A Search for Viable Venus and Jupiter Sample Return Mission Trajectories for the Next Decade

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ABSTRACT

Planetary exploration using unmanned spacecraft capable of returning geologic or atmospheric samples have been discussed as a means of gathering scientific data for several years. Both NASA and ESA performed initial studies for Sample Return Missions (SRMs) in the late 1990's, but most suggested a launch before the year 2010. The GENESIS and STARDUST spacecraft are the only current examples of the SRM concept with the Mars SRM expected around 2015. A feasibility study looking at SRM trajectories to Venus and Jupiter, for a spacecraft departing the Earth between the years 2011 through 2020 was conducted for a university project. The objective of the study was to evaluate SRMs to planets other than Mars, which has already gained significant attention in the scientific community. This paper is a synopsis of the study's mission trajectory concept and the conclusions to the viability of such a mission with today's technology.

1. INTRODUCTION

The Apollo Program's lunar landing missions represent the only successful attempts to date in which a specimen from a celestial body has been return to the Earth for study. It has been over thirty years since the Apollo Program. The concept of a SRM affords the scientific community the opportunity to closely study material from another planet. This opportunity can help answer question such as the chemical composition of a planet's surface and the atmospheric composition of a planet of interest. This type of information assists scientists in better understanding the Earth and its future.

While not a new concept, SRMs previously have not been in the forefront of planetary exploration. Because of the requirement to "return a sample," considerable energy is required to successfully complete such a mission. The energy requirement for SRMs equates to propellant mass and ultimately the launch costs of the mission. Previous efforts have concluded multiple launches are necessary because of the large required propellant for the SRM [1].

The current state of rocket and propulsion technology, however, warrants a feasibility evaluation for the SRM occurring in the next decade. The Evolved Extended Launch Vehicle (EELV) represents the backbone of US launch vehicles in the near future.

Since Mars is the current focus of NASA, alternative planets are used to evaluate the feasibility of SRM. The planet Venus, which was explored by the Magellan spacecraft from 1990 until it was de-orbited into the planet in October 1994, was chosen for the inner planet study case. Venus represents an alternative to the currently popular Mars exploration missions. Venus is also closer to Earth than Mars thereby increasing the chances for a feasible mission. The outer planet case utilizes the planet Jupiter, which was explored by the Galileo spacecraft until it was de-orbited into Jupiter's atmosphere in September 2003. Jupiter's moon Europa is of particular scientific interest because of its icy surface, which is believed to hold the building blocks of life, warrants a comprehensive "Mars-like" exploration may be in the future [2]. Jupiter itself is the closest of the outer planets, which makes it a suitable bounding case.

To narrow the scope of the architecture study, three constraints were observed. The first constraint is the Earth departure date. NASA has speculated the year 2011 as the earliest launch of a MSRM. Other SRMs, which utilize reuse of the MSRM technology, can therefore occur no earlier than 1 January 2011. The second constraint is the ten-year MET for the operational phase of the mission. This constraint was chosen for two reasons. Ten-years is just short of the orbital period of Jupiter, the closest of the outer planets. Ten-years also reduces the required computational simulation time to a manageable level.

2. BACKGROUND

A Matlab math model is utilized to compute the transfer orbit trajectories and thus the ΔV for this study. The ΔVs are essential to perform the viability assessment of each case study mission. To solve for the trajectories, two fundamental orbit mechanics methods are used:

- The patched-conic Approximation
- Lambert's Problem

2.1. The Patched-conic Approximation

The model makes use of the patched-conic approximation to determine the interplanetary trajectories. The patched-conic method breaks the interplanetary trajectory into three small distinct problems, and then solved using the two-body system. The patched-conic approximation is an industry-accepted method when making a first-ordered analysis for interplanetary trajectories. The method has the advantage of shorting the computational time without sacrificing the integrity of the generated data [3].

