Research on Launch Vehicle Interface for Interplanetary Mission

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Abstract: *Japan Aerospace Exploration Agency (JAXA) launched Akatsuki and IKAROS on 21st of May 2010 into space using H-IIA launch vehicle. This event marks the use of H-II A launch vehicle for interplanetary mission. Despite the launch was successful, further improvements are still planned to address the inflexibility of launch window and to take advantage of launch capability. These improvements are specifically aimed to better serve future interplanetary missions. Research is currently undertaken to achieve the improvements for H-IIA/B upgrade type (planned to be released in 2013) and H-IIA/B Evolution (planned to be released in latter half of the 2010s). The important point is collaborative activity between launch vehicle and spacecraft because of the complexity for interplanetary mission. In this paper, we introduce the research progresses have been achieved so far on the interface between spacecraft and launch vehicle in resolving the issues.*

Keywords: *Launch vehicle, Interplanetary mission, Interface, Trajectory design.*

1 Introduction

Interplanetary mission take more complicated interface between spacecraft and launch vehicle. For example, the target orbital elements to insert would be changed time by time, which require the on-time launch and the change of flight sequence depend on launch delay, and the possible days to launch would be restricted because of the location between the departure and the arrival planet. On the other hand, sun-synchronous orbit (SSO) and geostationary transfer orbit (GTO) mission usually take over half an hour launch window per a day, and from half to a month per a season depend on the mission. Therefore, interplanetary mission need much more research to realize enough launch window and flexible operation, than SSO and GTO.

Japanese flagship launch vehicle is H-IIA/B launch vehicle (Fig. 1). The Launch Vehicle is designed with two stages, each powered by engines using liquid hydrogen and liquid oxygen propellant. The H-IIA/B consists of the first stage which diameter is 4m or 5.2m, two or four SRB-As, the second stage which is 4m and the payload fairing. Since maiden flight of H-IIA is performed at 2001, there is only 1case (H-IIA-F6) to failure within 20 flights. This indicate the high reliability of H-IIA/B. In case of interplanetary mission, H-IIA-F17 launched Akatsuki and IKAROS on 21st of May 2010 at first time. But, some inflexibilities are detected to operate H-IIA at interplanetary mission. Furthermore, there are few people to understand interplanetary mission at launch vehicle side which are similar to understand H-IIA at spacecraft side. These situation is not so good to take interplanetary mission design and optimization (e.c.:about cost, weight and time).

On the other hand, many interplanetary missions are under consideration in Japan. For example, Hayabusa-2 which plan to visit a C-type asteroid JU3 and return to earth, Melos which plan to visit the Mars, SOLAR-C_planA which plan to escape the plane of the ecliptic and SPICA which plan to visit Sun-Earth L2 halo orbit and observe with space infrared telescope. These missions must be launched by Japanese launch vehicle to indicate the autonomy of space program.

As above, the research on launch vehicle interface for interplanetary mission is necessary to succeed the future interplanetary mission effectively. As it is important for constructing good interplanetary mission to understand each other constraint and leave behind the techniques, the research is proceeding with developing a novel interplanetary design system. Furthermore, there are two system design development for H-IIA/B which are H-IIA/B upgrade type (planned to be released in 2013) and *H-IIA/B Evolution* (planned to be released in latter half of the 2010s). As the research should be tried by systematic point of view, we try to research in accordance with these project as far as possible. The framework of this paper is below.

Type	H-IIA202	H-IIA204	H-IIB
Payload Capability*	4.0 ton	6.0 ton	8.0ton
Gross Mass	280ton	410ton	550ton
Diameter	4m (1st & 2nd stage)		5.2m (1st stage), 4m (2nd stage)
Number of Solid Rocket Booster	2	4	4
Fairing	4m diameter		

*) GTO: ha=35976km, hp=250km, inc=28.5deg, ω=179deg

Figure 1. H-IIA/B launch vehicle

At first, some challenges of H-IIA/B for interplanetary mission, which is focused on our research, would be extracted and the research status for them are introduced. Furthermore, in this paper we explain a novel interplanetary design system which is purposed to make interplanetary mission design becomes more efficient. The final goal of this system would be the construction of new interplanetary mission, which is optimized from launch vehicle phase to the orbit of arrival planet, and the proposal after confirming the technical possibility and advantage. We call this system Interplanetary Route Integration System (IRIS). Finally, trial constructions of interplanetary mission with IRIS are presented for Jupiter and Mars.

2. Improvement of Interface for Interplanetary Mission

After general trajectory design procedure of H-IIA/B for interplanetary mission is explained, three inflexibilities on launch vehicle interface would be introduced. From 2.3 to 2.5 section, we figure out how to handle these matters.

2.1 General Trajectory Design Procedure

At first, the arrival parameters of spacecraft for interplanetary mission are the time, the radius and the inclination when the spacecraft arrives at the periapsis of the arrival planet. These parameters are set in consideration mainly of minimizing the propellant consumption for spacecraft and of link conditions from the earth. At the same time, the departure parameters, which are Declination ("Dec") , Right Ascension ("RA") and "C3", would be settled. Then, the sets of Dec and C3 would be restricted by the payload capability of launch vehicle, where the payload capability for individual C3 are prepared for each Dec.

For the departure parameters, the flight sequence of H-IIA/B until second engine 1st cutoff is fixed for efficient interface with spacecraft and operation of H-IIA/B, where the vehicle is injected into parking orbit (e.c.:altitude=300km, inclination=30deg). The difference of departure orbital elements (Dec, RA, C3) depend on the launch timing would be optimized by three parameters of H-IIA flight sequence, which are “the lift off time”, “the coast time” between two burn phase of second engine and “tangential velocity increment” at second engine 2nd burn. Currently, as H-IIA/B can only takes one chance per a day to launch these missions for the spec of on-board software (OBS) configuration and operation, the second engine 2nd ignition points of H-IIA/Akatsuki mission are different depend on the launch day like Fig. 2. The mission takes continuous launch windows from 19th May to 3rd June, but, for simplification, only four days are indicated on this figure.

