrafts. Since the availability of Ka-band is affected by the atmospheric condition on the Earth, the Ka-band alone cannot be a solution to the requirement set by the continuous helioseismic observations.

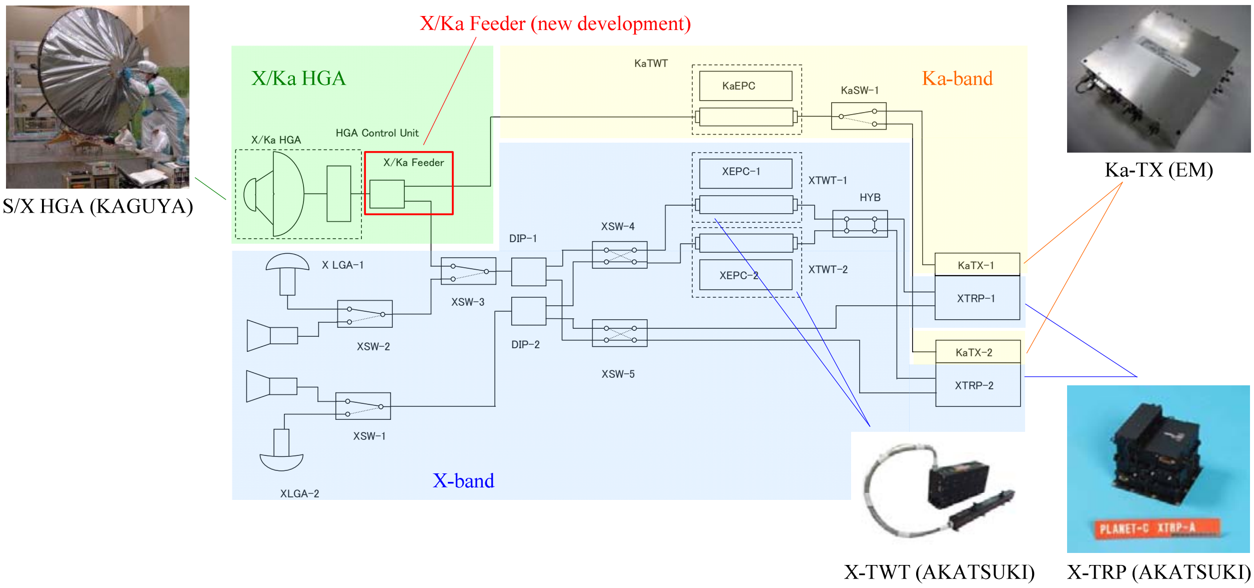


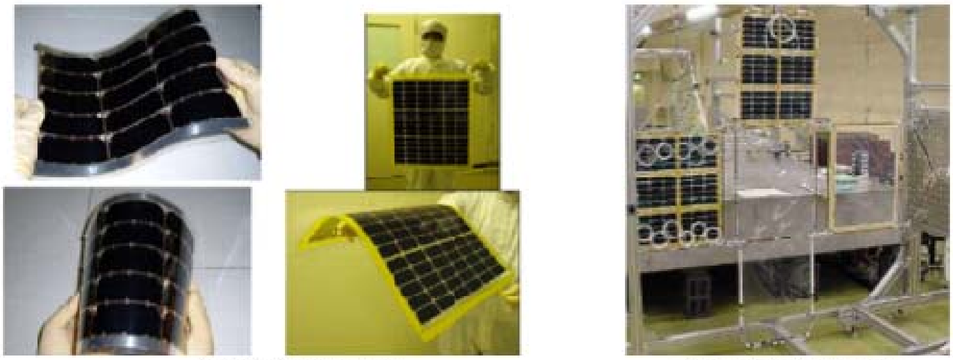
Figure 26. SOLAR-C Communication System

3.4.3 Power Supply System

Power supply system of SOLAR-C Plan-A consists of Power Control Unit (PCU), Series Switching Regulator (SSR), Solar Array Paddle (SAP), Battery (BAT), and Pyrotechnic Controller (PYC). In the system, the output voltage of SAP is lowered and stabilized by SSR. The functions of PCU are SSR control, BAT charge/discharge control and management, and the bus power distribution. BAT is used for power supply when SAP cannot supply sufficient power. The cases supposed are, the launch phase, the attitude anomaly, and the large attitude maneuver. The use of Lithium ion battery is supposed based on the heritage of the previous interplanetary spacecrafts, *Hayabusa* or *Akatsuki*.