The simplifying assumption of the patched-conic is based on the utilization of the two-body system to approximate the motion of the spacecraft through its trajectory. The idea of the two-body system is when considering only two bodies, the spacecraft and the celestial body it orbits, the gravitational attraction of the celestial body is the dominant effect on the spacecraft. Because of the dominating gravitational effect of the celestial body such as a planet, the spacecraft is said to be within the "sphere of influence" of the planet. Fig. 1 illustrates the various trajectory geometries. For all three parts of the patched-conic approximation, the perturbation effects by all other bodies including the spacecraft itself are considered small because of the sphere of influence concept.

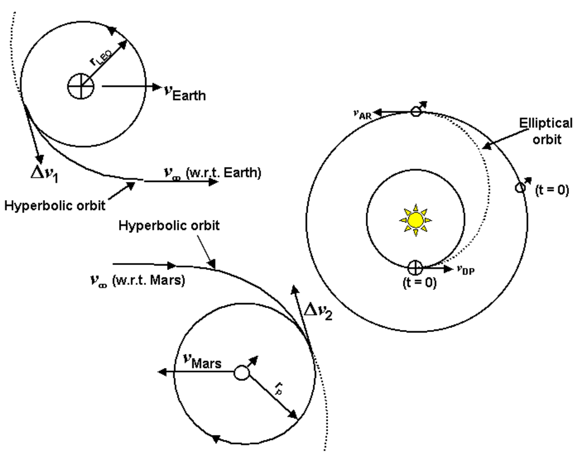

Fig. 1: Patched-Conic Departure, Interplanetary, and Arrival Trajectories

Additional assumptions are made for trajectory simulation purposes. The first assumption is that a Launch Vehicle (LV) places a spacecraft into an initial circular Low Earth Orbit (LEO) altitude of 300 km. This assumption is used as an initial condition for the model. For the interplanetary transfer orbit, a Hohmann transfer is assumed, which yields the theoretical minimum ΔV. Since the atmosphere of Venus begins at approximately 100 km. altitude, a circular orbit at this altitude is assumed for the insertion and corresponding departure orbits for the Venus SRM (VSRM) study case. Because of the general scientific interest, the orbital altitude of Jupiter's moon, Europa, is assumed for the Jupiter SRM (JSRM) study case. No attempt is made to model the orbit about Europa itself.

2.2. Lambert's Problem

Although the patched-conic method provides the information for the spacecraft to perform the ΔVs, the orbital elements still need to be determined. For the model, the user selects the departure and the arrival dates for the transfer orbit. This provides the Time of Flight (TOF) for the interplanetary trajectory. The departure date also fixes a departure position of the departure planet and thus the spacecraft with respect to the sun. Likewise, the arrival date fixes the arrival position of the target planet also with respect to the sun. Combining these three elements together, the two position vectors and the time between them, uniquely defines the transfer orbit. The problem of two position vectors and the TOF between them is known as "Lambert's Problem." There have been several methods for solving Lambert's problem. Because of its robustness, the algorithm developed by Battin [4] is ideal for general-purpose use and is implemented for this study.

2.3. Methodology

Because of the ten-year departure period and ten-year mission life, over 26-million trajectories are generated per case for this study. To solve for the transfer orbit trajectories and the mission ΔVs, a systematic mission analysis process is developed. Fig. 2 illustrates the mission analysis process, at a high level, as implemented in the Matlab software code. The model takes a departure date then steps through each arrival date increasing TOF. Once all arrival dates have been iterated upon, the departure date is incremented and the process repeats itself. After all the departure dates have been iterated, the model reduces the data by searching for the minimum ΔV for that particular case.

Once the user inputs the range of departure dates and derives the last arrival date, the model converts the dates to Julian days for analysis. A matrix of arrival dates verses departure dates is then created with each value representing an elapsed Julian day from the departure date. The planetary ephemeris of the target planet is updated using Lambert's problem based on reference planetary ephemeris form observational data and the arrival date. Similarly, the planetary ephemeris of the departure planet is updated based on the departure date. This gives enough information, the two position vectors and the time between them, to solve Lambert's problem for the interplanetary transfer orbit.