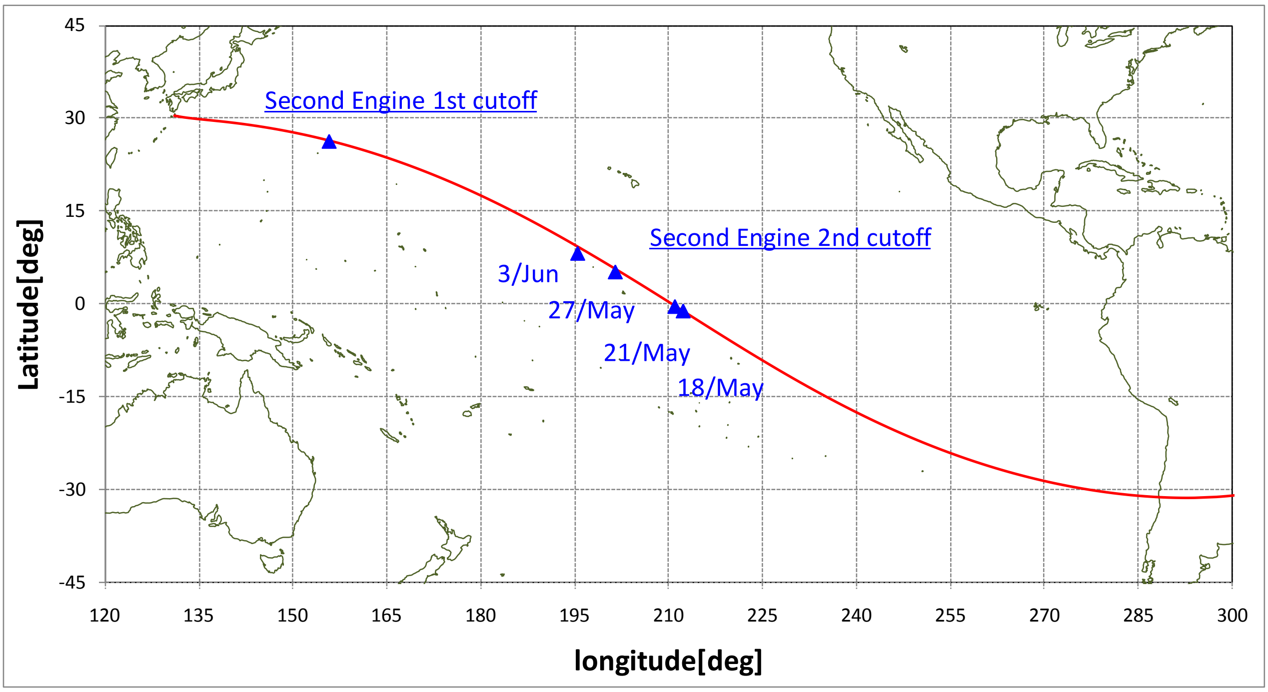


Figure 2. Second engine burn out points on ground track of H-IIA/Akatsuki mission

2.2. Challenges on H-IIA

There are three challenges to launch spacecraft into interplanetary orbit with H-IIA. They are lack of launch capability, precise instant launch window and inflexibility for changing launch day. Each substance would be introduced below.

The launch capability is restricted mainly by flight safety constraint and mission time. The flight safety constraint mainly restrict the flight azimuth around TNSC, the dropped area of fairing and first stage, and the over-all flight trajectory for radio frequency (RF) link from ground station (GS) and for expectation casualty (Ec). The flight azimuth is limited from 90deg to 115deg, from the north, by the narrow launch range of TNSC. For typical mission like GTO, SSO and HTV, instantaneous impact points (IIP) with rough sketch of dropped area are indicated at Fig. 3. There are many island and island chain near IIP like Mariana, Carolyn, Marshall, Palau, New Guinea, Australia and New Zealand for the South-East of TNSC. For the restricted flight azimuth and avoiding the interference with islands, the flight trajectory would be bent especially at high inclination mission, e.c. SSO. Therefore, loss of capability would be exhibited by comparison with potential capability of H-IIA configuration. So, H-IIA does not have enough capability especially for high inclination (>31deg), which means Dec<-31deg and 31deg<Dec.