SOLAR-C Plan-A spacecraft is propelled by the ion engine system. In order to operate three μ20 ion engines simultaneously, approximately 6 kW of power is required to be supplied by SAP. When conventional rigid type SAP is adopted, its mass is estimated to exceed 100 kg. On the other hand, the reduction of components' mass is strongly requested as a general nature of interplanetary spacecraft. From this consideration, the newly developed ultra-lightweight solar panel is used to meet the requirements of high specific power (W/kg) and low storage volume. This component is identified as the key technologies to achieve this mission.

The ultra-lightweight solar panel is now under development in ARD/JAXA, and uses newly developed space solar sheet supported by a frame-type structure (Figure 27). It is targeting the specific power greater than 100W/kg in panel level. Future perspective of the ultra-light weight solar panel development is shown in Figure 28. Breadboard model was developed last year, and a series of development tests (vibration, thermal vacuum, and deployment test) were successfully conducted. The technical readiness level (TRL) is estimated to be 6+α.



(a) Space solar sheet (b) BBM deployment

Figure 27. Ultra-Lightweight Solar Panel

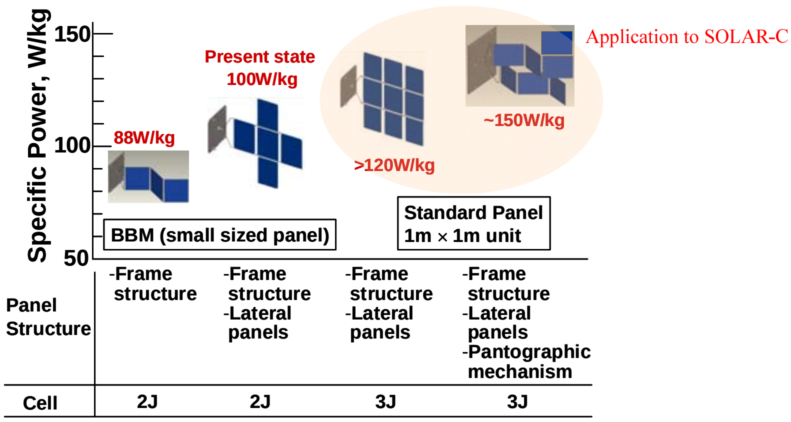


Figure 28. Future Prospects of Ultra-Lightweight Solar Panel Development

As mentioned in Section 3.3.1, in the configuration of this spacecraft, Z-axis (the direction perpendicular to SAP) is basically pointed toward the Sun. Therefore, SAP is fixed to the spacecraft body, and no driving mechanisms are installed.

Li-ion battery is adopted as a high-density power cell in the Plan-A spacecraft. The overcharge and over-discharge protections are dealt by the Series Switching Regulator (SSR) that has been developed in the JAXA MMO mission. The bus voltage regulation is made by SSR monitoring the electric current at BAT. The period in which the spacecraft is operated by the power from BAT is

(1) during initial phase just after launch, (2) during orbit control without using the ion-engine system, and (3) during the safety state in an attitude control for emergency cases. The eclipse period in which the Sun cannot be seen does not happen in the SOLAR-C Plan-A orbit except for the launch operation. The capacity of battery is selected by these conditions and it is found to be about 27 Ah (TBC) in the first estimate.