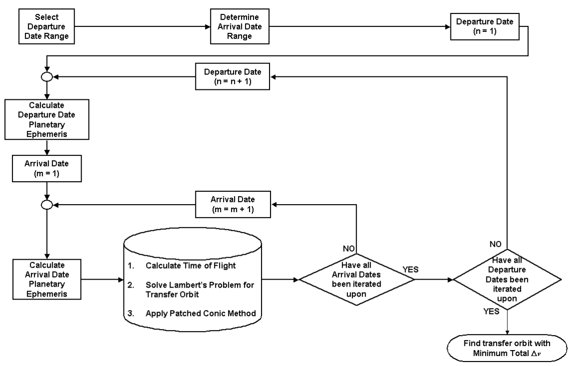

Fig. 2: Matlab Mission Analysis Model

3. RESULTS

To evaluate the trajectories for both ΔV_{Total} and TOF a figure of merit (FOM) [5] analysis is performed with the goal of finding the trajectory, which maximizes the total FOM. Since minimizing the ΔV_{Total} equates to a savings in propellant mass, the ΔV parameter is weighted twice as important than TOF. The scoring for the ΔV_{Total} is simply a ten to one scale with the smallest ΔV_{Total} as ten and the largest ΔV_{Total} as one. The TOF is also scored on a ten to one scale with the shortest TOF as ten and the longest TOF as one. Following this evaluation criterion, an equation is developed to score each candidate trajectory:

$$2 \text{ x } \Delta V_{Total} \text{ score} + \text{TOF score} = \text{FOM}_{Total}$$

The results show the minimum ΔV_{Total} transfer orbit opportunities for each year of the next decade. For each departure date, ten years of arrival dates were evaluated to determine the minimumΔV_{Total} for that year.

3.1. Venus Sample Return Mission Trajectories

Tables 1 and 2 show the Earth to Venus velocity and flight time characteristics for the minimum ΔV transfer orbits per Earth departure year respectively. Table 1 provides the required ΔV maneuvers to depart Earth and insert the spacecraft into Venus orbit. The table also includes the velocity at infinity (V_∞) of the hyperbolic departure orbit, and the corresponding escape energy (C_3). Table 2 provides the departure and arrival dates and the associated TOF of the trajectory.

Table 1: Velocity characteristics for Earth to Venus transfer

Year	ΔV_1 (km/sec)	ΔV_2 (km/sec)	V_∞ (km/sec)	C_3 (km^2/sec^2)
2011	4.1554	3.1367	4.6678	21.7884
2012	3.3039	3.8206	1.5093	2.2780
2013	3.3349	3.7323	1.7213	2.9629
2014	3.6720	3.3514	3.2456	10.5339
2015	3.5653	3.4376	2.8486	8.1145
2016	3.2167	4.1041	0.6028	0.3633
2017	3.3580	3.6342	1.8643	3.4755
2018	3.3854	3.5954	2.0208	4.0836
2019	4.0909	3.1399	4.5008	20.2576
2020	3.2585	3.9054	1.1306	1.2783

Table 2: Time of Flight for Earth to Venus transfer

Year	Departure Date	Arrival Date	TOF (Days)
2011	12/29/11	8/25/12	240
2012	12/31/12	11/20/13	324
2013	1/8/13	11/27/13	323
2014	12/28/14	9/22/15	268
2015	1/1/15	9/26/15	268
2016	1/1/16	12/18/16	352
2017	12/28/17	10/20/18	296
2018	1/4/18	10/26/18	295
2019	12/31/19	8/26/20	239
2020	12/26/20	11/15/21	324

Tables 3 and 4 show the Venus to Earth velocity and flight time characteristics respectively for the minimum ΔV transfer orbits per Venus departure year. The parameters in Table 3 are similar to that of Table 1 with the exception of excluding V_∞ and the corresponding C_3. Although these values are calculated in the Matlab model, their main use is for LV sizing. In the case of the return trajectory, there is no LV because the spacecraft itself performs the ΔV required to insert the payload into a hyperbolic Venus departure orbit. The values of V_∞ and the corresponding C_3 are thus not applicable to the return trajectory in the context of this project. The data in Table 4 is similar to that in Table 2.

