SPACE LAUNCH SYSTEM MISSION FLEXIBILITY ASSESSMENT

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The Space Launch System (SLS) is envisioned as a heavy lift vehicle that will provide the foundation for future beyond low Earth orbit (LEO) missions. While multiple assessments have been performed to determine the optimal configuration for the SLS, this effort was undertaken to evaluate the flexibility of various concepts for the range of missions that may be required of this system. These mission scenarios include single launch crew and/or cargo delivery to LEO, single launch cargo delivery missions to LEO in support of multi-launch mission campaigns, and single launch beyond LEO missions. Specifically, we assessed options for the single launch beyond LEO mission scenario using a variety of in-space stages and vehicle staging criteria. This was performed to determine the most flexible (and perhaps optimal) method of designing this particular type of mission. A specific mission opportunity to the Jovian system was further assessed to determine potential solutions that may meet currently envisioned mission objectives. This application sought to significantly reduce mission cost by allowing for a direct, faster transfer from Earth to Jupiter and to determine the order-of-magnitude mass margin that would be made available from utilization of the SLS. In general, smaller, existing stages provided comparable performance to larger, new stage developments when the mission scenario allowed for optimal LEO dropoff orbits (e.g. highly elliptical staging orbits). Initial results using this method with early SLS configurations and existing Upper Stages showed the potential of capturing Lunar flyby missions as well as providing significant mass delivery to a Jupiter transfer orbit.

I. Introduction

In early 2011 immediately following the Space Launch System (SLS) Requirements Analysis Cycle (RAC) and Mission Concept Review (MCR), a study was undertaken by members of RAC Team 2 that assessed alternative mission scenarios that could make use of this highly capable heavy-lift launch vehicle. This study focused on the technical feasibility of performing wide ranging single launch mission scenarios using many in-space propulsion options in multiple fashions across a variety of conceivable future heavy-lift performance capabilities. To demonstrate this approach, we assessed a specific mission to the Jupiter system. Results of this analysis identified minimum-energy opportunities, mass delivery capabilities, and ability to reduce mission costs through reduced trip duration and increased mass delivery (ample margin and spacecraft simplification potential).

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II. Heavy Lift Capabilities Overview

Leading up to the SLS MCR, the RAC teams generated feasible vehicle concepts that could meet the established SLS threshold requirements. These studies used rigorous technical analysis to verify that vehicle concepts could meet these requirements, and detailed the affordability approaches that were being used to reduce the design, development, test, production, and operational costs of the concepts.

Initial vehicle concepts delivered at least 70 metric tons to a representative low-Earth orbit, a lower boundary for acceptable payload delivery performance. In fact, most concepts were able to deliver a minimum threshold of 100 metric tons to LEO. This 70-100 metric ton to LEO range is a reasonable best estimate for the type of early performance that can be expected from the next heavy lift launch system.

Each of the concept teams developed intermediate or early block upgrades to improve vehicle performance. While some concepts utilized additional stages for optimized performance, others increased engine thrust/efficiency or increased the quantity of engines on the stages. The expected performance potential of these early upgrade options ranged from about 120 metric tons up to about 140 metric tons to LEO.

Evolved performance (referred to informally as the “vision vehicle”) represented a reasonable estimate for the maximum achievable performance within a given vehicle family. These vision vehicles varied in their upgrade approach, but they typically required major element changes, additional engines, or major upgrades to the existing engine or booster systems if applicable. Performance for this class of vehicles ranged from about 150 metric tons up to almost 200 metric tons to LEO.

To summarize, the LEO performance of an eventual heavy lift vehicle can be expected to be in the 70 to 100 metric ton range initially, with an intermediate capability of between 120 to 140 metric tons achievable with some upgrades, and a maximum achievable performance potential of 150 to 200 metric tons. The vehicles’ performance for beyond-LEO scenarios will be discussed in a subsequent section.

III. In-Space Stage Options

Another way to increase performance is by adding an in-space stage to the vehicle configurations. For purposes of this analysis, dedicated in-space stages were assessed that are representative of the type and size of stages that are either existing or currently envisioned for further development. The beyond LEO performance of these stages is primarily a function of efficiency. This can be characterized using specific impulse (Isp) of the engines (propulsion system efficiency) and propellant mass fraction (pmf, structural design efficiency).

Current Domestic Stages

The four existing domestic stages considered during this analysis are described below.

Atlas V Centaur[1]

This stage has a long flight heritage on the Atlas and Titan vehicle families and can be used in either a single or dual engine configuration depending on mission needs. Characterized by using the RL-10 engine, specific impulse of this cryogenic LOX/LH2 system is above 450 seconds. Further, this stage is approximately 3 meters in diameter and almost 13 meters in total integrated length. This allows the stage to hold approximately 21 metric tons of usable LOX/LH2 propellants in a very efficient structural packaging that results in a stage inert mass of about 2 metric tons (pmf ~0.91).

Delta IV 4-meter Upper Stage[1]

This stage is used on certain configurations in the Delta family of vehicles and utilizes a single RL-10B2 engine with a deployable nozzle that achieves a specific impulse greater than 460 seconds. This stage has a 4 meter diameter and is about 12 meters is total integrated length. Total usable propellant mass is greater than 20 metric tons, and the inert mass is slightly less than 3 metric tons (pmf ~0.84).

Delta IV 5-meter Upper Stage[1]

Also used on certain configurations in the Delta family of vehicle, including the Delta IV-Heavy, this stage is similar in design to the 4-m Upper Stage. Using the RL-10B2, the stage is 5 meters in diameter and about 12 meters in total length. This diameter change allows over 27 metric tons of usable propellant to be packaged efficiently with an inert mass of about 3.5 metric tons (pmf ~0.89).

Falcon 9 2nd Stage[2]

A new entrant into the domestic launch market, the Falcon 9 is an all-LOX/RP launch vehicle that has successfully launched from Cape Canaveral on two occasions as of this writing. The second stage uses what is typically referred to as the Merlin 1V (vacuum) that is similar in design to the Merlin engines used on the First Stage of the Falcon 9 vehicle. Large efficiency increases for the 2nd Stage engine are realized through the usage of a niobium nozzle extension (Isp greater than 340 seconds). Furthermore, LOX/RP liquid propellant stages are typified by very high structural efficiencies. The Falcon 9 2nd Stage holds about 49 metric tons of usable propellant with only 3 metric tons of inert mass required (pmf of about 0.94).

Selected Current International Stages

In addition to the domestic stages that were considered, the technical feasibility of utilizing selected international stages was also assessed. The international stages that were considered includes:

Japanese H-IIB 2nd Stage[3]

The H-IIB 2nd Stage is a LOX/LH2 cryogenic stage that has been used on recent configurations of the H-II family of vehicles. It has a 4 meter diameter and is slightly greater than 9 meters in length. It is powered by the LE-5B engine that achieves a specific impulse of nearly 450 seconds. Holding almost 17 metric tons of propellant with a 3 metric ton inert mass, the pmf is about 0.85

Ariane 5 Cryogenic Upper Stage (ESC-A)[1]

Since 2005, the Ariane 5 vehicle has successfully flown with the ESC-A LOX/LH2 Upper Stage 32 times. This stage utilizes the HM7B engine that provides around 445 seconds of Isp. With a diameter of almost 5.5 meters and only having a length of 5 meters, it is a very short and compact stage which would work well with very large payloads where total encapsulated length could become an issue. However, it is small relative to other similar assessed options, holding only 15 metric tons of LOX/LH2 propellants with an inert mass of almost 3.5 metric tons (pmf ~0.81). A follow-on stage has been proposed that would utilize a to-be-developed Vinci engine (Isp proposed at 465 seconds) and increase propellant mass to 28 metric tons while inert mass only increases to about 4 metric tons (pmf ~0.88).

Ariane 5 Hypergolic Upper Stage[1]

A hypergolic stage (EPS) that is used as a simple alternative to the ESC-A on Ariane 5 launches, this stage has a much lower specific impulse at 324 seconds. Further, the propellant mass is only 10 metric tons, while the burnout mass is low at slightly more than 1 metric ton (pmf ~0.88).

Cryogenic Propulsion Stage (CPS) Concepts

Likely candidates for a chemical-based in-space transportation system were also included in the trade space. A CPS is included in most current NASA exploration plans; however, the trade space surrounding this system is still very large. In order to provide a representative system for this evaluation, two CPS options were considered: a Next Generation Engine based solution and a J-2X based solution.

CPS using the Next Generation Engine (NGE)[4]

This set of CPS concepts utilizes an advanced in-space engine that achieves a superior specific impulse to any liquid rocket engine ever developed (greater than RL-10B2). These concepts may utilize more than one NGE depending on the total propellant mass and acceleration requirements. While diameter and length vary according to concept, most are constrained by either an 8.4 meter (potential SLS shroud diameter) or 10 meter diameter while length varies with total propellant load. The values used as a representative CPS concept for this assessment include an approximate 68 metric ton propellant mass and a 12 metric ton inert mass (pmf ~0.85). Specific impulse was captured at two design settings, 455 seconds and 465 seconds.

CPS using the J-2X[4]

Another engine system alternative for CPS is the J-2X liquid rocket engine that is currently in development testing at NASA Stennis Space Center. Relative to the NGE, J-2X is expected to deliver increased thrust (for the single engine configurations considered) with a decrease in efficiency (Isp ranges from 435 to 448 seconds for J-2X concepts; 440s used). For this evaluation, the same stage parameter for propellant mass was used, while the inert mass was increased slightly to account for this larger engine (inert mass ~16 metric tons; pmf ~0.81).

Table III.1: High Level Summary of Stage Options

Stage	Propellant (t)	Inert (t)	pmf	Isp (s)
Centaur	21	2	0.91	450
DIV 4m US	20	3	0.84	460
DIV 5m US	27	3.5	0.89	460
F9 2nd Stage	49	3	0.94	340
H-IIB 2nd	17	3	0.85	450
Ariane ESC-A	15	3.5	0.81	445
Ariane EPS	10	1	0.88	324
CPS (NGE)	68	12	0.85	455-465
CPS (J-2X)	68	16	0.81	435-448

IV. Single Launch Architecture Design Options

Historically, most space flight missions have been accomplished with a single launch profile. That is, a single launch vehicle delivers payload to either an Earth orbit or its beyond-LEO initial energy state. Even in the human space flight realm, most of the Mercury, Gemini, Apollo, and Shuttle missions only required one vehicle launch to achieve the objectives of the mission. Notable exceptions are the rendezvous and docking missions of Gemini and the overall assembly and crew rotation process used for Earth orbiting stations (Skylab, Mir, ISS). Although currently envisioned near Earth asteroid (NEA) and Mars missions will require more than one launch due to large intitial mass requirements, it is reasonable to assume that initial SLS missions or science applications will be accomplished with a single launch. There are a variety of ways to undertake these single launch mission scenarios. Four methods were evaluated: direct injection, single in-space stage with fixed dropoff, multiple in-space stage with fixed dropoff, and single in-space stage with variable dropoff.

Direct Injection by Launch Vehicle

The simplest of all cases, the direct injection method, implies that the launch vehicle alone delivers the spacecraft to its final departure energy (i.e. no additional in-space stages are required). An optional Earth staging orbit could be included to provide a brief spacecraft checkout period, but this assessment did not include such a feature. From launch, the trajectory is optimized to deliver the maximum possible payload to a characteristic energy (C3) sweep that ranges from LEO (C3 approximately equal to -60 km^2/s^2) to a reasonable Jupiter system transfer orbit (C3 approximately 90 km^2/s^2). While this is a simple case from an operational complexity perspective, the larger stages that launch vehicles typically require cause rapid performance degradation at high energies.