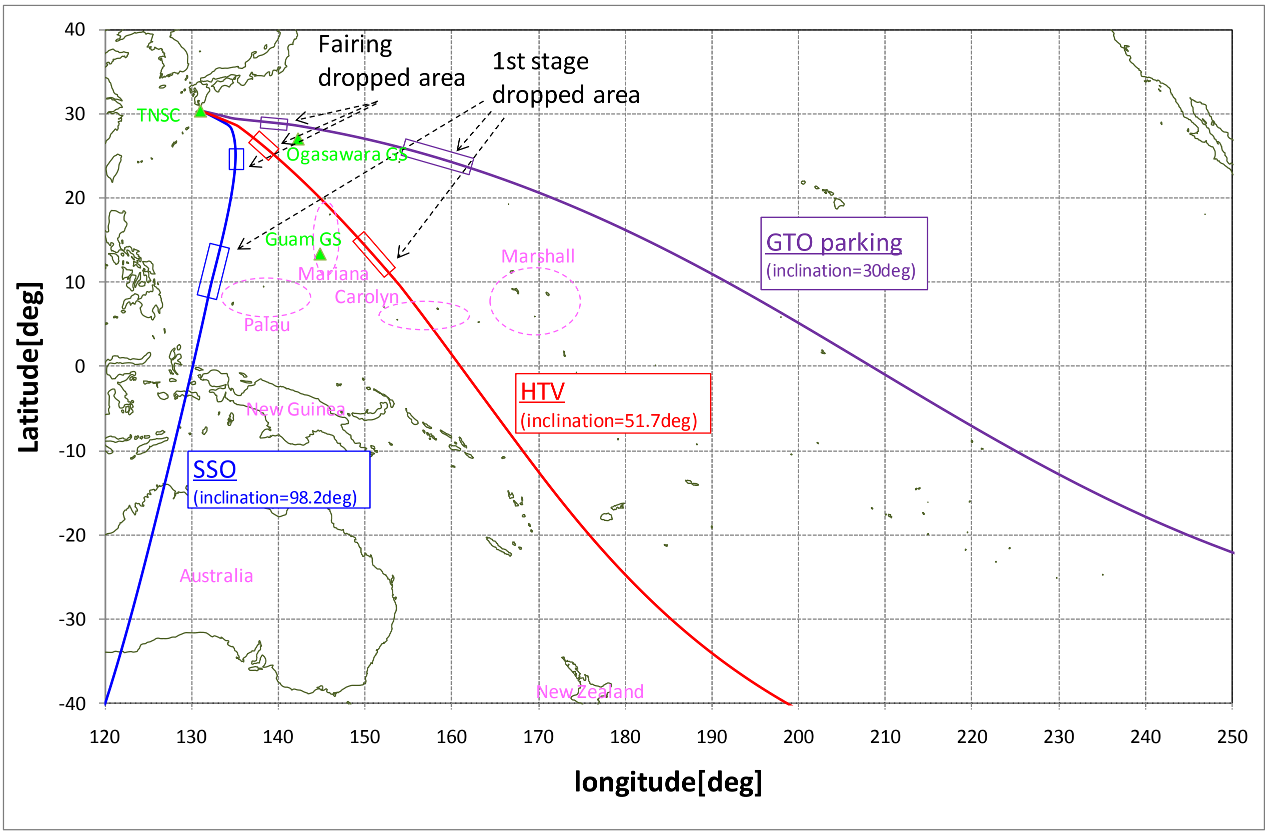


Figure 3. IIP with rough sketch of dropped area

The mission time of H-IIA is limited for 3000 seconds, which is decided by battery, radiation tolerance of avionics system and so on. Therefore, if the requirement orbital parameter need over 3000s, H-IIA cannot launch. For example, if the demanding Dec is -65deg, H-IIA have to ignite second engine 2nd burn at approximately 65deg north latitude, when it pass 5100s after launch. How we handle the matter and its result are discussed in section 2.3.

Second, when used for interplanetary mission, H-IIA unfortunately has restricted the opportunity of liftoff. The liftoff time is limited only to a precise instance of time. Considering the geographic characteristic of Tanegashima Space Center (TNSC), there is a high possibility that H-IIA launching could be delayed because of poor weather condition especially lightning. For comparison, Delta-II/Phoenix launch vehicle has a pair of instance per day. Therefore, despite the possibility of installation, the technical feasibility of launch window extension must be researched to reduce the total launch cost. How we handle the matter and its result are discussed in section 2.4.

The last, if scheduled launch day is delayed, the mission parameter of OBS have to be changed. That means the need of preparing many trajectory, analysis and OBS for each launch day and replacement of OBS at TNSC. For example, there are 17cases at H-IIA/Akatsuki mission. That lead to increase of launch cost and interface time in comparison with SSO and GTO. How we handle the matter and its result are discussed in section 2.5.

2.3. Update of Launch Capability

In terms of flight safety constraint, the control of 1st stage dropped area is focused because of the big effect on launch capability of high inclination. At first, the place of the dropped area would be researched for avoiding many islands. The control of cross range direction can be shifted with flight azimuth and yaw rate maneuver after atmospheric flight phase. But the control from optimized flight trajectory without flight safety constraint lead to much loss of launch capability. On the other hand, downrange direction can be shifted with altitude of parking orbit, where the low altitude lead the area to TNSC side. The descent from 300km altitude parking orbit lead to increase of launch capability, but that also lead to worse of thermal condition and RF link from GS (positions are Fig. 3). Therefore, we are trying to make optimized flight trajectory with maximum capability at individual Dec & C3, where control variables are flight azimuth, yaw rate maneuver and the altitude of parking orbit with the constraints of flight safety, thermal condition and RF link. Second, the size of the dropped area would be researched, which is ongoing now and only the principle to solve is introduced in this paper. For decrease of the size, we try to research the reduce of error and command shutdown of 1st stage engine which are usually depletion shutdown. The former is being undertook with the past result of H-IIA/B. The latter could reduce the size to half for down range direction, but the risk of failure for launch vehicle may be increased. Therefore, we will have to make fault tree analysis (FTA) to calculate the increase level of launch failure possibility. For example, Fig. 4 indicate the IIP of rough trajectories for C3=20, Dec=65deg, inclination=65deg and H-IIA. In this case, caseA is no constraint trajectory through 350km parking orbit and get 900kg payload, but 1st stage dropped area interfere Carolyn islands and Mariana islands. At caseB, we tried to change IIP course for west direction to avoid Mariana islands, but Carolyn islands is still interfered and launch capability is decreased to 400kg. On the other hand, when the altitude of parking orbit is down to 100nm (about 185km), the avoidance of 1st stage dropped area with islands is succeeded. Furthermore, the capability is 900kg. As these flexibility would be important to increase the capability, over-all system confirmation for these case would be done as soon as possible.