Table 3: Velocity characteristics for Venus to Earth transfer

Year	ΔV_1 (km/sec)	ΔV_2 (km/sec)	V_∞ (km/sec)	C_3 (km^2/sec^2)
2012	4.4792	3.4601	N/A	N/A
2013	3.8434	3.3575	N/A	N/A
2014	3.2888	3.9291	N/A	N/A
2015	4.3108	3.4127	N/A	N/A
2016	3.3875	3.6384	N/A	N/A
2017	3.1493	4.4450	N/A	N/A
2018	4.0564	3.3329	N/A	N/A
2019	3.5294	3.6128	N/A	N/A
2020	4.4987	3.5093	N/A	N/A
2021	3.8439	3.3414	N/A	N/A

Table 4: Time of Flight for Venus to Earth transfer

Year	Departure Date	Arrival Date	TOF (Days)
2012	5/12/12	7/7/13	421
2013	8/5/13	7/8/14	337
2014	10/29/14	7/9/15	253
2015	6/9/15	7/6/16	393
2016	12/21/16	1/4/17	14
2017	11/26/17	7/10/18	226
2018	7/8/18	7/9/19	366
2019	10/2/19	7/10/20	282
2020	5/12/20	7/8/21	422
2021	8/4/21	7/8/22	338

Table 5 shows the results of the FOM scoring for the VSRM. The results show planning a mission utilizing the highest score for both the departure and return trajectories yields no solution since the return trajectory occurs before the departure trajectory. The dilemma for mission planning is which trajectory, departure or return, to select. Utilizing the FOM evaluation criteria based on departure trajectory yields TOF to Venus as 295 days but, limits the return trajectories to after 26 October 2018. Basing the mission on the return trajectory criteria yields TOF to Venus as 268 days. In the course of this study, a 14-day return trajectory in late 2016 was found which seems to take advantage of a favorable planetary alignment between Venus and Earth.

Table 5: Trajectory Figures of Merit for Venus SRM

Departure Trajectories				Return Trajectories			
Year	ΔV Score	TOF Score	FOM Score	Year	ΔV Score	TOF Score	FOM Score
2011	2	9	13	2012	2	2	6
2012	5	3	13	2013	7	6	20
2013	6	4	16	2014	6	8	20
2014	7	8	22	2015	3	3	9
2015	8	8	24	2016	10	10	30
2016	1	1	3	2017	4	9	17
2017	9	5	23	2018	5	4	14
2018	10	6	26	2019	9	7	25
2019	3	10	16	2020	1	1	3
2020	4	3	11	2021	8	5	21

3.2. Jupiter Sample Return Mission Trajectories

Tables 6 and 7 show the Earth to Jupiter velocity and trajectory characteristics for the minimum ΔV transfer orbits per Earth departure year respectively. Tables 8 and 9 show the Jupiter to Earth velocity and trajectory characteristics respectively for the minimum ΔV transfer orbits per Jupiter departure year. The velocity and trajectory characteristics in all the JSRM tables are similar to their VSRM counterparts.

Table 6: Velocity characteristics for Earth to Jupiter transfer

Year	ΔV_1 (km/sec)	ΔV_2 (km/sec)	V_∞ (km/sec)	C_3 (km^2/sec^2)
2011	6.1493	7.7272	8.5524	73.1430
2012	6.4899	10.1822	9.0945	82.7106
2013	7.5551	14.3265	10.6831	114.1287
2014	7.2125	10.7404	10.1870	103.7759
2015	7.1223	10.7077	10.0544	101.0916
2016	7.2544	5.6990	10.2485	105.0314
2017	7.1542	5.7088	10.1014	102.0386
2018	7.0373	5.7384	9.9284	98.5723
2019	6.8993	5.8030	9.7220	94.5172
2020	6.7369	5.9309	9.4759	89.7929

Table 7: Time of Flight for Earth to Jupiter transfer

Year	Departure Date	Arrival Date	TOF (Days)
2011	7/11/11	6/17/14	1072
2012	7/15/12	8/15/14	761
2013	10/18/13	12/31/23	3726
2014	9/22/14	12/31/24	3753
2015	9/21/15	12/31/24	3389
2016	7/13/16	4/10/26	3558
2017	7/13/17	4/10/26	3193
2018	7/13/18	4/10/26	2828
2019	7/13/19	4/10/26	2463
2020	7/11/20	4/10/26	2099