Single In-Space Stage; Fixed LV Dropoff

This option is slightly more complex than the direct injection method. It requires that the stage and payload be integrated and encapsulated for the duration of the Earth-to-orbit ascent phase of the mission. These “payloads” are placed into a common “fixed” orbit by the launch vehicle (in this case assumed to be -87 km x 241 km at 29^o inclination). This allows the launch vehicle to have a very specific, rigidly defined mission requirement, and the payload and in-space stage to have very specific and predictable initial conditions from which to begin their operational phase. Further, performance will degrade much less rapidly than with the direct injection mission, but the optimal coupling of launch vehicle and in-space stage performance may not be realized. An overview of this method is depicted in Figure IV.1

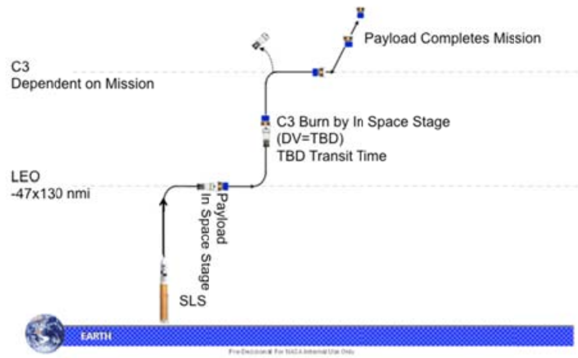

Fig. IV.1: Single In-Space Stage; Fixed LV Dropoff Mission Profile

Multiple In-Space Stages; Fixed LV Dropoff

In order to more optimally match the performance of the launch vehicle to the size and performance of the chosen in-space stage, we considered the effect of using multiple stages of a given type (no mixing of stage types was assessed). This is a more complex method from an overall integrated architecture perspective, and these considerations (e.g., ground processing, vehicle integration, fueling of more than one stage among others) were not fully developed during this performance-based assessment. As shown in Figure IV.2, a common, fixed dropoff orbit by the launch vehicle (-87 km x 241 km at 29^o inclination) was assumed for this type of assessment as well. The optimal staging sequence was approximated by assessing concurrently burning stages and serially burned & disposed of stage arrangements. Maneuvers were performed at perigee, with a slight adjustment required in order to raise it to an acceptable altitude in order to avoid stage reentry.

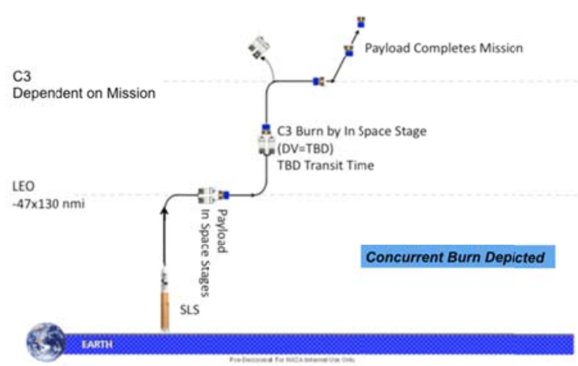

Fig. IV.2: Multiple In-Space Stages; Fixed LV Dropoff Mission Profile (Concurrent Burn)

Single In-Space Stage; Variable LV Dropoff

The final architectural option assessed was the full, integrated optimization of launch vehicle and in-space stage performance. In this case, the dropoff condition of the launch vehicle was allowed to vary through the usage of highly elliptical orbits. In this manner, we identified the optimal initial energy state was found from which the in-space stage and payload in their operational phase. These highly elliptical orbits ranged from LEO altitudes (-87 km x 241 km) up to orbits of Earth-Moon type of distances (-87 km x 400,000 km). Basic assumptions for these cases included perigee being fixed at -87 km, inclination fixed at 29 degrees, and optimization of apogee based on the final energy and payload mass. Additionally, the perigee was adjusted at apogee in order to avoid stage reentry. This mission profile is depicted in Figure IV.3:

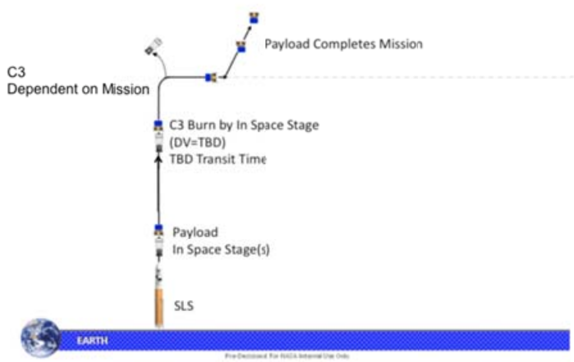

Fig. IV.3: Single In-Space Stage; Variable LV Dropoff Mission Profile

V. Overview of Beyond LEO Mission Assessment

As discussed in previous sections, we have defined a multivariable trade space that includes launch vehicle performance, in-space stage size and capability, and mission design considerations. While a very specific design reference mission was not assessed, targets for beyond LEO performance were set. These targets were expressed as desired performance (mass in metric tons) delivered to a desired escape energy state (expressed as C3). For example, a crewed lunar flyby could be achieved by delivering about 25 metric tons to a C3 = -1.8 km^2/s^2. A lunar lander could be delivered to the lunar surface with about 40 metric tons to the same C3, while providing the crew and capsule on the same flight would require at least 60 metric tons. Other low-energy points along the C3 curves include Earth escape or the Sun-Earth L2 point (C3 ~0 km^2/s^2), Mars transfer (C3 ~9 km^2/s^2), asteroid belt object transfers (C3 ~40 km^2/s^2), Jupiter transfers (C3 ~80 km^2/s^2), Saturn transfers (~107 km^2/s^2), and growing progressively larger up to solar system escape transfers (C3 > 150 km^2/s^2). Based on recommendations from the Planetary Science Decadal Survey, a Jupiter/Europa mission concept (which requires about 5 metric tons of launch vehicle performance to a Jupiter transfer) was evaluated.

In order to establish a reference performance threshold, the launch vehicle with no in-space stage option was assessed ("Direct Injection by Launch Vehicle" case). Several concepts were chosen that provided representative payload delivery capabilities to LEO (approximate equivalencies shown at C3 -60 km^2/s^2 in Figure V.1). The key conclusions from this assessment include the following:

1) Only very high performing vehicles were capable of delivering 5 metric tons to a direct Jupiter transfer (i.e. no gravity assists).
2) With increasing C3, vehicles that utilized smaller final stages performed better than vehicles that had either large final stages or did not include 2nd Stages.

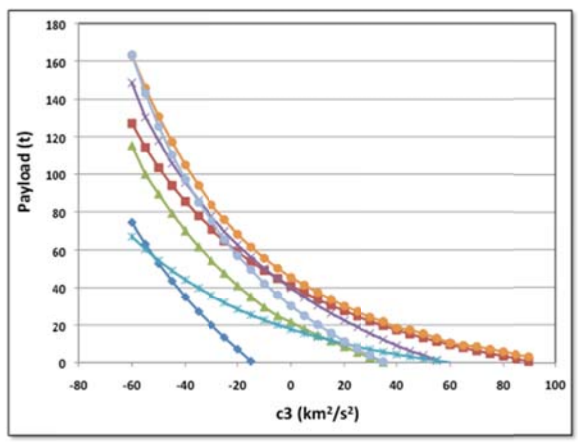

Fig V.1: Representative performance capabilities across a range of C3 energy. Vehicle configurations varied, but were not considered the purpose of this study. Therefore, configuration specific curves are not identified.

For the "Single In-Space Stage w/ Fixed LV Dropoff" mission, LV capabilities were assessed at 70, 100, 130, and 160 metric tons. As shown in Figure V.2, the 70 metric ton vehicle performs well with lower mass, higher efficiency LOX/LH2 stages. Most of the cases deliver similar performance to higher energies (e.g. the Europa mission capture space), while none deliver > 25 metric tons to a lunar flyby scenario. The curves in this case are truncated as C3 decreases (mass increases) at the point when the total stage mass and payload mass equal the capability of the launch vehicle (70 metric tons in this case).

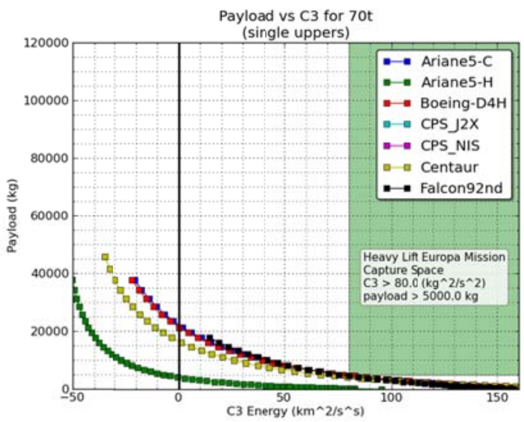

Fig. V.2: Beyond LEO performance of various in-space stage options with a 70t launch vehicle from a fixed initial point.

Unfortunately with this fixed initial point rigidly defined, the launch vehicle is severely underutilized. The only performance increase that is realized is in the growth of the initial mass capability. This allows the curves that are depicted in Figure V.2 to grow to the left and stage options that are greater than 70 metric tons to enter the trade space. Figure V.3 depicts the performance curves for the 100 metric ton launch vehicle scenario. It can be seen that the performance values depicted in Figure V.2 are exactly the same (i.e. a point that is depicted in Fig. V.2 does not change when going to Fig. V.3). This is due to the launch vehicle contribution to energy being the same. Additional points are depicted though because more mass is available at that initial condition. Also in Figure V.3, CPS is included due to this mass availability increase. It is the highest performer because of the extraordinary propellant mass that it has available.

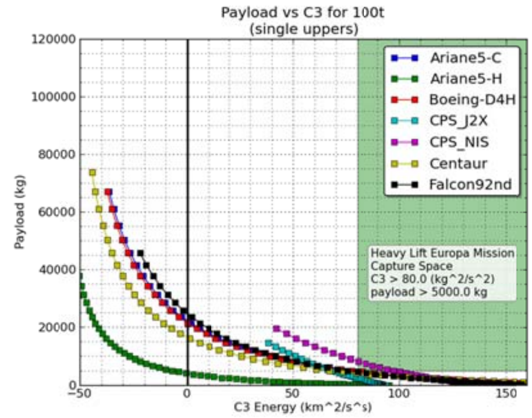

Fig. V.3: Beyond LEO performance of the same stages as V.2 with launch vehicle capability growing from 70 up to 100 metric tons

When proceeding to the 160 metric ton launch vehicle scenario, a continuation of this phenomenon is depicted. The curves are merely extended up to the point where the total stage mass and payload mass is less than or equal to 160 metric tons. This is shown in Figure V.4:

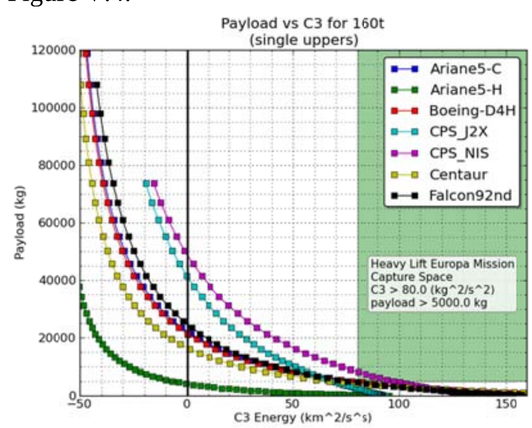

Fig. V.4: Beyond LEO performance of the same stages as V.3 with launch vehicle capability growing from 100 up to 160 metric tons

As shown in Figure V.4, the only case that can deliver greater than 40 metric tons to a lunar flyby scenario are the CPS cases (especially when using the highly efficient NGE or "National In-Space (NIS)" engine as shown). This is also the only case that delivers > 5 metric tons to a C3 = 80 km^2/s^2.

If for technical or other programmatic reasons the launch system becomes constrained in delivering payloads to a rigidly defined dropoff condition, a method that would utilize most of the capability of the system is through the usage of more than one existing in-space stages. Volumetrically, a 10-meter shroud would allow at least two of every existing stage considered for this assessment, and in most cases 4 or more stages can be accommodated. Even an 8.4-meter shroud would allow two or more stages in almost every case. Only one CPS can be accommodated within the volumetric constraints.

When assessing performance, it can be assumed that the stages either burn in a concurrent fashion (preferred from an operational perspective) or in a serial fashion (preferred from a performance perspective). Figure V.5 is a depiction of the performance when multiple Centaur stages are used in a concurrent fashion on a 160 metric ton vehicle. This figure shows that over 50 metric tons can be delivered in the lunar flyby scenario and over 20 metric tons can be delivered in the Jupiter transfer scenario. This represents an ~300% performance increase over the single stage case and the launch vehicle direct injection case.

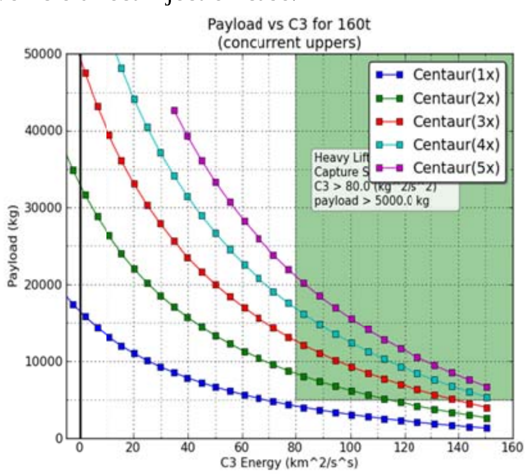

Fig. V.5: Performance with concurrently burning Centaurs. While 6 can be lifted with a 160 metric ton vehicle, only 5 fit within a 10-meter shroud.