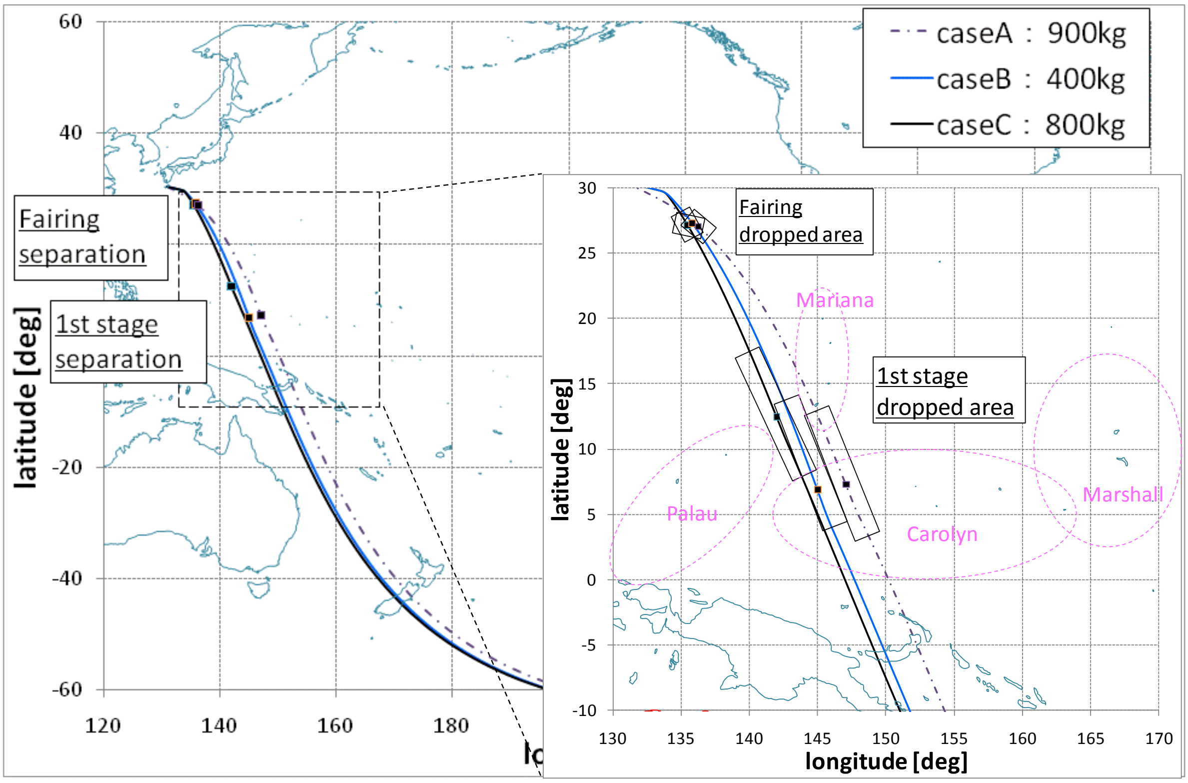


Figure 4. IIP for C3=20, Dec=65deg and H-IIA

The mission time matter are took place in accordance with "H-IIA/B upgrade type " program. In the program, there are various system development item to extend the mission time to 20,000s covering direct geostationary earth orbit (GEO) mission with transition from 250km to 35786km circular orbit by Hohmann transfer. The program mainly focuses the second stage of H-IIA. For example, the reduce for evaporation of LH2 and pre-cool LOX of propulsion system are developed for increase of launch capability. And avionics and attitude control propulsion system are also developed to resist the long time mission. Therefore, after 2013, when the program will be completed, the interplanetary mission for every Dec would be possible.

2.4. Extension of Launch Window

In case of Delta-II, their flight azimuth from launch pad are two cases to extend launch opportunity, which are 93deg and 99deg. For example, the launch window of Delta-II/Phoenix mission at 3/Aug were 5:35 at 93deg and 6:11 at 99deg. The switch of flight azimuth can control the time from lift off to final velocity increment of launch vehicle. As the interplanetary mission require the increment approximately at opposite side of escape direction, the case of 6:11 is designed to arrive same latitude faster than 5:35, which could be understood by the comparison of arrival time from TNSC (latitude=30deg) to equator between inclination 30deg (flight azimuth=90deg) and 90deg (flight azimuth=180deg). Therefore, with the control of flight azimuth, if liftoff time would be delayed, Delta-II can take similar orbit interface as nominal case.

On the other hand, H-IIA takes the same parking orbit as mentioned section 2.1 and doesn't have such flexibility of flight azimuth and trajectory for geographical situation of TNSC, as mentioned section 2.2.

However it is possible to extend the launch time per a day for the H-IIA by changing the direction of second engine 2nd burn velocity increment from tangential line. In this case, "the lift off time" is uncontrollable among three control parameters as mentioned section 2.1. Therefore, three control parameters would be "the coast time", "tangential velocity increment" and "orbital angular momentum direction of velocity increment". The launch windows with plural instants or duration can be performed by such procedure even if only one kind of parking orbit is prepared. With case of 26/May departure and 5/Dec arrival for H-IIA/Akatsuki mission, the change of orbit designs between TNSC and Venus for lift off time range from -60min to 60min are presented on Fig. 5. On the left side figure, departure points (TNSC) are changed from west to east in inertial space. The velocity increment points of H-IIA, which are changing point between aqua and purple color line, are lined in inertial space. Therefore, the coast time is shortened with launch delay. On the right side figure, although Venus orbit insertion (VOI) of all cases are indicated, they are almost crossover. Therefore, if the lift off time would be changed, the orbit design of same arrival time is possible.

