Cryogenic Propulsive Stage (CPS) Mission Sensitivity Studies
Low Earth Orbit Departure Results

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Mark Schaffer
Senior Aerospace Engineer, SpaceWorks Engineering
mark.schaffer@sei.aero | +1.202.503.1752

Brad St. Germain, Ph.D.
Director, Advanced Concepts Group, SpaceWorks Engineering
brad.stgermain@sei.aero | +1.770.379.8010

John E. Bradford, Ph.D.
Principal Engineer, SpaceWorks Engineering
john.bradford@sei.aero | +1.770.379.8007

- This presentation summarizes the results of a parametric study to characterize the influence of **four design parameters** on the design of Cryogenic Propulsion Stages (CPSs) for **four candidate missions** starting in Low Earth Orbit (LEO).

Design Parameters
Propellant Mass Fraction (PMF)
Engine Specific Impulse (Isp)
Boil-Off Rate
LEO Duration (loiter time)

Candidate Missions
Earth-Moon L1
Lunar Surface
Near Earth Object (NEO)
Mars Orbit

- These missions are representative of **future human exploration missions** in the 2020 to 2030 timeframe. Masses for the stage payloads (i.e. MPCV and habitation elements) for this study were taken from ongoing NASA studies.

- The ranges of values selected for the design parameters span currently available conservative values to realistic achievable, near-term technology advancement goals

- The primary **figure of merit for this study is launch mass to LEO**. Other systems-level design factors, such as cost or reliability, were not considered in this study

Introduction: Study Overview

Study Background

- 6 month study from June through December 2011
- Sponsored by **United Launch Alliance**
- Performed by **SpaceWorks** with technical support from **United Launch Alliance**

About SpaceWorks

- **Aerospace engineering services** and **space systems analysis** firm founded in 2000
  - A responsive and nimble **multidisciplinary engineering team** focused on independent concept analysis and design, technology assessment, and life cycle analysis at fidelity levels suitable for concept initiation through PDR
  - **Over a decade of experience** supporting advanced design and long range planning activities for customers in private industry, NASA, DoD, DARPA, and entrepreneurial space organizations
- Three primary operating divisions: **Engineering**, **Commercial**, and **Software**.
- Two partner companies: **Generation Orbit Launch Services, Inc.** and **Terminal Velocity Aerospace, LLC.**

Introduction: Background

TASK 1.1: PAYLOAD MASS VERIFICATION

TASK 1.2: DELTA-V REQUIREMENTS

- Mission 1: Earth-Moon L1 Lagrange Point
- Mission 2: Low Lunar Orbit (LLO)
- Mission 3: Near Earth Object (NEO) Encounter
- Mission 4: Mars Orbit

TASK 2.0 Mission Scenario Analysis Tool (MSAT) Modeling

- Mission 1: Earth-Moon L1 Lagrange Point
- Mission 2: Low Lunar Orbit (LLO)
- Mission 3: Near Earth Object (NEO) Encounter
- Mission 4: Mars Orbit

OBSERVATIONS AND CONCLUSIONS

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TASK 1.1: PAYLOAD MASS VERIFICATION

- **SpaceWorks was tasked with investigating and verifying the appropriate masses for the following Cryogenic Propulsive Stage (CPS) payloads:**
  - Orion/Multi-Purpose Crew Vehicle (MPCV)
  - Lunar Lander (Altair)
  - Long Duration Habitat (for deep space missions)
  - Gateway Habitat (for Earth-Moon L1 or L2)

- **SpaceWorks conducted a literature search of publicly available papers, presentations, and reports to investigate these masses and find reliable sources of data.**
  - In addition to masses, SpaceWorks recorded other relevant data about these payloads, including:
    - Crew size and mission duration assumptions
    - Technology level assumptions
    - Dimensions and volumes
    - Weight breakdown statements
  - As a part of the Task 1.0 deliverable, SpaceWorks will provide the contractor with all of the collected data and reference documents where applicable

Payload Mass: Overview

Source	Date	Crew Module Mass (kg)	Service Module Mass (kg)	Total Mass in LEO (kg)
Space.com Article[1]	Apr 2011	-	-	**21,250**
Iowa State Univ. NEO Paper[2]	Oct 2009	-	-	**20,500**
NASAfacts Orion Datasheet[3]	Jan 2009	8,913	12,337	**21,250**
NASA / White Sands Report[4]	Aug 2007	-	-	**23,395**
NASAfacts Orion Datasheet[5]	Aug 2006	-	-	**23,395**
NASA Project Orion Overview[6]	Aug 2006	8,485	-	**-**
CEV - ESAS Final Report[7]	Nov 2005	9,506	13,647	**23,153**