Additional performance is available from the serially-burning scenario if the operational considerations can be satisfied. This over 400% performance increase would have to be balanced against the additional burn times, multiple passes of perigee and staging orbit changes, mass and thrust imbalances, multiple stage jettison events, and a host of others. An ideal scenario for multiple stages would be diametrically opposed burns (2+2+1 for the five Centaur case) to alleviate some of the issues. This case should be considered an upper bound on performance rather than a specific recommendation.

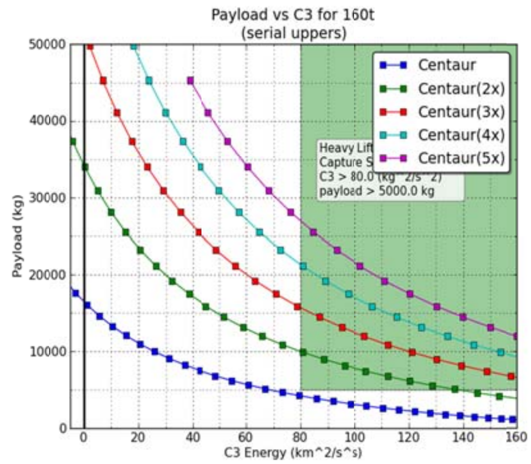

Fig. V.6: Performance with serially burning Centaurs. Again, only 5 can be accommodated within the largest shroud option being considered.

The Centaur stage exhibited the highest performance due to its diameter allowing many stages to be considered, its high specific impulse and stage efficiency, and its ability to provide the most optimum "denominator" with respect to launch vehicle performance. In other words, it was the best solution found when examining multiple stages for allowing the launch vehicle to be fully utilized.

Another method to fully utilize the launch vehicle is allowing it to optimize the dropoff orbit based on the payload that it carries and the final energy required for that payload. In this manner, a single in-space stage can be used because the majority of the energy is provided by the launch vehicle, while the final injection is provided by an efficient and light in-space stage. For this scenario, launch vehicle performance as a function of final energy is determined. Once this is known, the final stage and maximum payload is run through the desired final energy sweep. As this process is looped over the range of vehicle dropoff conditions, an optimum curve is created that shows the progression through final energy states of the launch vehicle.

As expected, the maximum payload at specific C3 energies occurred when the maximum potential of the launch vehicle was used (i.e. the point at which the curve would otherwise truncate). These points were captured in order for the optimum dropoff variability to be characterized. Figure V.7 depicts how the dropoff for lower energies (final C3 < 0 km^2/s^2) is very low on the dropoff energy scale (dropoff C3 < -45 km^2/s^2). As final energy increases (final C3 > 120 km^2/s^2), the dropoff C3 energy increases accordingly (approaching a dropoff C3 = 0 km^2/s^2). Allowing this launch vehicle dropoff condition to shift increases performance significantly. The fixed launch vehicle dropoff case is also shown in Figure V.7 for reference. A performance increase of over 250% (~18 metric tons) is realized for the Jupiter transfer case, while over 100% increase (~50 metric tons) is delivered to a lunar transfer.

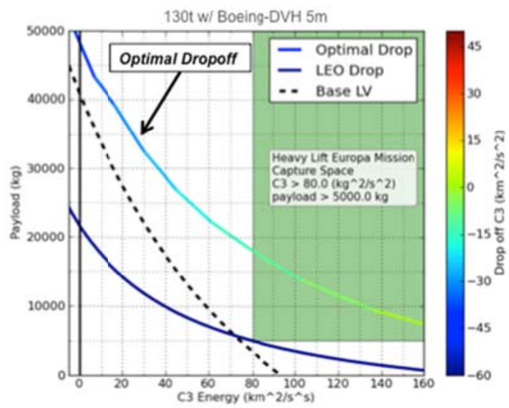

Fig. V.7: Performance characteristics of the variable dropoff case. Results show a large positive performance increase over both the "Direct Injection by LV" and the "Single Stage; Fixed LV Dropoff" case.

Additionally, the orbit changes that are required to achieve the desired intermediate staging energies were captured. Assuming an initial perigee shift from -87 km up to 241 km, the apogee was determined according to the required launch vehicle delivery energy. These highly elliptical orbits varied from 241 km x 1,852 km for a staging energy of -52 km^2/s^2 up to 241 km x 167,000 km for a staging energy of -5 km^2/s^2.

VI. Application to Jupiter Mission

In-Space Mission Design Overview

In order to apply vehicle design options to missions to the Jovian system, energy requirements to Jupiter must be determined. There are several ways to judge the energy requirements for inter-planetary transfers and many levels of detail to be considered. Since the relative performance of vehicles is more important than absolute performance in this study, a high-level model was used. The main consideration is to find trajectories in future years that offer the lowest energy requirements but which feature short-duration direct routes. In order to explore a vast number of possible transfers and determine duration and energy requirements, a tool was developed to automate the process.

Trajectory Tool[5]

The basic problem of finding the shortest direct transfer between two planets has been documented and written into several types of software systems. Based on its function, ease of use, and adaptability, the PyKEP library developed by the European Space Agency was chosen as the trajectory-solving tool. PyKEP is a sophisticated, C++ library that can solve the multiple revolutions Lambert's problem, has efficient Keplerian propagators, and utilizes Taylor integrators. The name comes from the fact that the entire library is exposed to the Python programming language. Because of the Python language support, it is a simplified process to design and implement a tool that iterates over desired launch dates and catalog departure energies.

Mission Opportunity Assessment

For a given year, various trajectory opportunities exist with an arbitrary launch date and transfer duration. For the purposes of this study, launch opportunities were sampled every 5 days, and durations were sampled from 1 to 10 years every 0.1 year. Of the resulting 6,571 possible trajectories, the 100 with the lowest departure energy were chosen for plotting and further analysis. See Figure VI.1.

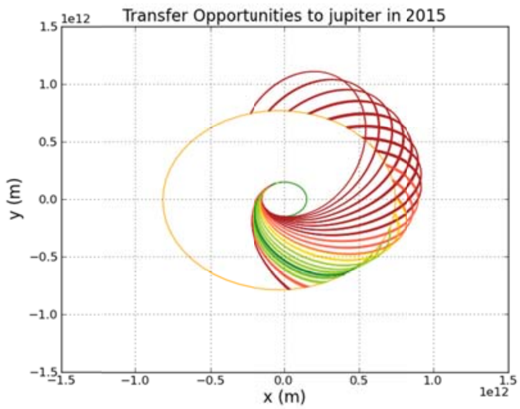

Fig. VI.1. Best 100 transfer trajectories from Earth to Jupiter ranked on departure energy. Axes represent the ecliptic plane. All launch dates were in 2015. Color contour is based on delta-v required to capture into the Jovian system. Green is 6 km/s, red is 9 km/s.

Repeating this process over many years results in a general sensitivity to energy requirements vs. trip duration. Color contouring can also be applied to determine the trends associated with the delta-v required to achieve capture in the Jovian system. See Figure VI.2.

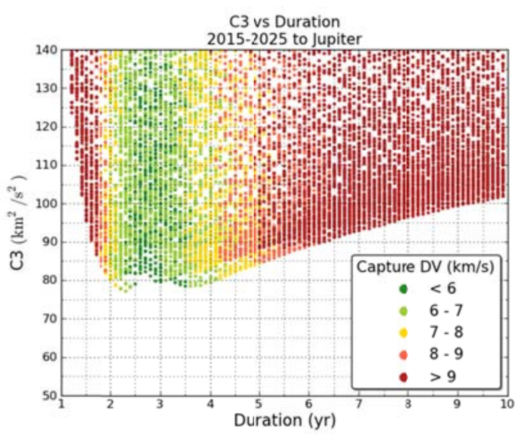

Fig. VI.2. Scatter plot of C3 energy (km^2/s^2) versus transfer duration (yr) for transfers between 2015 and 2025.

From Figure VI.2 we see that the lowest energy opportunities (~77-82 km^2/s^2) take on the order of 2-4 years for outbound transit, but these transfers are very sparse. More launch opportunities exist if planning for up to 90 km^2/s^2. Energy requirements begin to become unreasonable at or below 1.5 year outbound transit.

Affordability Impacts Overview

For missions to the Jovian system, the standard approach is to utilize an Atlas 5 launch vehicle. This approach requires the use of a 6-year Venus-Earth-Earth Gravity Assist (VEEGA) trajectory in order to meet the mass requirements for the mission. The eventual stay at Jupiter based on this approach is on the order of 2.5 years, which includes a large initial elliptical capture orbit in order to further reduce the delta-v requirements.

By using a 3-year direct transfer rather than the baseline approach, 3 years can be eliminated from the operations window from the transfer duration alone. Further, this approach potentially allows for a further 1 year reduction in the total tour duration due to the increased mass availability for the spacecraft. These duration savings result in a total mission time of less than 4.5 years. The reduction from 8.5 years to 4.5 years could save up to $300M in operational costs. With a sufficient capture stage, outbound trips can be reduced to 2 years further cutting total mission costs. In addition, the payload design and fabrication could save up to $300M due to reduced radiation mitigation requirements.

Because of the high payload mass to a C3 of 80 km^2/s^2, the payload spacecraft can have a much simpler and cheaper design. Higher mass budgets permit the use of heavier heritage subsystems and components. Higher mass margins in design also afford a streamlined development cycle that compresses design & development time and costs. Rather than simplifying spacecraft design, another option is to utilize the extra mass allocation for multiple spacecraft on a single launch, thereby greatly improving odds of mission success through redundancy (e.g., Spirit and Opportunity Rovers).

VII. Final Study Conclusions

This study has shown the potential to provide an optimum system solution for single launch, beyond LEO missions if the dropoff condition of the launch vehicle is allowed to vary based on mission need. This variable dropoff scenario shows significant performance increases over both the “Direct Injection by Launch Vehicle” and “Single In-Space Stage; Fixed Dropoff” cases. It also has many advantages over the “Multiple In-Space Stages; Fixed Dropoff” case. These include both the elimination of operational & integration complexity as well as removing the need to purchase many stages over the life of the exploration program.

Additionally, the variable launch vehicle dropoff method allows for existing stages to be used in the capture of near-term, meaningful mission scenarios including crewed lunar flybys and high-priority science missions. This alleviates the need for NASA to begin a large in-space stage development program in the near term and allows for a competitive procurement approach & subsequent development of strategic partnerships across the industry.

Finally, by being more flexible with the dropoff conditions, we were able to reduce the Europa mission scenario by up to 5 years and provide mass margin that could be used to greatly reduce spacecraft complexity. These improvements would result in an estimated $1B cost savings for this high-priority science mission.

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[5] http://keptoolbox.sourceforge.net/

SPACE LAUNCH SYSTEM MISSION FLEXIBILITY ASSESSMENT

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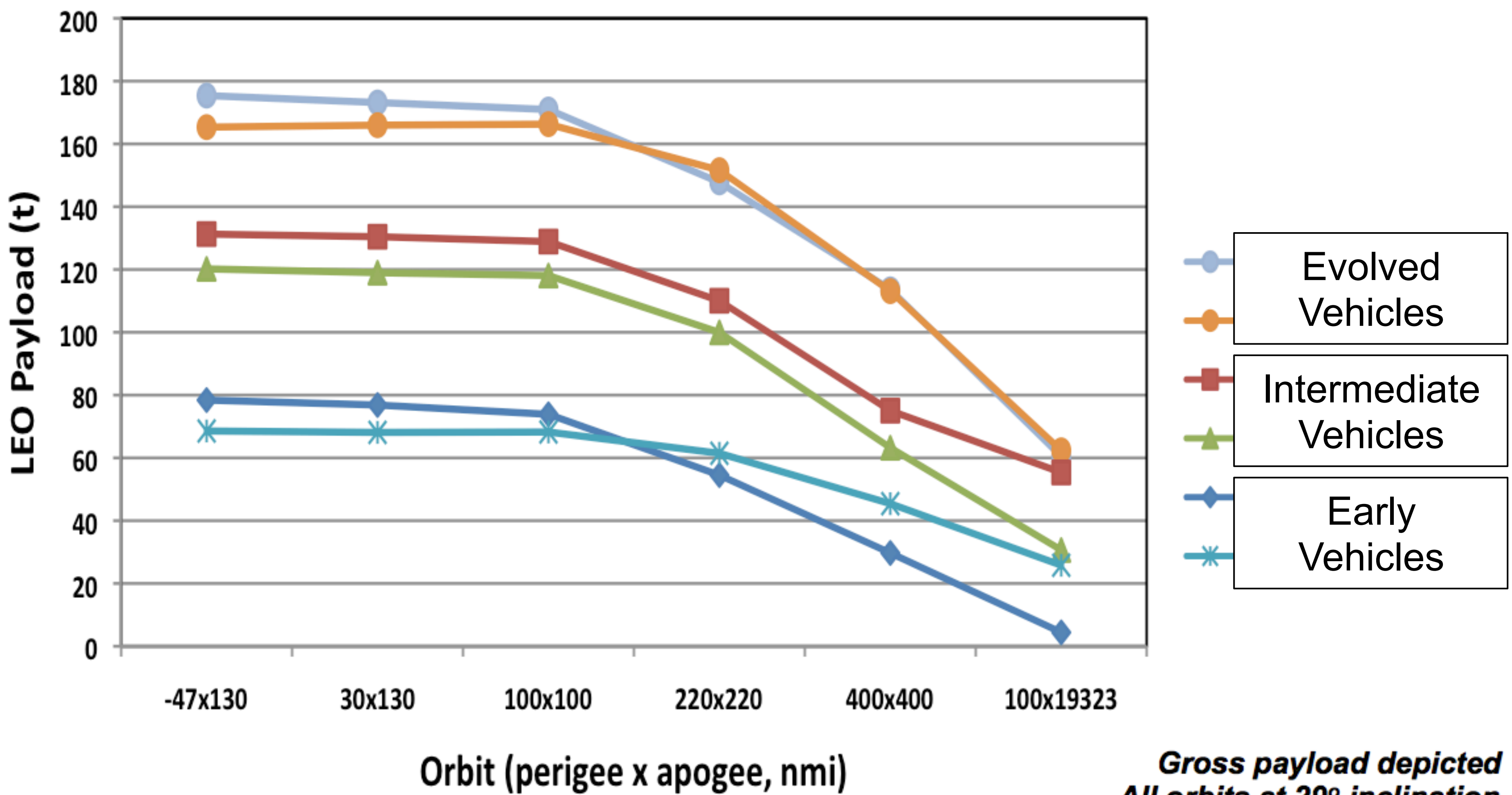
Beyond LEO Mission Options