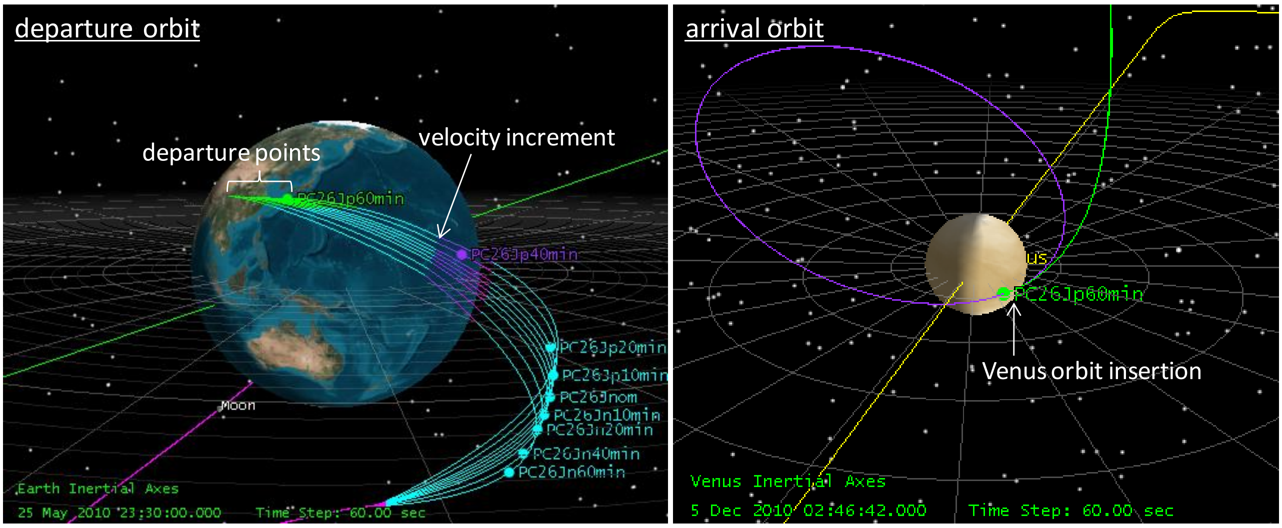


Figure 5. Change of orbit designs between TNSC and Venus for lift off time

Then, the effect on launch vehicle and spacecraft are studied on H-IIA/Akatsuki mission. The left side of Fig 6 shows the propellant consumption of second stage 2^{nd} burn related to the changes of the lift off time. In case of 20 minutes, the propellant is increased about 20kg which means decrease of 20kg spacecraft weight. As Akatsuki's mass is 510kg and, the effect is not so big until these change level for H-IIA. The right side of Fig 6 shows the propellant consumption of Akatsuki at VOI related to the changes of the lift off time. That is 70g which is only 0.06% of nominal plan, if the lift off time is changed 60minutes. Therefore, the change of lift off time would be negligible for spacecraft.

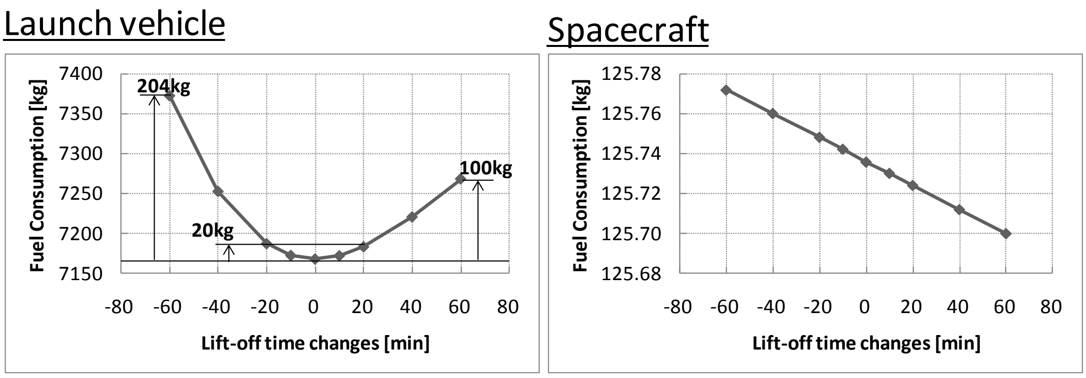


Figure 6. Effect on launch vehicle and spacecraft

Finally, the application for H-IIA would be studied. For technical feasibility, H-IIA has the capability called real time targeting guidance (RTG). That approximate subsets of guidance parameters with cubic function of delay time from nominal lift off time. Using this function, H-IIA can be launched without changing mission parameters at any time in the specified range. But a few upgrade of this function have to be done for applicable function. On the other hand, cost-effectiveness have to be studied. In parallel with development and operational cost of launch vehicle, the decrease possibility of launch delay would be researched for picking up the postponed mode of launch operation within 20minutes. These will be future research.

2.5. Flexibility for Changing Launch Day

For decrease of launch cost and interface time, we try to reduce the number of flight trajectory to design, which are for every launch day right now, and launch with approximated trajectory in OBS for the day without prepared flight trajectory to design.

At first, when the departure day would be changed, the target orbital elements are different. Therefore, the controll parameter would be divided whether it could be modified by guidance module of OBS. Then, besides the target orbital elements, two parameters were set by function of launch day, as uncontrollable parameter with guidance module of H-IIA. They are the coast time and the attitude of second stage 2nd ignition. On the other hand, it is possible to control the attitude rate and engine cutoff timing with guidance module of H-IIA. On that basis, the number of omitted days are expected to decrease 50% at least.

Further research are ongoing.

3. Interplanetary Mission Design Tool

IRIS core platform system is constructed from three modules (Fig. 3). They are “Interplanetary design module” which calculate needful delta-V to take orbital transfer between departure planet and arrival planet, “Launch vehicle capability module” which indicate launch capability for earth escape mission of variety launch vehicle and “Adjust maneuver module” which calculate necessary delta-V to make adjustment of launch vehicle error for transfer orbit. With this system, when launch vehicle, impulsive method and adjust method are decided, the maximum “wet weight” and necessary “propellant weight” of spacecraft for each departure and arrival day could be induced. That lead to conceptual system design of new interplanetary mission. Individual modules are introduced as follows.

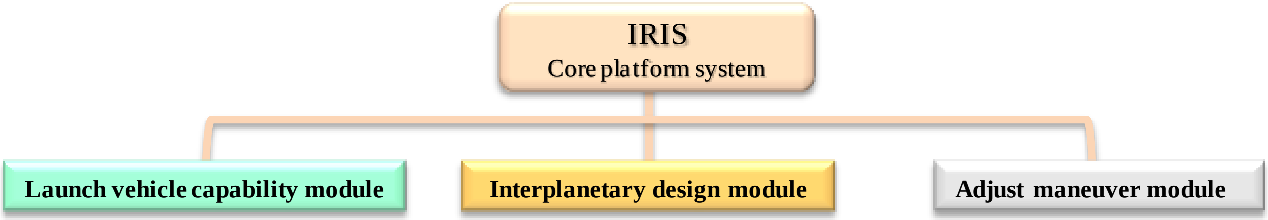


Figure 7. INTERPLANETARY ROUTE INTEGRATION SYSTEM (IRIS)

3.1 Interplanetary Design Module

As shown in Fig. 8, interplanetary design module provides two options: impulsive transition similar to Akatsuki mission and low-thrust transition similar to Hayabusa mission.

On the impulsive transition sub module, the trajectory is designed by three steps. At first, the transfer orbit between departure and arrival planet are solved as two-body problem of sun and spacecraft. Each given set of departure day and arrival day could derive rough trajectory, because of given departure position, arrival position and transfer time. That is famous for Lambert's problem. At the second step, the tool designs the trajectory inside individual planet's gravitational sphere of influence (SOI). Then, its trajectory design is treated as two-body problem of the planet and spacecraft, and the two trajectories are connected by the Patched Conic Approximation. At this point, rough design of interplanetary orbit is finished. At the third step, the trajectory are integrated from departure point, which is parking orbit of launch vehicle, by taking main perturbations into account.