The differences between the candidate departure orbits are seen in the ΔV data in Table 6. Between 2016 and 2020 the ΔV_1 magnitude has a decreasing trend while the ΔV_2 magnitude has a corresponding increasing trend. The change in the ΔV's are directly related to the changes in the TOF in Table 7 because of the differences in the calculated trajectories despite the similar departure dates and the same arrival date. The trend also suggest a lower ΔV_{Total} in the years just beyond 2020, which is beyond the scope of this study.

Table 8: Velocity characteristics for Jupiter to Earth transfer

Year	ΔV_1 (km/sec)	ΔV_2 (km/sec)	V_∞ (km/sec)	C_3 (km^2/sec^2)
2020	7.2233	5.7540	N/A	N/A
2021	7.3096	5.8867	N/A	N/A
2022	7.5485	6.2501	N/A	N/A
2023	7.5431	6.2420	N/A	N/A
2024	7.4279	6.0674	N/A	N/A
2025	7.3931	6.0144	N/A	N/A
2026	7.4131	6.0449	N/A	N/A
2027	10.4812	10.3805	N/A	N/A
2028	20.5148	22.4696	N/A	N/A
2029	32.4785	35.5349	N/A	N/A

Table 9: Time of Flight for Jupiter to Earth transfer

Year	Departure Date	Arrival Date	TOF (Days)
2020	7/5/20	1/8/23	917
2021	1/1/21	12/17/22	715
2022	2/4/22	11/4/24	1004
2023	12/31/23	9/22/26	996
2024	12/31/24	8/26/27	968
2025	6/17/25	8/15/27	789
2026	1/26/26	7/18/28	904
2027	1/1/27	7/31/29	942
2028	1/1/28	8/28/29	605
2029	1/1/29	9/27/30	634

Table 10 shows the results of the FOM scoring for the JSRM. The results show a similar mission planning dilemma as the VSRM results.

Table 10: Trajectory Figures of Merit for Jupiter SRM

Departure Trajectories				Return Trajectories			
Year	ΔV Score	TOF Score	FOM Score	Year	ΔV Score	TOF Score	FOM Score
2011	5	9	19	2020	10	5	25
2012	4	10	18	2021	9	8	26
2013	1	2	4	2022	4	1	9
2014	2	1	5	2023	5	2	12
2015	3	4	10	2024	6	3	15
2016	6	3	15	2025	8	7	23
2017	7	5	19	2026	7	6	20
2018	8	6	22	2027	3	4	10
2019	9	7	25	2028	2	10	14
2020	10	8	28	2029	1	9	11

4. DISCUSSION

A major concern for any space mission is the minimization the total mass of the spacecraft, which includes the mass of the propellant necessary to perform the mission. For SRMs in general, a significant amount of propellant is required since the spacecraft must perform a ΔV to place itself onto a return hyperbolic trajectory. A viable mission must also have mass allocated for the spacecraft subsystems such as electrical power, guidance and navigation, thermal control, and communications in addition to the physical structure which amount to the "dry mass" of the spacecraft. The "wet mass" of the spacecraft is the dry mass with the addition of the required propellant mass. The wet mass represents the gross mass of the spacecraft injected into space by the LV. The LV "throw" capability is therefore the parameter used for evaluation of mission feasibility.

For this study, the Delta IV is assumed for mission feasibility evaluation. Fig. 3 shows the Earth escape energy performance for the various configuration of the Delta IV. From Fig. 3 the Delta IV Heavy configuration can "throw" approximately 7800 kg of gross spacecraft mass if the required energy to escape Earth's gravity is 10 km^2/s^2 or less.