1. **Direct inject of payload using SLS "base vehicle" only**
   - Easy trades, limited to what SLS configuration can provide
2. **Drop-off payload with an in-space stage into LEO**
   - Large trade space with all possible in-space stages options
   - SLS dropoff point: -47x130 nmi @ 29^o
   - # of additional in-space stages: one
3. **Drop-off payload with multiple in-space stages into LEO**
   - SLS dropoff point: -47x130 nmi @ 29^o
   - # of additional in-space stages: multiple
   - Two options for analysis:
     a. Burn all stages concurrently
     b. Burn stages in series, dropping spent stages
4. **Drop-off payload with an in-space stage into LEO**
   - Large trade space with all possible in-space stages options
   - SLS drop-off point: variable (optimized based on mission)
   - # of additional in-space stages: one

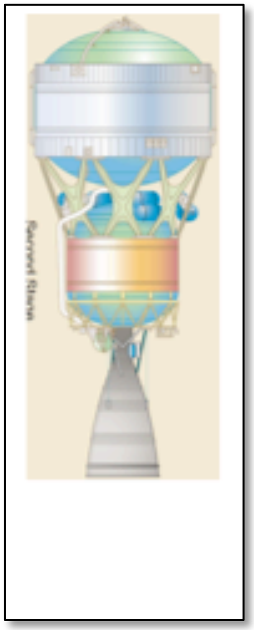
LEO Performance Characterization



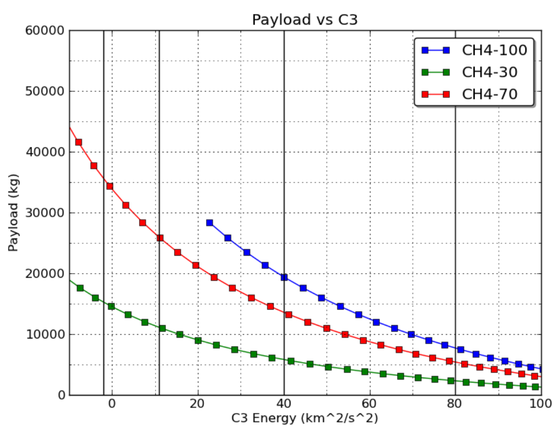
Gross payload depicted
All orbits at 29º inclination

All vehicles exhibit small performance deltas at lower altitudes.
As altitude increases smaller upper stages exhibit better performance trends

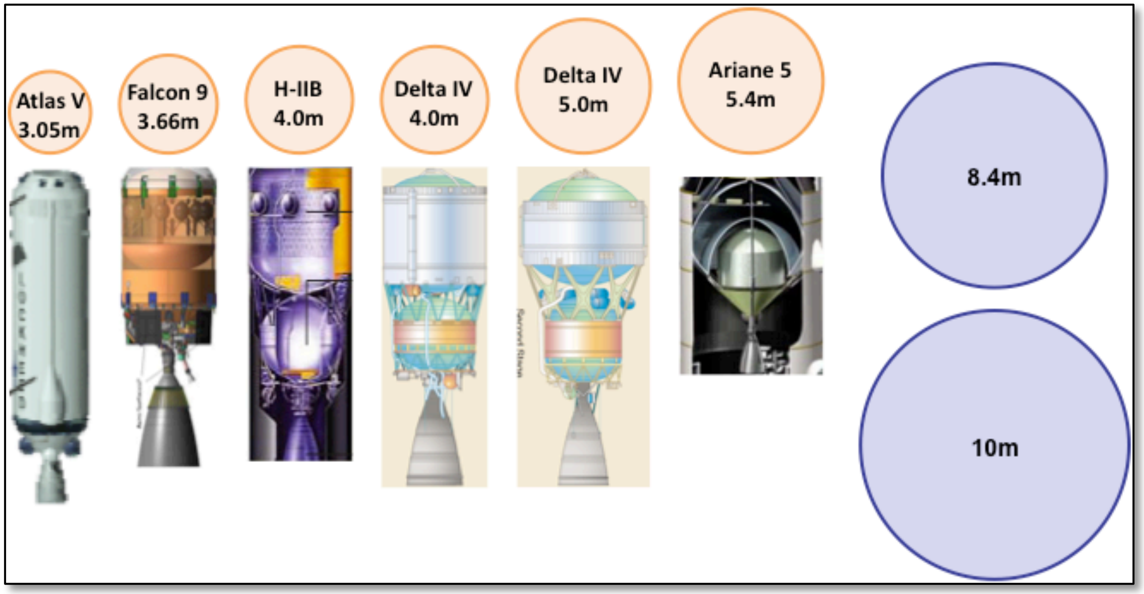
Optimizing an In-Space Stage



- **Propulsion drives performance:**
  - Specific impulse is key driver (thrust can play a roll)
  - Special characteristics (i.e. restartability, etc.)
- **Gather assumptions for determining success:**
  - Destination/mission profile
  - Payload requirements
  - Loiter duration/functional requirements
  - Many others
- **Performance assessment**
  - Calculate delivered payload to desired delta-V
  - Try all possible in-space stage configuration options
  - Determine final delivered energy for all cases
  - Plot payload and energy and compare to success criteria
  - Use data, trends, and requirements to determine winners