While for the low-thrust transition sub module, the procedure is similar the above sub module. But, if departure and arrival day are decided, there are many cases of transfer orbit between departure and arrival planet depend on the operation timing of thruster. Therefore, the optimization of propellant consumption would be performed with given hardware spec, which include ion thruster and hall thruster, and relative velocity of departure and arrival. In addition, as the optimization is difficult and takes a few hour for one case, the effective combination from rough trajectory impulsive transition sub module would be prepared. These are ongoing research.

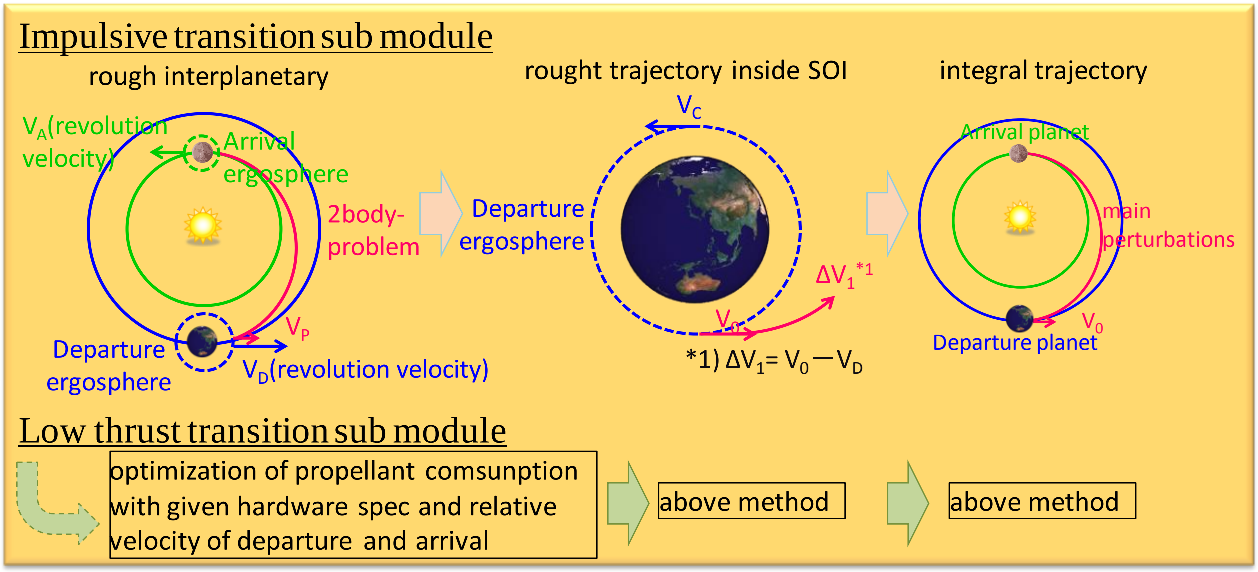


Figure 8. Interplanetary design module image

3.2 Launch Vehicle Capability Module

On the launch vehicle capability module, capability information of variety launch vehicles, available options for upper stage and solid motor are provided like Fig. 9. The capability to achieve each increment velocity for each declination in form of graph is given to improve the mission design process practicality. Fig. 9 indicate the capability, when the declination is from -30deg to +30deg. However to derive such graph for Tanegashima Space Center has inherent challenge thanks to a large number of islands around south-eastern of the cite that causes launching to deep inclination becomes relatively more difficult. In fact, the launch capability of H-IIA doesn't decrease in parallel with inclination. Hence, an optimized graph is alternatively given for this case. These are ongoing research.

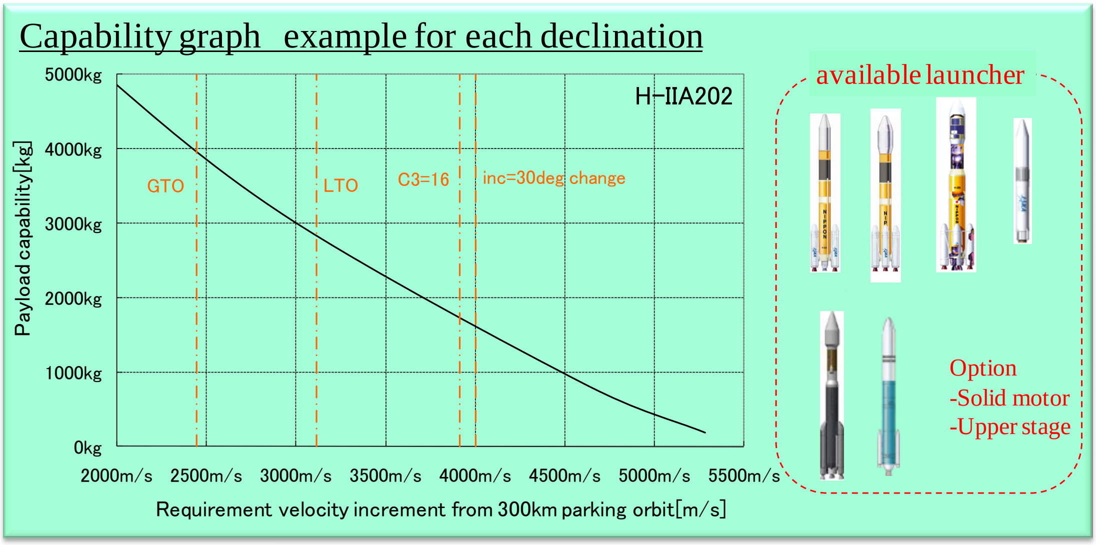


Figure 9. Launch vehicle capability module image

3.3 Adjust Maneuver Module

As shown in Fig. 10, the adjust maneuver module present two method: meeting condition maneuver method and fixed time of arrival method. The module is used to calculate the necessary propellant weight to correct the insertion error for transfer orbit of launch vehicle. The error would be inputted by single error pattern or variance-covariance matrix.