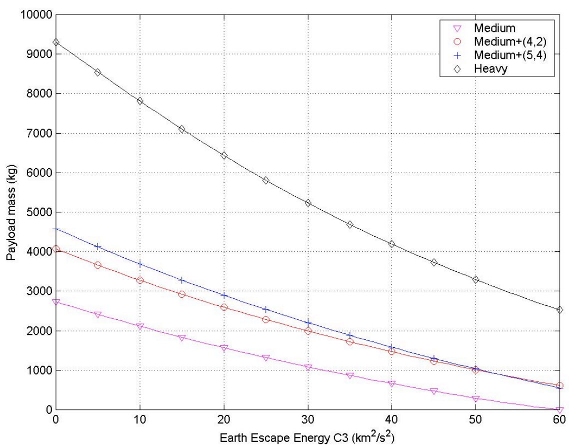

Fig. 3: Predicted Planetary Mission Performance of Delta IV Launch Vehicles [6]

4.1. Case Selection

To determine the feasibility of the two mission cases, a representative mission plan from each study case is selected. In both study cases, the representative mission plan uses the FOM criteria for the return trajectory then matches the "best" departure trajectory, based on FOM score. Although no JSRM mission plan meets the ten-year MET constraint, the representative case is evaluated for completeness.

Table 11 shows the selected VSRM evaluation case. The total MET for the VSRM is 734 days or two years, three days. Comparing the transfer time to Venus with the Magellan mission, the selected VSRM plan arrives 195 days sooner than the Magellan spacecraft, which had a transfer time of 463 days.

Table 11: Selected Mission Plan for VSRM

Parameter	Earth to Venus	Venus to Earth
Departure Date	1/1/15	12/21/16
Arrival Date	9/26/15	1/4/17
Time of Flight (Days)	268	14
ΔV_1 (km/sec)	3.5653	3.3875
ΔV_2 (km/sec)	3.4376	3.6384
ΔV_{Total} (km/sec)	7.0029	7.0259
Departure V_∞ (km/sec)	2.8486	N/A
Launch Vehicle Escape Energy C_3 (km^2/sec^2)	8.1145	N/A

Table 12 shows the selected JSRM evaluation case. The total MET for the JSRM is 4,177 days. Comparing the JSRM departure transfer time to that of the Galileo mission to Jupiter, the JSRM arrives at Jupiter in just under three years while Galileo took six years of transfer time using planetary gravity assists.

Table 12: Selected Mission Plan for JSRM

Parameter	Earth to Jupiter	Jupiter to Earth
Departure Date	7/11/11	1/1/21
Arrival Date	6/17/14	12/17/22
Time of Flight (Days)	1072	715
ΔV_1 (km/sec)	6.1493	7.3096
ΔV_2 (km/sec)	7.7272	5.8867
ΔV_{Total} (km/sec)	13.8765	13.1963
Departure V_∞ (km/sec)	8.5524	N/A
Launch Vehicle Escape Energy C_3 (km^2/sec^2)	73.1430	N/A

4.2. Mission Feasibility Evaluation

Examination of the C_3 from both missions and comparing them to the curves in Fig. 3, a maximum spacecraft mass is determined. For the VSRM, an interpolated payload mass of 8085 kg is found. By assuming direct injection, the first ΔV performed at the 300 km altitude is ignored. An extrapolated mass of 1918 kg is determined for the JSRM. The spacecraft mass for the JSRM is nearly 25% that of the VSRM because of the large C_3 term. Given these two mass figures and the ΔV information from both mission plans, wet and dry mass estimates are derived using the rocket equation, with specific impulse (Isp) as the variable. Current bi-propellant systems have and Isp in the range of 200 to 450 seconds.

Fig. 4 illustrates the wet/dry mass estimates for both missions assuming direct injection of the spacecraft into the transfer orbit by the Delta IV Heavy LV. The spacecraft performs the remaining three ΔV maneuvers outlined in the mission plans.