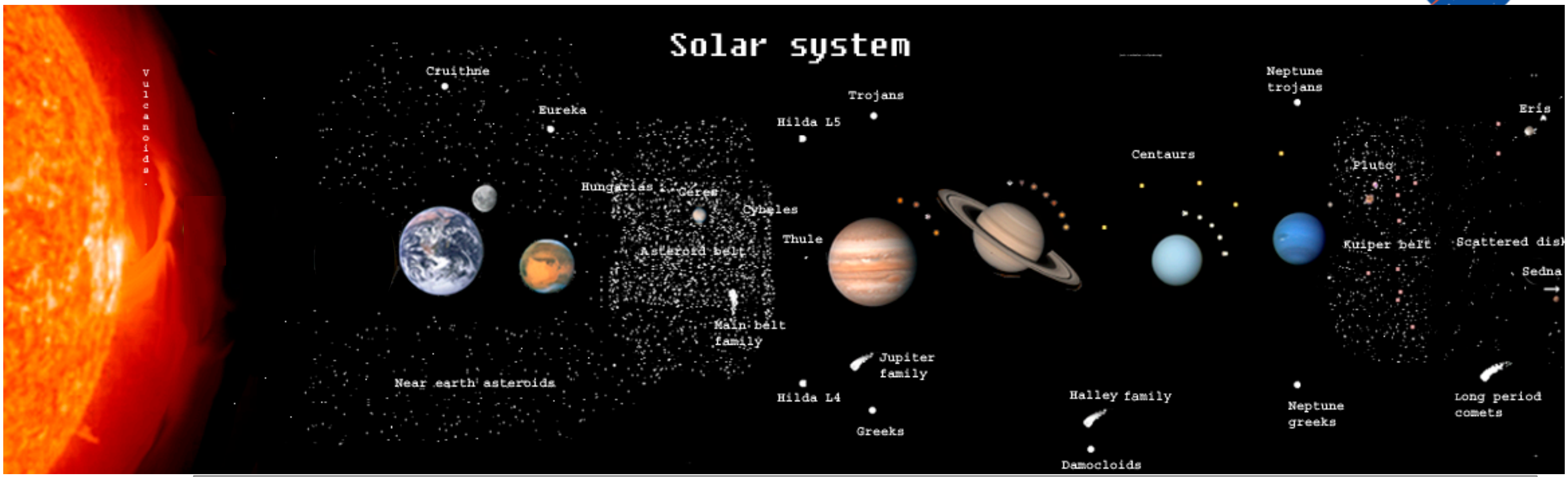


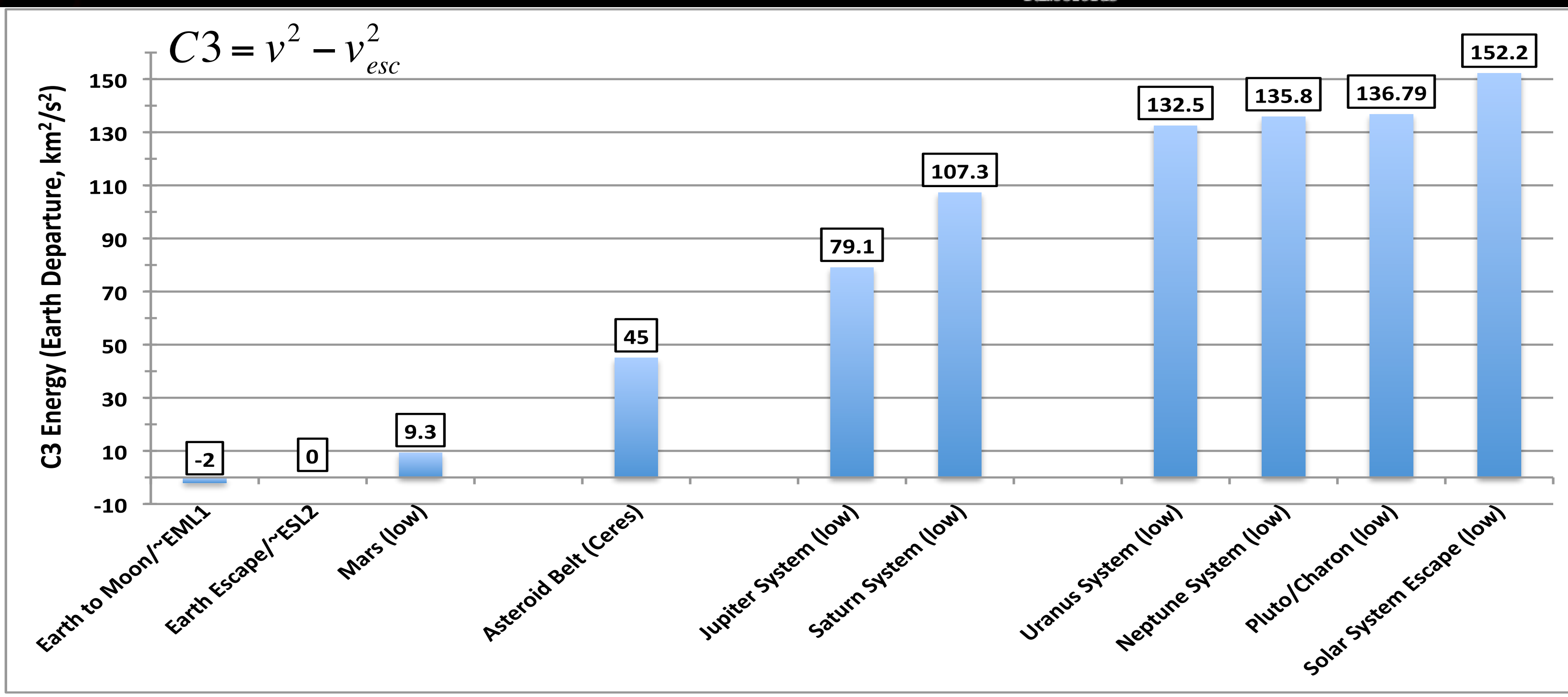
Overview of Upper Stage Options



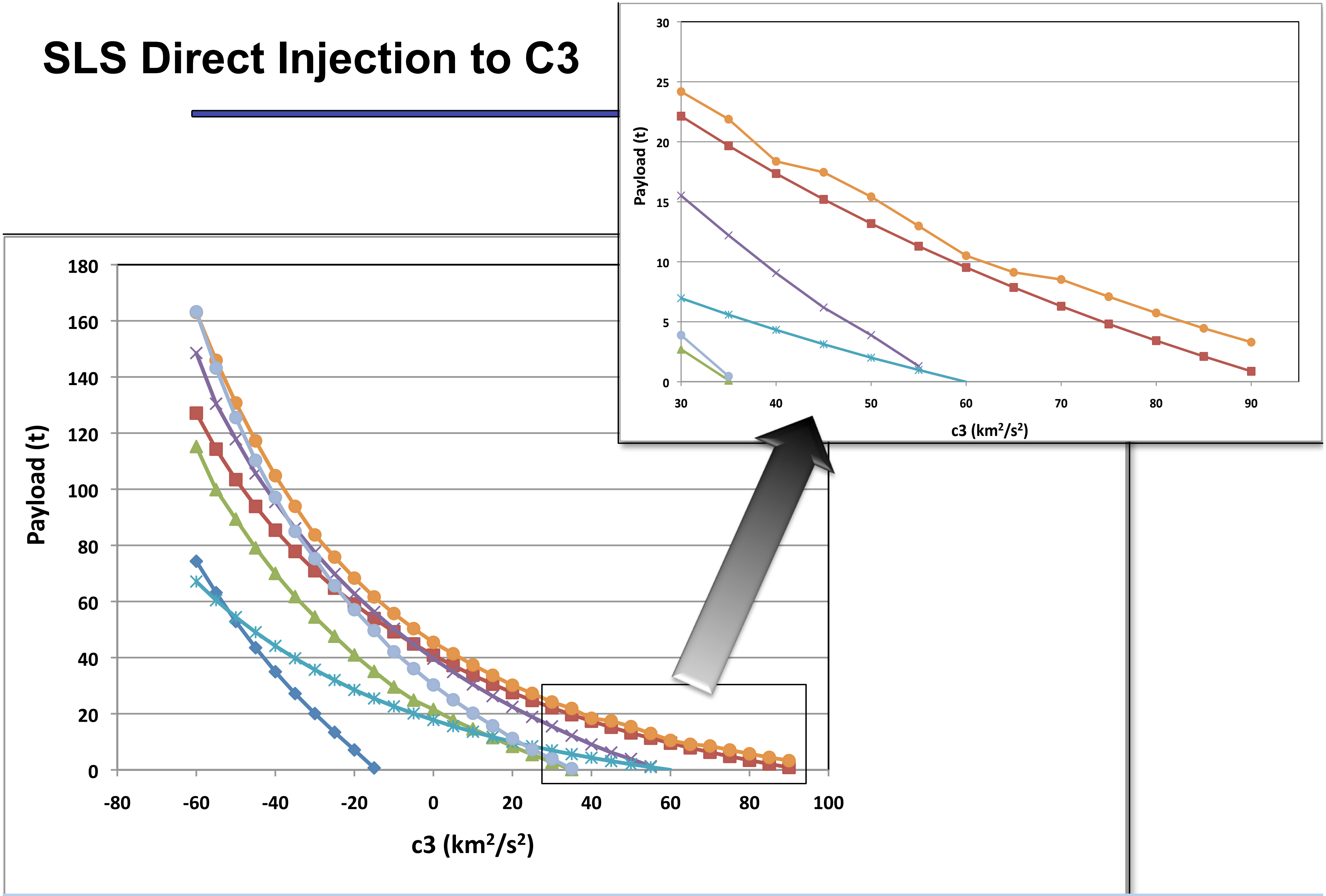
Stage	Propellant (t)	Inert (t)	pmf	Isp (s)
Centaur	21	2	0.91	450
DIV 4m US	20	3	0.84	460
DIV 5m US	27	3.5	0.89	460
F9 2nd Stage	49	3	0.94	340
H-IIB 2nd	17	3	0.85	450
Ariane ESC-A	15	3.5	0.81	445
Ariane EPS	10	1	0.88	324
CPS (NGE)	68	12	0.85	455-465
CPS (J-2X)	68	16	0.81	435-448

Beyond LEO: Solar System C3 Map



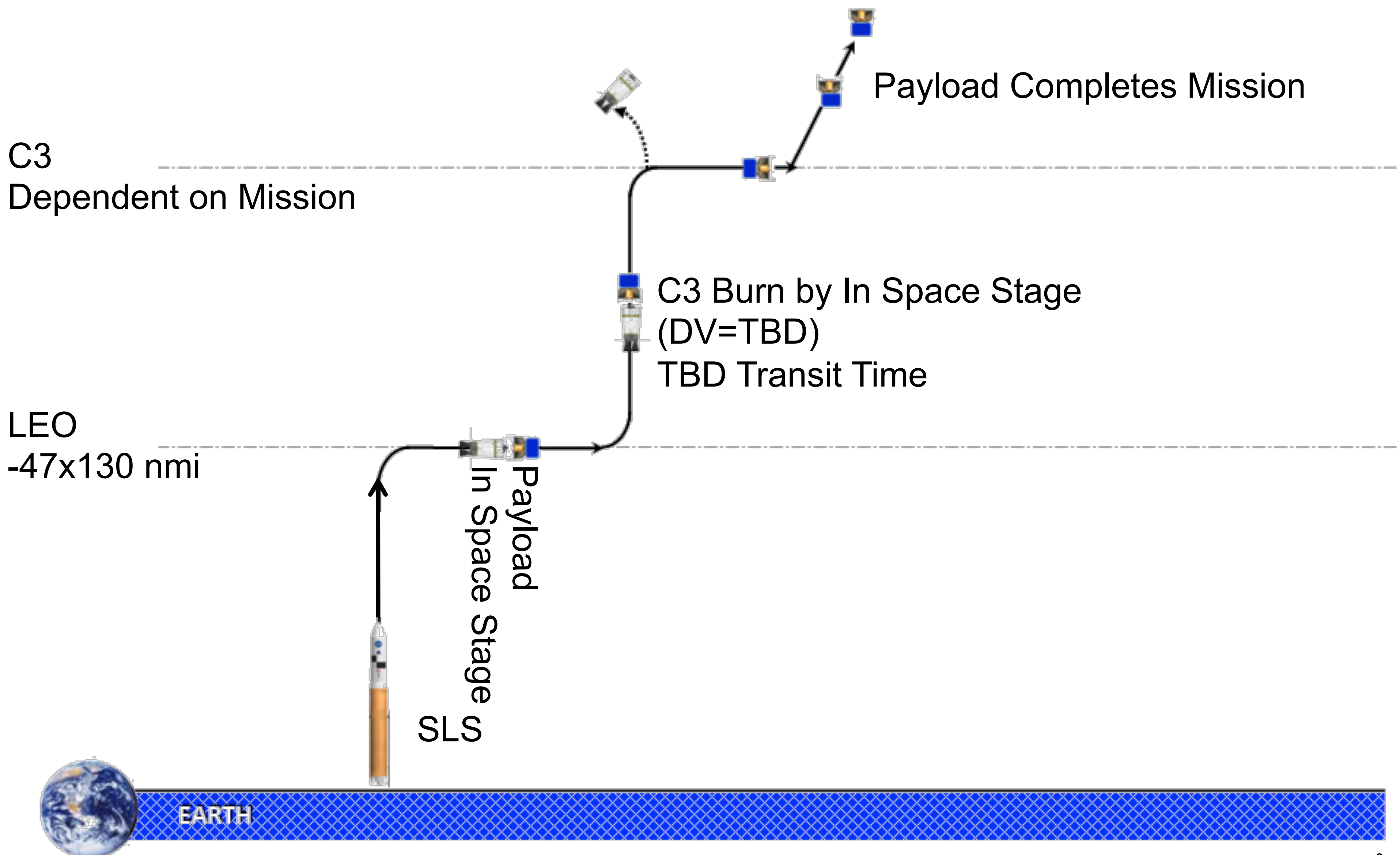


SLS Direct Injection to C3



LV Direct injection does not deliver a high payload to a high energy C3

LV Drop-off w/ Single Stage Burn to C3



In-Space Stage Performance Keys

Example in-space stage performance curve.
LV injects 160t at a c3 of -61.8 (LEO).

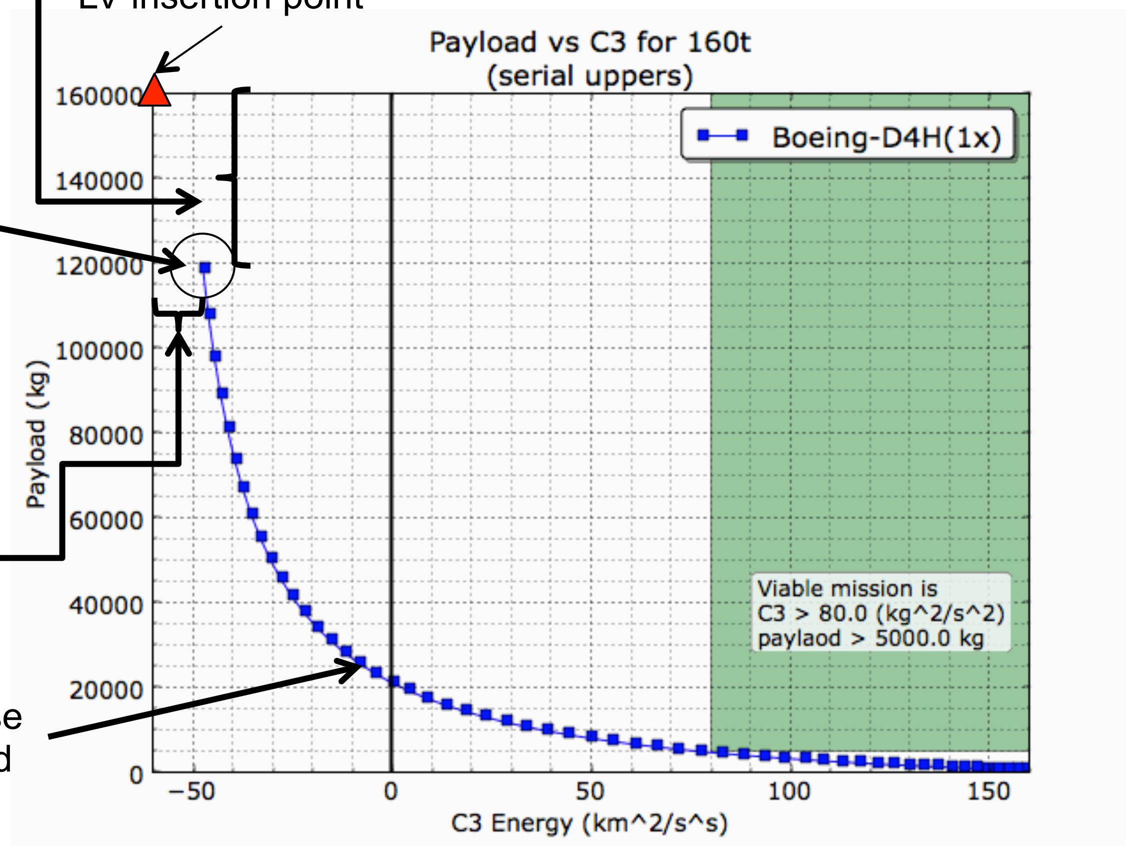
In-space stage wet mass plus losses.

LV insertion point

Key point:
Shows maximum payload for in-space stage. To get this point higher the LV would need to deliver more mass to orbit.

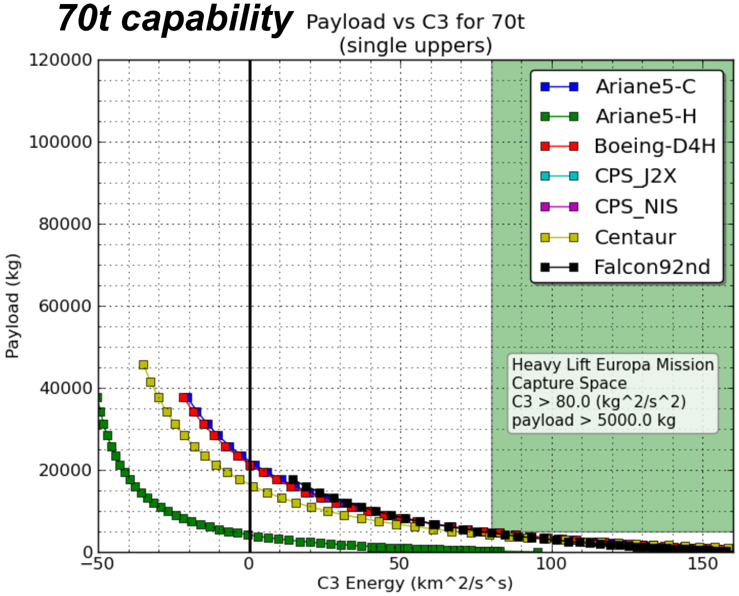
C3 gained by burning the in-space stage. To move point more to the right would require higher Isp.