On the meeting condition maneuver method, the error correction maneuver is only once. The maneuver vector is controlled only to insert the orbit of arrival planet at the same time as nominal case like explicit guidance. In this case, the necessary propellant is suppressed but the operation schedule of GS may be changed.

On the other hand, the fixed time of arrival method try to make maneuver to insert nominal transfer orbit at given time like implicit guidance. That require two times maneuver. The merit and demerit are upside-down of Meeting condition maneuver method.

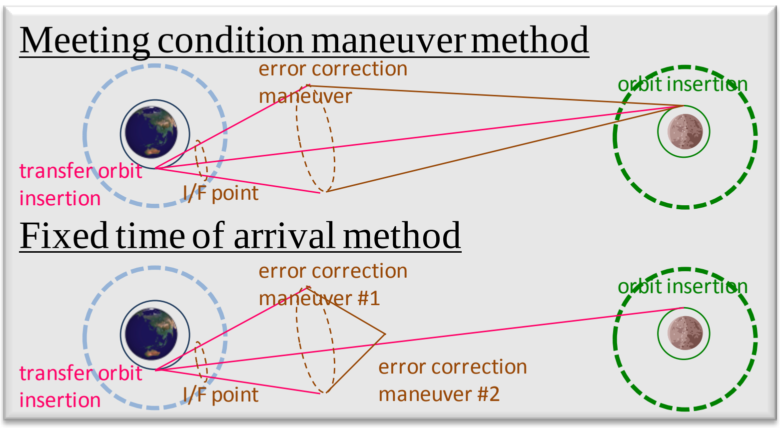


Figure 10. Adjust maneuver module image

3.4 Further Improvement

Based on the mission and user requirements, incorporating additional items are also considered. This includes, swing-by, round trip, DeltaV Earth Gravity Assist (DeltaVEGA) and Electric Delta-V Earth Gravity Assist (EDVEGA). But, as the incorporation of all items may induce bad operability, the selection would be done with considering cost-effectiveness and usability. As DeltaVEGA and EDVEGA would be used near future like Hayabusa and Hayabusa-2, how to incorporate them are being studied.

4. Trial Construction of Interplanetary Mission

In this section, the trial constructions of interplanetary mission with IRIS are presented for Jupiter and Mars. At this time, as interplanetary design module are upgrading for low thrust, only the impulsive sub module are used.

4.1 Jupiter Mission

Jupiter is fifth planet from inside the solar system and has maximum size in the solar system. It has many attractive phenomena to observe, which are liquid under atmospheric layer, occurrence of radio wave and ring around it. But there is no spacecraft to explore it in detail at orbit of the Jupiter. On the other hand, “Juno”, which is explore spacecraft for the Jupiter on NASA, will examine the composition , gravity and magnetic field. It will be launched at 2011, swing by the Earth at 2013 and arrive the Jupiter orbit insertion at 2016.

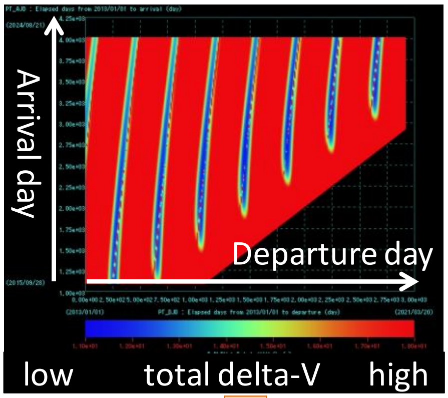
As above, the Jupiter is not explored enough and it is significant to explore Jupiter at the orbit. Although the Jupiter orbit insertion would require much energy and normally transfer through multi-swing-by with multi-impulsive method, for simplicity, two impulsive transfer orbit is designed like Fig. 11.

The mission departure the earth from 300km altitude parking orbit and arrive the Jupiter to 300km-300,000km orbit. At first, departure days of Earth are from Jan-2013 to Dec-2020 and the transfer time is limited to 3 years which is about half cycle. If the total delta-V of departure and arrival is limited to 15km/s, the possible windows are left side of the figure, which indicate 7 times. That also confirm the Juno case (No1).

After that, the departure Dec confine under 30deg for launch capability from TNSC and the arrival Dec confine under 85deg for observe condition. Furthermore, the departure velocity increment takes limit to 6.5km/s which is set by 1000kg launch capability of H-IIB+Solid motor. As simple H-IIB cannot exceed 6.0km/s at only 10kg and the windows are vanished, the launch configuration is selected. Then, the windows are limited to only No.3.

Finally, the integral trajectory was indicated at left side of the figure which is a case picked up from No.3 windows. The trajectory is Nov-2015 of departure and Dec-2018 of arrival.

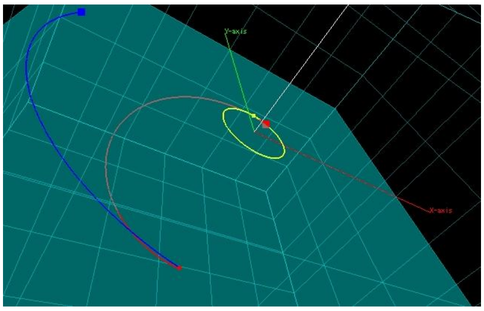
rough interplanetary



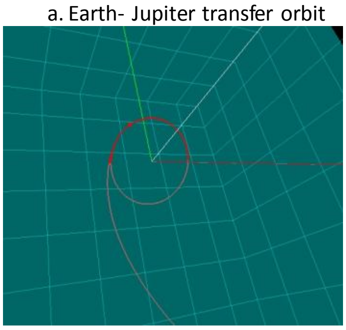
total delta-V < 15km/s

No	Departure			Arrival		
1	Jul-2013	～	Nov-2013	Jan-2016	～	Nov-2016
2	Aug-2014	～	Dec-2014	Jan-2016	～	Dec-2017
3	Sep-2015	～	Jan-2016	Mar-2017	～	Jan-2019
4	Oct-2016	～	Feb-2017	Apr-2018	～	Feb-2020
5	Nov-2017	～	Mar-2018	May-2019	～	Mar-2021
6	Dec-2018	～	Apr-2019	Jun-2020	～	Apr-2022
7	Jan-2020	～	May-2020	Jun-2021	～	May-2023

integral trajectory



a. Earth- Jupiter transfer orbit



b. Jupiter orbit insertion

Figure 11. Journey opportunities from Earth to Jupiter (2013-2020)