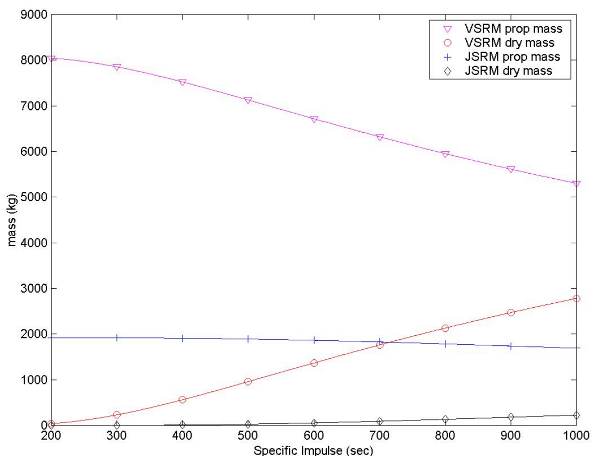

Fig. 4: Spacecraft mass breakdown (three-ΔV case)

For the VSRM case, the 8085 kg gross mass breaks down into 7329.4 kg of propellant and 755.6 kg of dry mass for an Isp of 450 seconds. More challenging, from a mass perspective, is the JSRM with only 16.8 kg of dry mass to allocate for the same Isp as the VSRM. Even with an Isp of 1000 seconds, the JSRM could only allocate 227.3 kg of dry mass to the spacecraft subsystems. To approach the 755.6 kg dry mass of the VSRM with an Isp of 450 seconds, the Isp required for the JSRM is 2300 seconds.

Because of the mass challenges, a two-ΔV scenario is conceived for study. A ballistic return trajectory is considered to eliminate the need to perform a ΔV to insert the spacecraft into Earth orbit. A ballistic trajectory assumes the mission is planned well enough such that the returning sample capsule will re-enter Earth's atmosphere on a precise trajectory to land on a predetermined spot on the Earth for recovery. Fig. 5 shows the mass estimates for the two-ΔV scenario.

For the VSRM two-ΔV case, the 8085 kg gross mass breaks down into 6362.2 kg of propellant and 1722.8 kg of dry mass for an Isp of 450 seconds. Similar analysis for the JSRM shows a propellant mass of 1854.4 kg and a dry mass of 63.6 kg. Although the dry mass for the JSRM has increased almost four times that of the three-ΔV case, the JSRM still has significant challenges when it comes to dry mass.

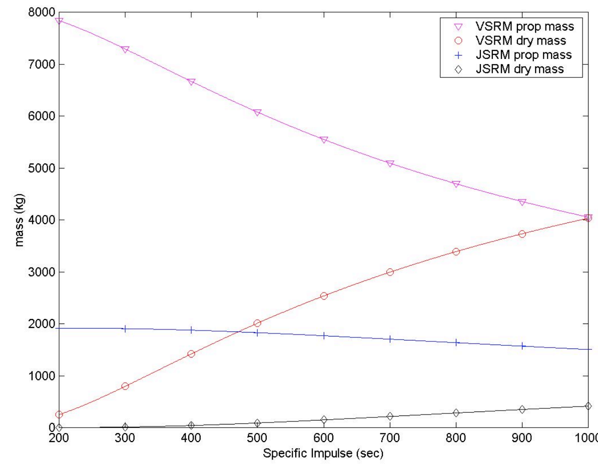

Fig. 5: Spacecraft mass breakdown (two-ΔV case)

Since Magellan and Galileo did not have a requirement to return to Earth, a strait comparison of dry mass is somewhat biased. The above two-ΔV and three-ΔV cases assume the total dry mass of the spacecraft including the lander returns to Earth. An actual SRM would most likely abandon the lander after that portion of the mission was over. Abandoning any unnecessary dry mass has the advantage of reducing mass of the spacecraft for the return trajectory. Each mission phase has it own unique dry mass configuration leading to a form of mass staging, which is common in multi-stage rockets. Table 13 shows the estimated dry mass for the various pieces of hardware form a JPL study. Assuming these are the required dry masses for any SRM, the gross mass of the spacecraft with propellant can be determined for both SRM study cases.