Data points generated by step-wise reduction in final delivered payload

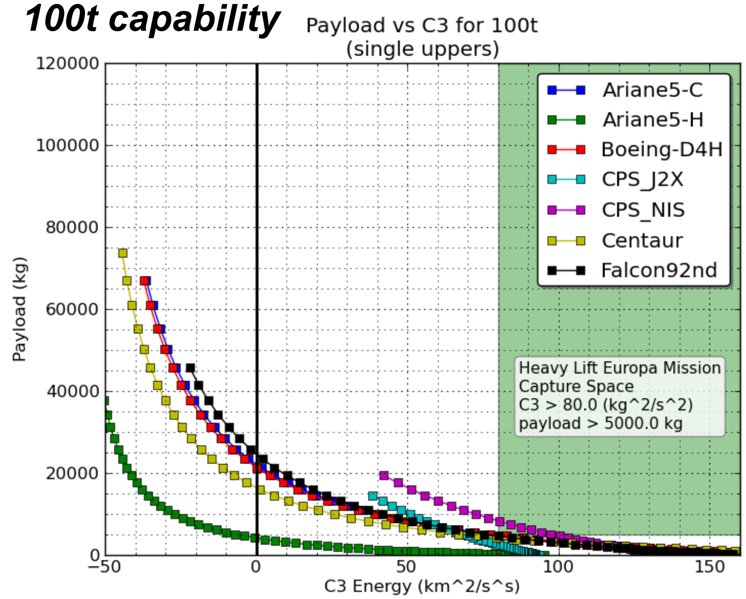


Single Injection Stage

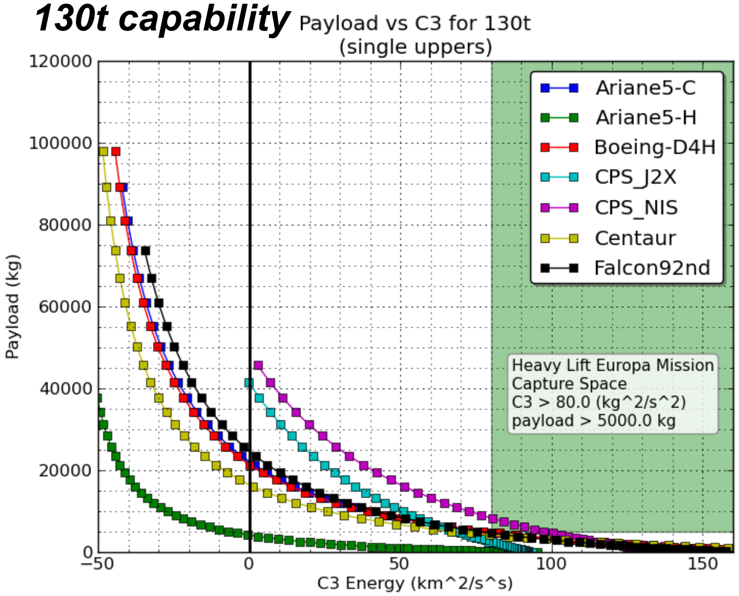
70t capability



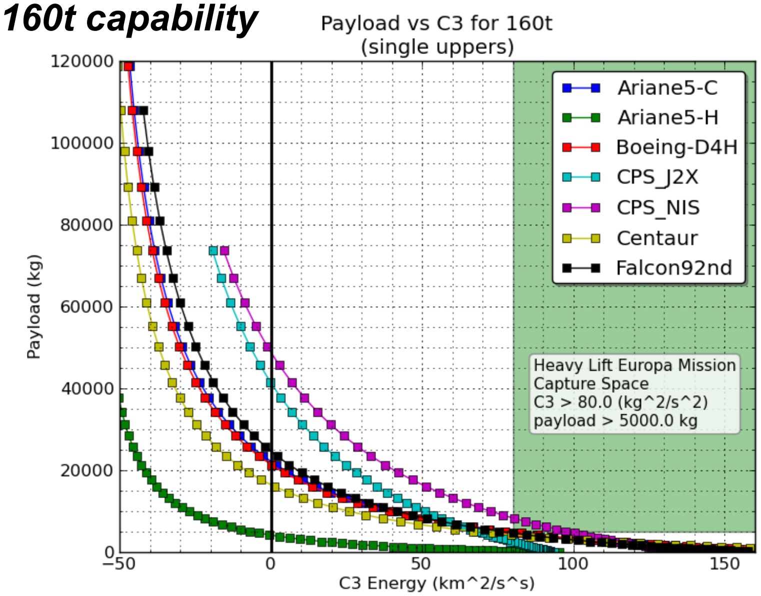
100t capability



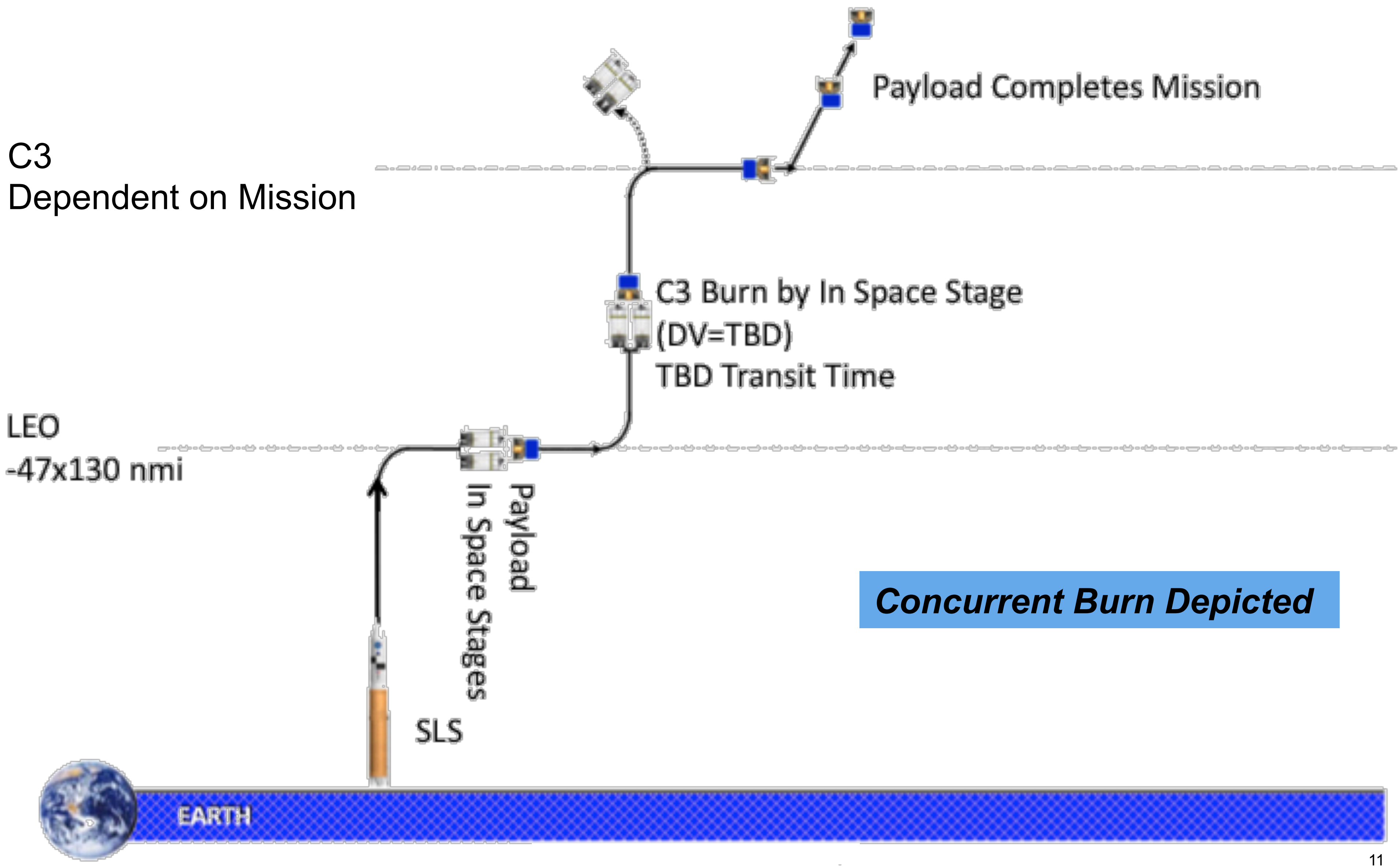
130t capability



160t capability



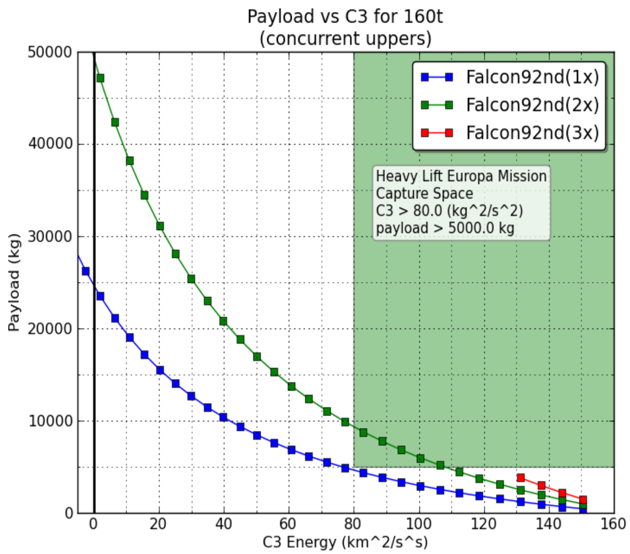
LV Dropoff w/ Multiple Stages Burning to C3