3.4.4 Attitude Control System

The requirements for the attitude control are summarized in Table 5. The three-axis stabilization is required for imaging observations of an arcsec spatial resolution. The attitude control system that is tentatively assumed here is what was used in *Hinode* mission. Since the short-term performance of the *Hinode* attitude control system completely satisfies the SOLAR-C Plan-A requirements, no improvement is necessary. The external disturbance in the interplanetary space is much smaller and more stable than that in the low-Earth orbit, so that the main source of disturbance is the mechanical motions within the spacecraft. One of the differences between SOLAR-C Plan-A and the Japanese solar missions in low-Earth orbits is the use of chemical thrusters to unload the angular momentum accumulated in the reaction wheels (RW). According to a simple calculation, the frequency of the RW unloading is estimated to be once per day in a nominal science operation. The helioseismic instrument may need to have its own image stabilization tip-tilt mirror to compensate the long-term pointing change or to stabilize the unexpected low-frequency disturbance.

3.4.5 Chemical Propulsion System

Chemical propulsion subsystem is used for unloading of the RW accumulated angular momentum and the attitude maneuver which requires large control torque (such as initial attitude establishment and safe hold mode operation). The thrusters are also used for the short term translational motion control such as the trajectory correction before the Earth gravity assists. Considering the level of the total impulse required, the conventional monopropellant (N_2H_4) system is assumed at the present phase.

3.4.6 Ion Engine System

Owing to the success of Japanese asteroid explorer HAYABUSA, which has an ion-engine μ10 system for interplanetary cruise, ISAS/JAXA has a heritage of ion engines. ISAS is developing the scale-up version of the ion engine thruster, μ20. Figure 29 shows the engineering-model thruster, which is in an endurance running test under a vacuum condition. We assume a SEP system in which four sets of ion engines are set on the spacecraft and three out of four can be turned on simultaneously. In the orbit design we assume the thrust of 40 mN and specific impulse of 3,800 s for each thruster or the maximum thrust of 120 mN for simultaneous use of three thrusters.

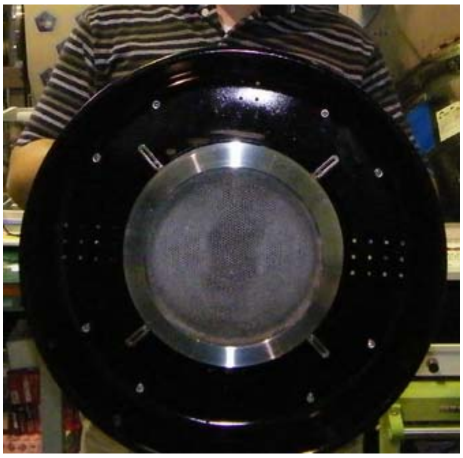


Figure 29. Engineering model of μ20 ion engine thruster

The μ20 ion engine system consists of Ion Thruster Control Unit (ITCU), IES Power Unit (IPU), Propellant Management Unit (PMU), Ion Thruster Assembly (ITA), Microwave Supply Unit (MSU), and IES Pointing Mechanism (IPM). Xe is the propellant of this engine. Xe ions are produced in a process of discharge by microwaves and are accelerated by electric fields between two grids. The role of each unit is summarized as follows:

IPM: This is a pointing mechanism of ITA.
IPU: This produces a high voltage (~2.3 kV) to accelerate Xe ions.
ITA: This unit produces thrust from Xe propellant, microwaves, and high-voltage applied.
ITCU: This unit controls the whole ion-engine system (IES).
MSU: This unit supplies microwaves to ionize Xe for acceleration and to make electrons for neutralization.
PMU: This unit provides Xe propellant to ITA.

Each ion engine thruster produces the thrust of 40 mN. By simultaneous use of three μ20 ion engines, one of which is currently being developed and tested at ISAS, the maximum thrust of 120 mN is obtained. Some components are set on the spacecraft bus structure, and some are set on the Ion Engine Module (IEM) that is an independent box located outside the spacecraft bus structure. IEM has six ion engine thrusters, two of which are redundant engines for unexpected situations. The IPM connects IEM with the spacecraft, and it also acts as a pointing mechanism that slightly tilts the whole IEM to control the direction of thrust under combinations of multiple ion engine thrusters, each has a slightly different thrust and is located at a different distance from the center of gravity of the spacecraft.