- Original estimate is a little low compared to published data
- SpaceWorks recommends 21,250 kg based on recent NASA datasheet

Original Estimate = 20,000 kg

SpaceWorks Recommendation = 21,250 kg

Image Source: http://www.lockheedmartin.com/products/Orion/

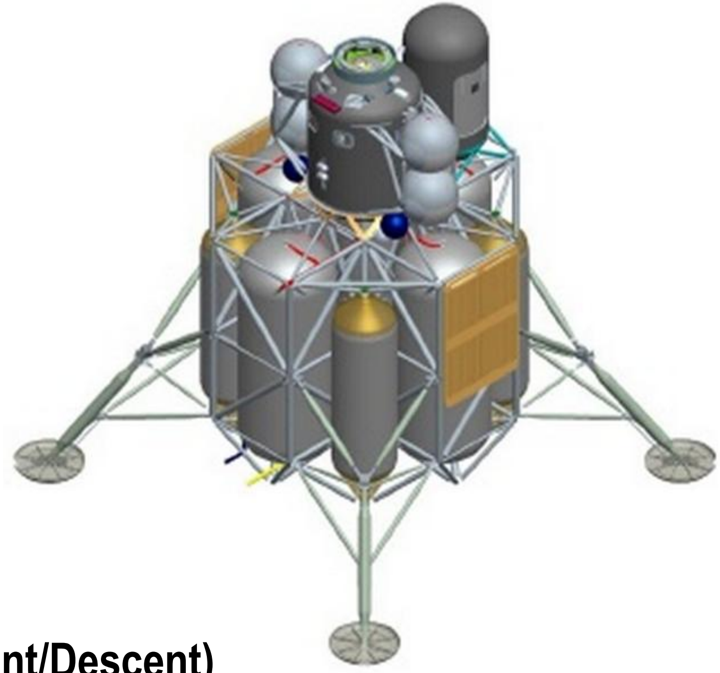
Payload Mass: Orion / MPCV (for Deep Space Missions)

Source	Date	Ascent Stage Mass (kg)	Descent Stage Mass (kg)	Total Mass in LEO (kg)
Iowa State Univ. NEO Paper[2]	Oct 2009	-	-	**43,000**
Andrews LDAC Presentation[8]	Jan 2009	5,331	33,887	**45,000****
NASA Altair Presentation[9]	Oct 2008	-	-	**45,000**
NASAfacts Altair Datasheet[10]	Sept 2008	6,141	37,045	**43,186**
NASA Altair Presentation[11]	Apr 2008	6,128	38,002	**45,586***
ESAS LSAM (Final Config)[7]	Nov 2005	10,809	35,055	**45,864**

* Includes separate airlock mass and payload

** Includes reserves and payload

Image Source: http://www.scribd.com/doc/18824958/Introduction-to-the-Altair-Lunar-Lander-Project#



- Original estimate of 30,000 kg is very low compared to published data . However, all published versions of the Altair lander referenced above perform the LOI maneuver. In this study, the CPS will perform LOI.
- SpaceWorks investigated further to look at lander configurations that do not perform LOI

Payload Mass: Lunar Lander (Combined LOI and Ascent/Descent)

Source	Date	Ascent Stage Mass (kg)	Descent Stage Mass (kg)	Total Mass in LEO (kg)
ESAS LSAM (analysis cycle 1)[7]	Nov 2005	9,898	18,010	**27,908**
ESAS LSAM (final w/o LOI propellant*)[7]	Nov 2005	10,809	21,125	**31,933**
DPT L1 Lunar Lander[37] **	Nov 2000	-	-	**29,656**
Apollo 14 Lunar Lander[38]	Jan 1971	4,943	10,334	**15,277**

* SpaceWorks estimates 13,930 kg of propellant required for LOI for this system
** L1 Lunar Lander travels between Earth-Moon L1 and lunar surface, rather than LLO and lunar surface

- The ESAS lander from analysis cycle 1 represents a closed design performing only descent and ascent from LLO
- Based on this design and the other collected data points, SpaceWorks recommends retaining the original estimate of 30,000 kg for the lunar lander