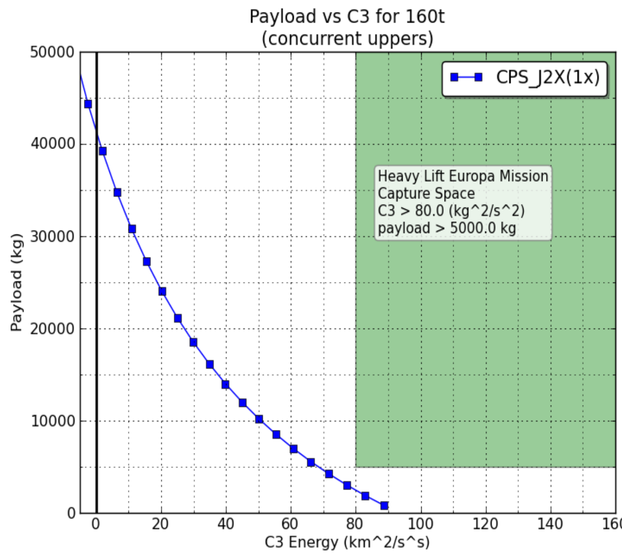


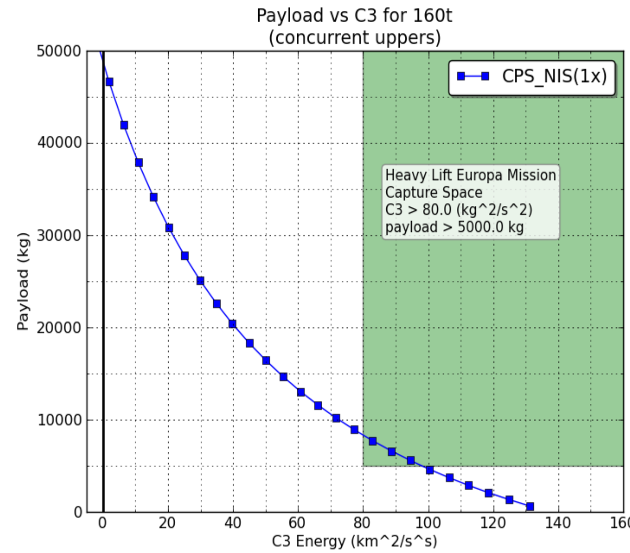
Concurrent Burn Depicted

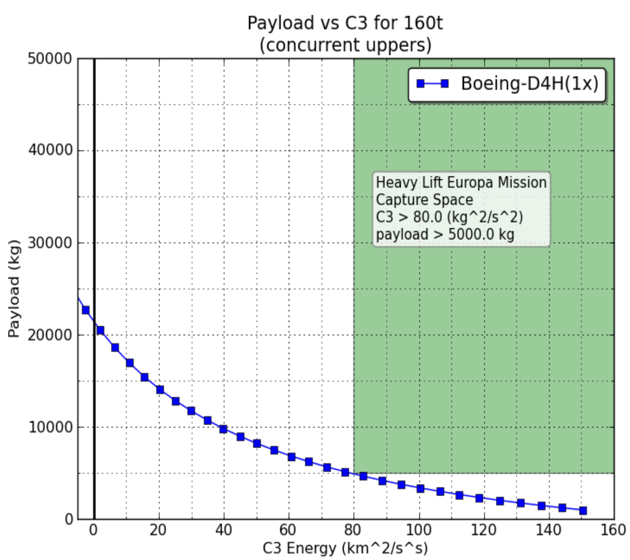
SLS Dropoff = -47x130 nmi @ 29º

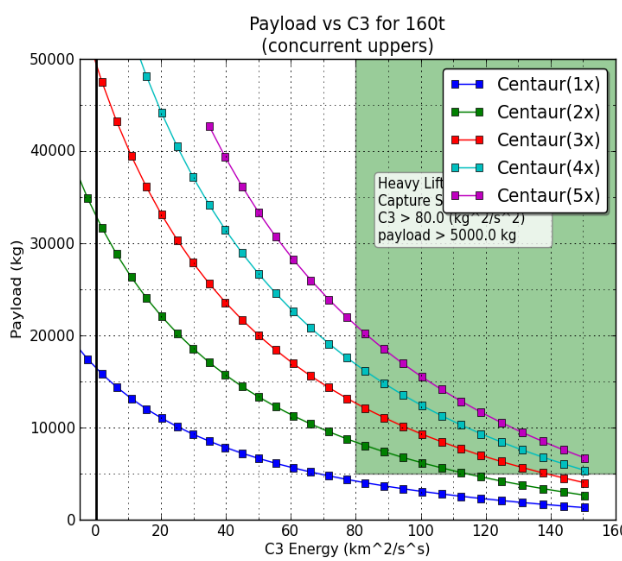
Multiple In-Space Stages: Concurrent Burn

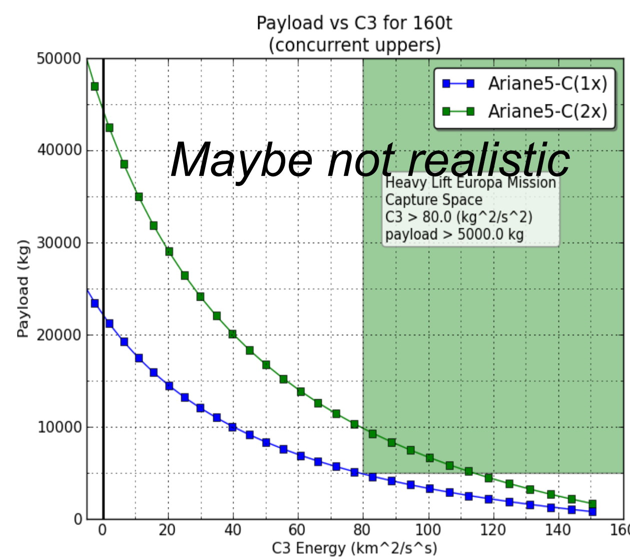






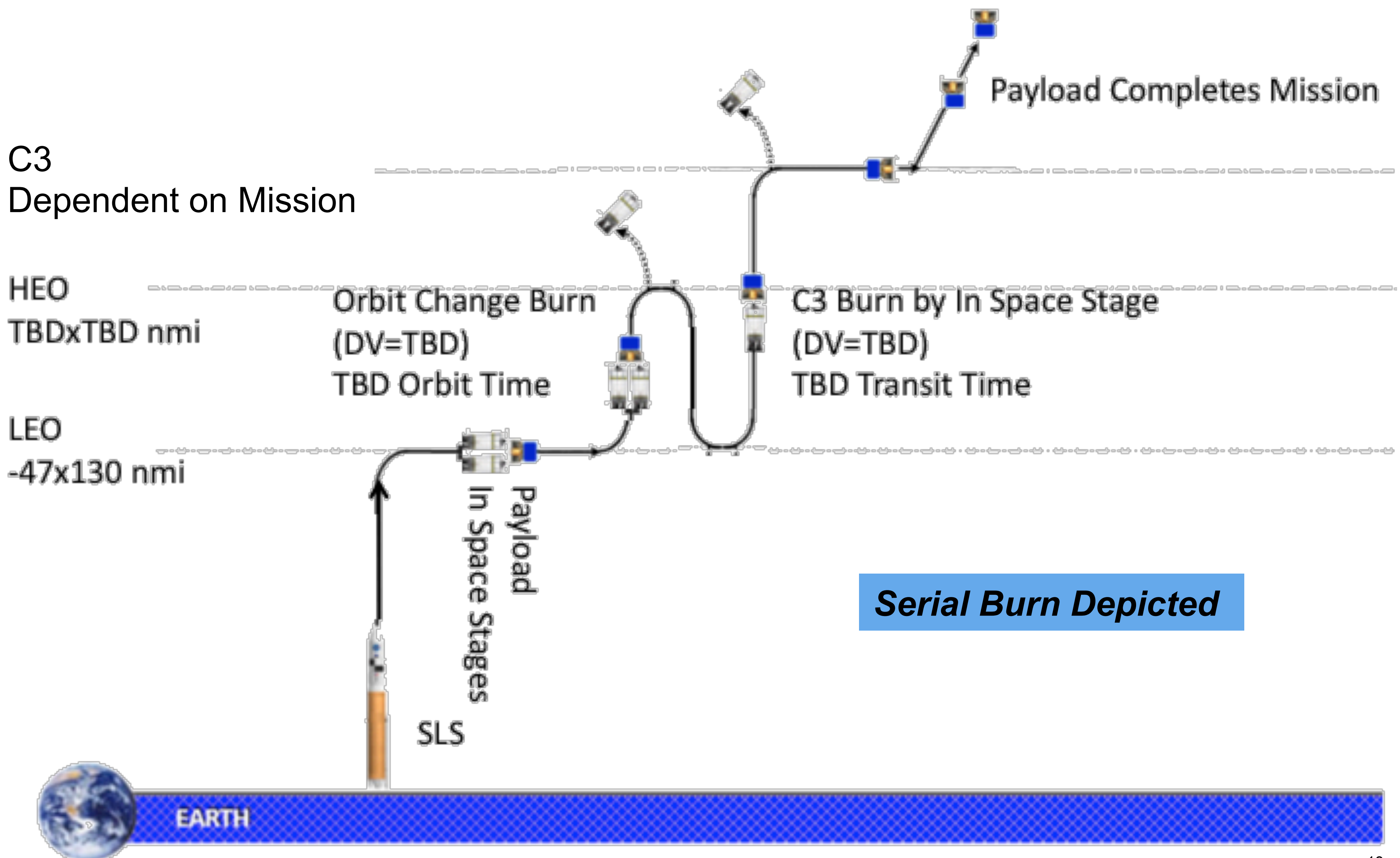






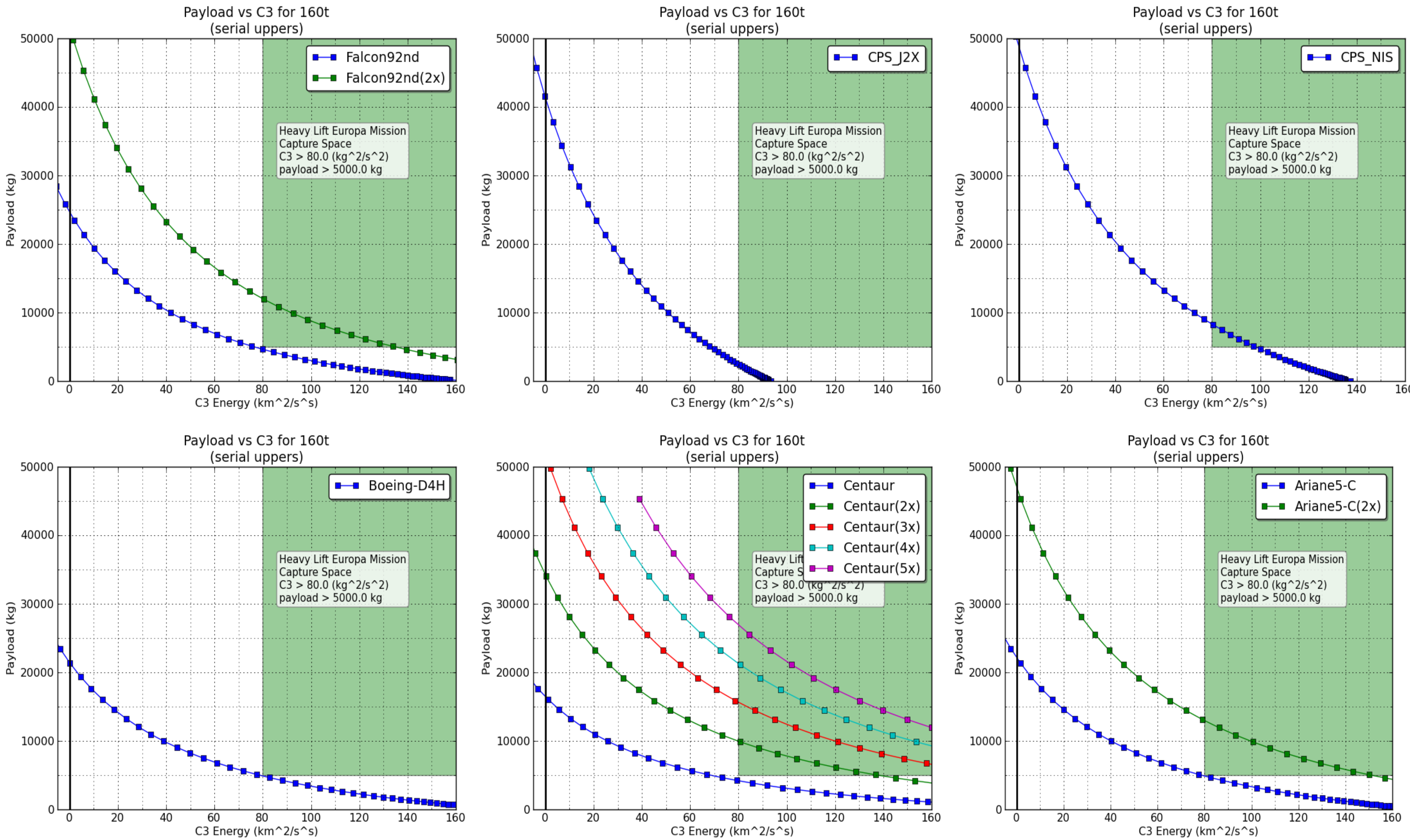
Concurrent means all stages burn together. Represents ~lower boundary.