4.2 Mars Mission

Mars is fourth planet from inside the solar system, that means nearest exoplanet of the Earth in the solar system, and has half diameter and 10% weight of the Earth. Therefore, many explore mission were inserted to the orbit and landed on the Mars. In Japan, “Nozomi” was launched for entering the orbit of Mars at 1998 but the mission could not be succeed enough for various causes. On the other hand, there are many future mission. For example, “Melos”, which is Japanese Mars exploration spacecraft, is under consideration. Additionally, at USA, the president Barack Obama says it should be possible to send astronauts to orbit the planet Mars by the mid-2030s and return them safely to Earth. On the other hand, Russia plan to launch Phobos-Grunt at 2011, which is a sample return mission to Phobos, one of the moons of Mars.

As above, the Mars takes much attention as the target of interplanetary exploration from the world. In this section, the possible opportunities are calculated with impulsive transition sub module of IRIS.

The mission departure the earth from 300km altitude parking orbit and arrive the Mars to 300km altitude circular orbit. At first, departure days of Earth are from Jan-2010 to Dec-2030 by 20 days step size and the transfer time is limited to 300 days which is about half cycle. If the delta-V of departure is limited to 4km/s (C3=20), where the launch capability is about 1.5ton of H-IIA202, 2.7ton of H-IIA204 and 3.1ton of H-IIB, the possible windows are prepared with departure Dec and arrival velocity increment at Fig. 12, which indicate 9 times. That also confirm the Phobos-Grunt which plan to launch at Nov-2011. For geographical situation of TNSC, H-IIA takes the maximum launch capability from Dec=-30deg to Dec=30deg. The corresponding windows like 2022 and 2024 are also short span. Therefore they are eliminated. At that time, the windows until 2030 are only 7times to arrive the Mars. So, besides Melos, we have to design additional Mars mission for the possible 7 windows as soon as possible.

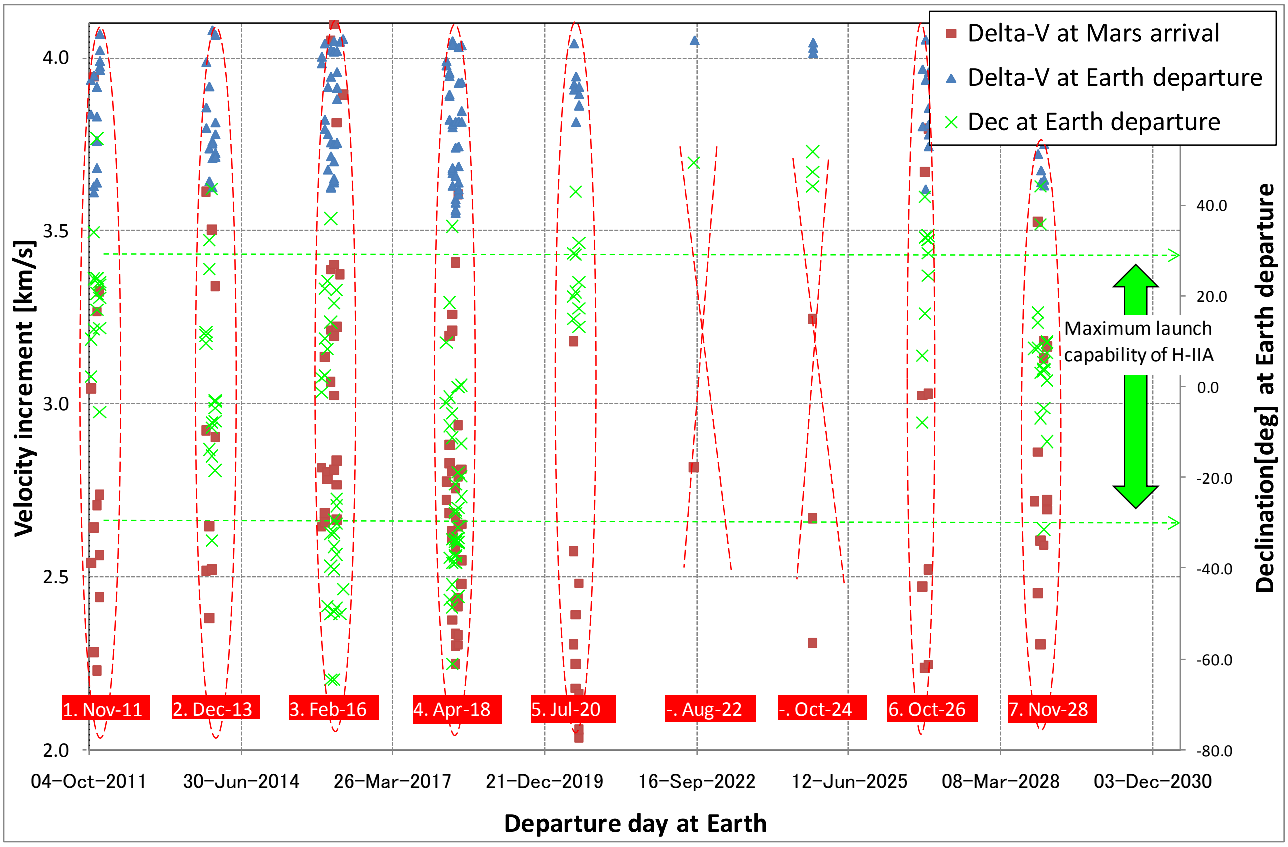


Figure 12. Journey opportunities from Earth to Mars (2010-2030)

5. Summary

Research on launch vehicle interface for interplanetary mission with H-IIA/B is indicated. In this paper, we mainly discussed the technical feasibility for update of launch capability, extension of launch window and flexibility of changing launch day. In addition, interplanetary mission design tool is introduced with some trial construction. The tool is aimed at understanding interplanetary mission from launch vehicle phase to spacecraft phase and effective design of interplanetary mission. These research would be continued for realizing various interplanetary mission in Japan or international partnership.

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