Power Control Unit (PCU) on the spacecraft provides the power of 350 W (1,520 W) with MSU (IPU) for operating each ion engine thruster. 1,870 W electric power is required for operating a single ion engine thruster. It was 350 W for *Hayabusa*'s μ10 ion engine system. Larger power consumption in the SOLAR-C Plan-A system gives large heat generation that needs to be radiated away from the spacecraft. The ion engine system in the Plan-A spacecraft produces a huge amount of heat that cannot be radiated away from the surface of IEM alone. In order to have a solution for the heat dissipation problem, some components, ITA and MSU, are set in IEM, and other components, ITCU, IPU, and PMU, are mounted in the spacecraft bus structure.

In the following points the system is different from *Hayabusa*'s ion-engine system. These are all related to the increase in weight or heat dissipation to achieve high thrust in good propellant efficiency.

1. Size of ITA from 10 cm diameter in *Hayabusa* to 20 cm diameter in SOLAR-C Plan-A
2. Operating power for ion acceleration
3. Pointing mechanism at IPM

Currently estimated specifications summarized in Table 11 assume higher specific impulse options that will soon be applied to the EM thruster under endurance test. Original test conditions of the EM thruster's ion source were 1,300 V (Screen Voltage), 30 mN (Thrust) and 2,800 sec (Specific Impulse), respectively, and it accumulated 10,000 hours of operation by Sep. 2010. The technical readiness level (TRL) is estimated to be 4-5.

Table 11. SOLAR-C Plan-A Ion Engine System

Propellant	Xe	
Specific impulse	3800 s	
Thrust	40 mN/engine	
Screen Voltage	2300 V	
Number of thrusters	4	
Maximum number of thrusters that run simultaneously	3	
Size of Ion-Engine Module	1,200×1,500×600 mm	
Pointing mechanism	Newly proposed linear actuators Range: ±5°	Need a development
Weight	162 kg	Estimate in 2010
Weight of Fuel (Xe)	236 kg	Estimate in 2010
Power	5,650 W Regulated bus: 1,090 W Non-regulated bus: 4,560 W	Estimate in 2010

3.4.7 Structure System

A box-type panel structure is adopted as the spacecraft main body (bus module). The approximate size of the main body is 2m×1.2m×1.5m. The scientific payload assembly is mounted on the +*Z* panel of the bus module, and the rocket attachment ring is installed on the –*Z* panel. The movable IES module is installed on the –*X* panel. Two axes pointing mechanism enables to direct the IES thrust axis to the spacecraft center of mass. The major deployable appendages are two wings of solar array panels and Ka/X-band high gain antenna. The solar array panels are installed on the $\pm Y$ panels, and they are latched and fixed to the body after deployment. Two axes gimbaled movable high gain antenna is installed on the $+X$ panel, and is deployed after the launch.

3.4.8 Thermal Control System

SOLAR-C Plan-A spacecraft is propelled by the ion engine system. When three μ20 ion engines are simultaneously operated, the heat dissipated from IES is estimated as approximately 1,800W in total. The major theme for the thermal control system is to radiate this large amount of heat to space. In order to achieve this objective, large areas on the structure panels that are not facing with the Sun ($\pm Y$, $-X$, $-Z$) are used as radiator panels. Concentrated heat dissipation from IES is transferred and averaged at each structure panel by heat pipes and thermal doublers. The area outside the radiators on the structure panels is covered with the multi layer insulator (MLI), to interrupt unnecessary heat input/output.

When not all the ion engines are operated, the reduced heat dissipation from IES is compensated by heaters. The heater control electronics (HCE) automatically controls the duty of the heaters based on their priority. It enables the appropriate management of the power allocation between thermal control and IES operation even when the available power is limited around the aphelion.