LV Dropoff w/ Multiple Stages Burning to C3



SLS Dropoff = -47x130 nmi @ 29º

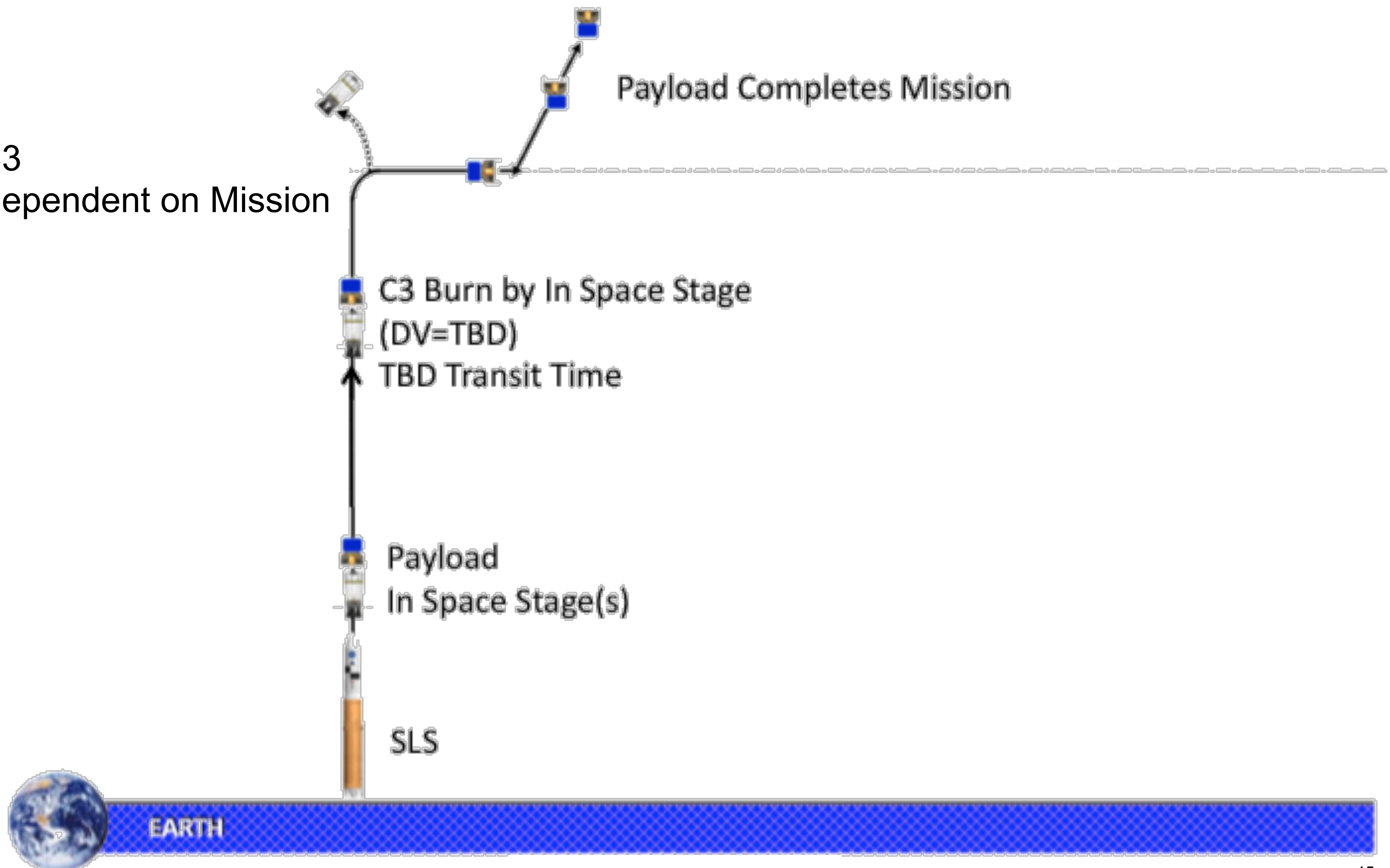
Multiple In-Space Stages: Serial Burn



Serial means a stage burns then is jettisoned, repeat, etc. Upper performance boundary.

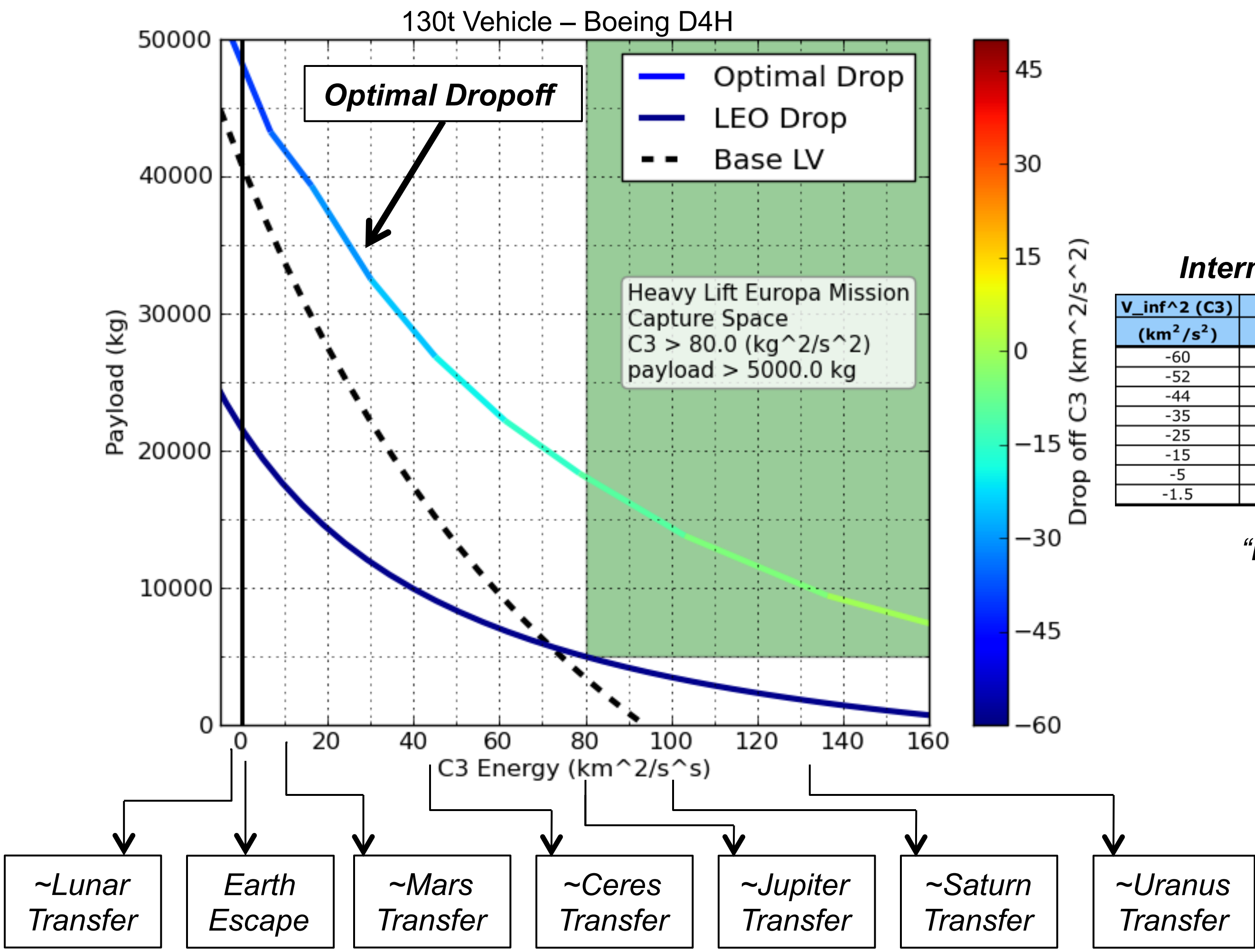
Integrated LV & In-Space Stage

C3
Dependent on Mission



Intermediate HLV (130t) + Delta IV-H Upper Stage (as in-space stage) Summary

Tracing out the highest mass as a function of the energy provided by the LV results in the following curve:



Intermediate Orbit Approximations

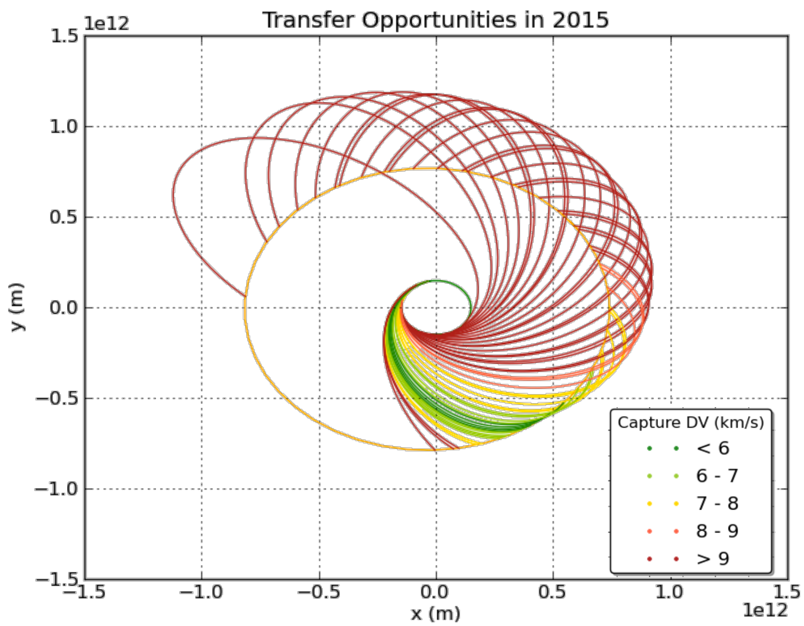
V_inf^2 (C3) (km^2/s^2)	Delta V (ft/s)	Approximate Orbit Apogee (130 x ?) nmi @ 29°	Notes
-60	0	-47 x 130 reference	Reference
-52	1,650	1,000	
-44	3,280	2,500	
-35	4,920	5,000	
-25	6,560	9,500	
-15	8,200	20,000	~GTO
-5	9,840	90,000	<1/2 Earth-Moon
-1.5	10,300	450,000	>2x Earth-Moon

"LEO Drop" assessed at -47x130nmi @ 29°

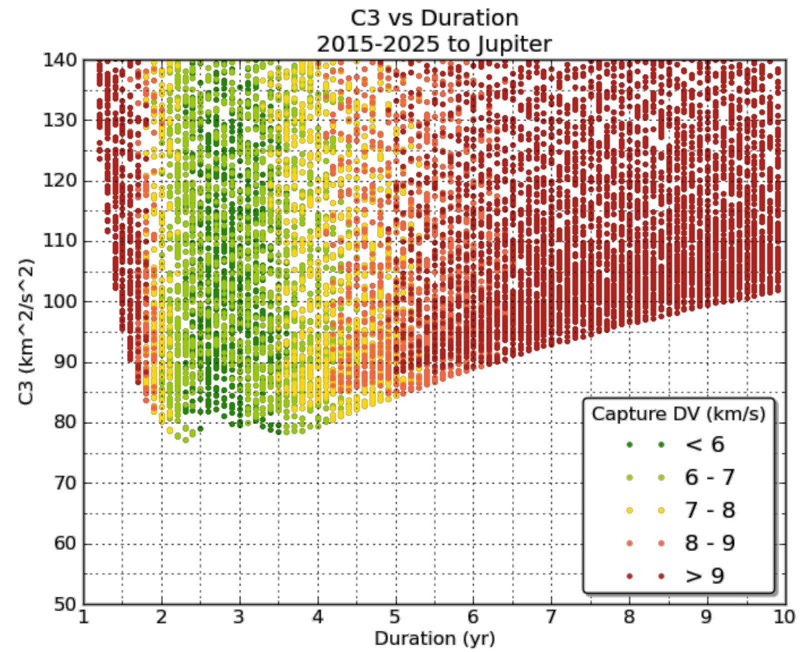
Performance depicted uses intermediate vehicle with a single in-space stage

Interplanetary Mission Design: Europa DRM

Transfer to Jupiter can be reduced from 2.75-3 years to ~2 years at a performance cost



Every year, various trajectory opportunities exist that have associated transfer durations and energy requirements. Depicted are the trajectories for 2015 from Earth to Jupiter with a direct transfer.



Repeating this yearly process over an extended period results in a general sensitivity to energy requirements vs. trip duration. The lowest energy opportunities (~77-82 km^2/s^2) take on the order of 2-4 years for outbound transit, but are very sparse. More launch opportunities exist if planning on up to 90 km^2/s^2.

Also, energy requirements begin to become unreasonable at or below 1.5 year outbound transit

Affordability Impacts

- **Recent assessment shows benefit of launch vehicle cost leverage**
  - Assessment shows using SLS rather than Atlas V can save up to $400M
- **Recent assessment shows benefit of trip time reduction**
  - Standard approach is a 6 year VEEGA trajectory on Atlas V w/ 2.5 year tour
  - Assessment shows up to $300M cost reduction by reducing outbound trajectory to ~3 years with a 1.5 year Jupiter tour
  - New assessment shows ability to reduce the outbound trajectory to ~2 years
  - Radiation mitigation strategy savings on the order of $300M as well
- **Unprecedented mass delivery to 80km^2/s^2:**
  - Simplified spacecraft design choice quantification
    - Eliminate some unique subsystem development by choosing heavier and/or heritage subsystems and/or components
  - Streamlined development cycle using add'l margin (DDT&E compress)
    - Non-mass driven design can lead to a cost & schedule driven DDT&E cycle
  - Multiple mission capability per launch vehicle
    - Quick assessment shows 5t JEO mission can be multiplied by at least 3 (w/ margin)
    - Spirit and Opportunity analog (yet on the same launch vehicle)

Total Mission Savings up to ~$1B plus savings realized from spacecraft simplification

General Study Observations

- **#1: Using SLS in an integrated fashion with a single, existing in-space stage delivers more performance to beyond LEO targets than most options assessed (variable SLS dropoff-orbit scenario)**

- **#2: All Evolved Vehicle concepts studied deliver comparable performance to most beyond-LEO targets**

- **#3: A heavy-lift vehicle can have a net positive cost impact on a particular beyond-LEO mission design based on trip duration reduction and subsystem design choices**