Table 13: JPL VSRM study dry mass estimate [1]

Subsystem	Departure Trajectory dry mass	Return Trajectory dry mass
Orbiter Vehicle	680 kg	680 kg
Earth Entry Vehicle	20 kg	20 kg
Lander	931 kg	N/A
Ascent System	476 kg	N/A
Sample	N/A	5 kg
Dry Mass Totals	2107 kg	705 kg

The table shows two mass configurations, the departure trajectory and the return trajectory configuration. For the departure trajectory configuration, the LV must lift the Orbiter Vehicle and Earth Entry Vehicle as well as the Lander and Ascent System. For the return trajectory, the Lander and the Ascent System are abandoned since they were only required to collect the five-kilogram sample, rendezvous with the Orbiter Vehicle, and transfer the sample to the Earth Entry Vehicle. Once sample transfer is complete, the Ascent System is jettisoned. The Lander itself remains on the surface of the planet surface. The dry mass of the return trajectory is approximately one-third that of the departure trajectory. Since the change in dry mass affects the required propellant mass, a savings in total gross mass is realized.

Fig. 6 demonstrates the advantages of mass staging technique for the VSRM case. It shows, that for an Isp of 450 seconds and the dry mass configurations for the different mission stages shown in Table 13, the total spacecraft gross mass at launch is 6363 kg. This gross mass includes 4256.0 kg of propellant mass for the two-Dv maneuvers. The propellant required to insert the spacecraft into Venus orbit is 3442.4 kg. To perform the Dv maneuver for a ballistic return trajectory to Earth requires 813.6 kg of propellant.

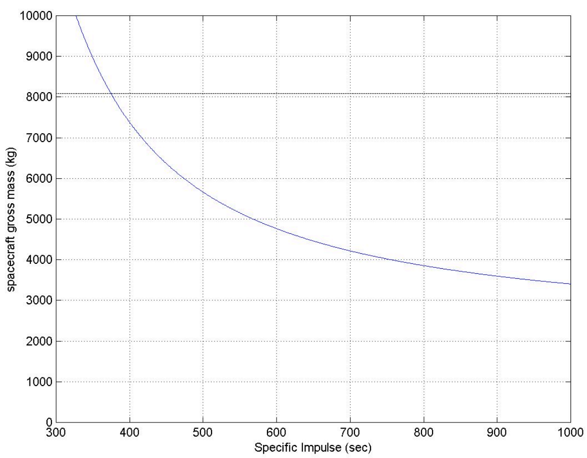

Fig. 6: VSRM gross spacecraft mass estimate (two-ΔV case) using staging

The staging technique indicates a VSRM is feasible with current bi-propellant technology because the gross mass at launch is less than the reference 8085 kg LV capability. The remaining lift capability of 1722 kg equates to a 21.3% margin. Margin of this magnitude is essential during the developmental phase of any program to cover any unforeseen contingency situations.

Since the dry mass of the JPL proposed SRM is greater than the 1918 kg lift capability of the Delta IV Heavy for the JSRM, the mission is automatically not feasible. Other techniques in mission design could be employed for the JSRM such as the gravity assist, but that technique has the detriment of significantly increasing the TOF. In the case of To make the JSRM possible, a LV with significantly more capability needs to be developed or a different mission design other than direct injection, such as gravity assist trajectories, is required.

5. CONCLUSIONS

This study has shown that an inner planet SRM and specifically a VSRM is feasible in the 2011 – 2020 timeframe. The selected mission plan meets the key parameters of minimizing both ΔV and MET for the mission. The mission duration of two years and three days meets the less than ten-year objective of the study. The spacecraft's gross mass estimate of 6363 kg meets the lift capability of the Delta IV Heavy LV with 21.3% margin. The current state of the art propulsion technology is sufficient to meet the required ΔVs of the mission. This study has also shown that a JSRM is not feasible in the 2011 – 2020 timeframe. The selected mission plan does minimize ΔV and MET, but the MET exceeds the ten-year parameter of this study. Significant LV development is required to increase the lift capability for a direct injection. This type of LV development is currently not planed to meet this study's timeframe. Even with improvements to LV lift capability, the JSRM spacecraft requires significant advances in propulsion technology to reduce the required propellant. Since Jupiter is the closest outer planet, study of SRM's to any of the other outer planets is not recommended given the current state of technology.

6. REFERENCES

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