Original Estimate = 30,000 kg
SpaceWorks Recommendation = 30,000 kg

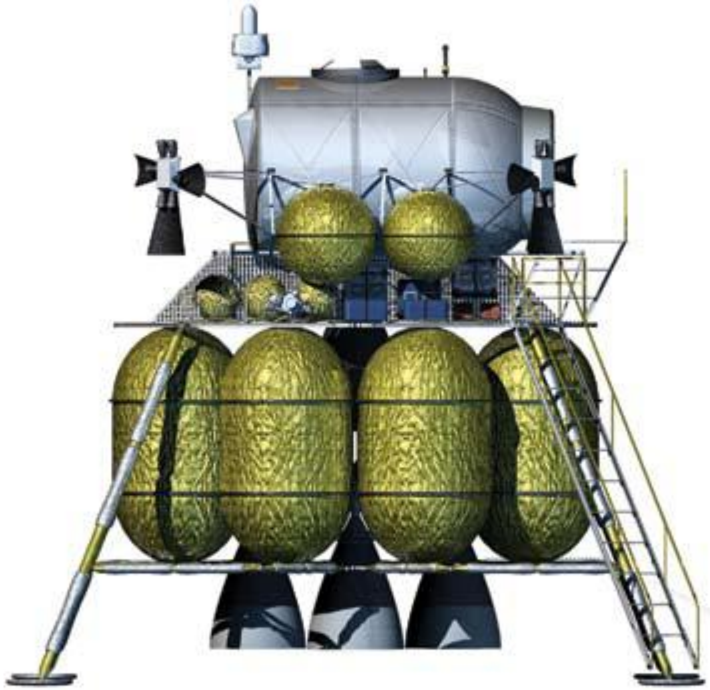


Image Source: NASA's Exploration Systems Architecture Study: Final Report [7]

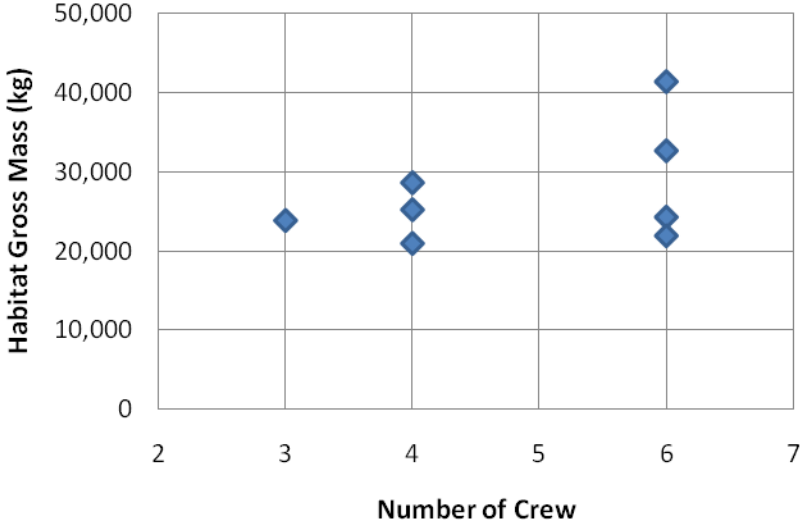
Payload Mass: Lunar Lander (Ascent/Descent Only)

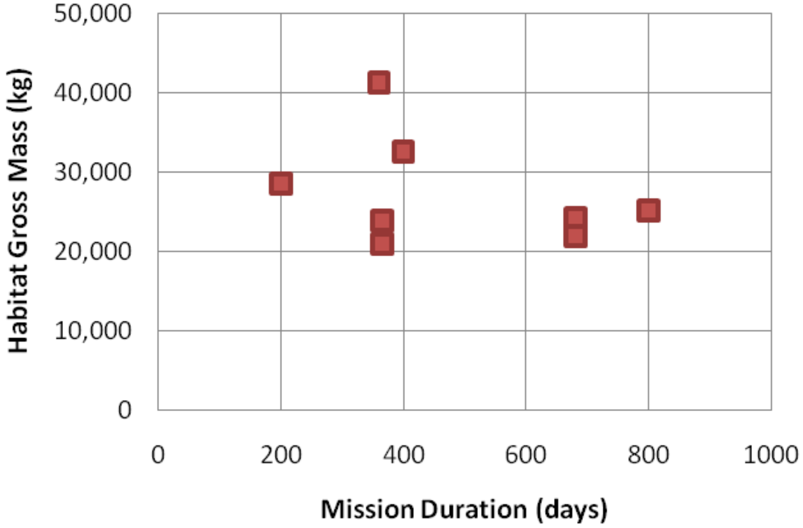
Source	Date	Number of Crew	Duration (days)	Habitat Inert Mass (kg)	Food Mass (kg)	Total Mass in LEO (kg)
NASA HEFT Phase 2 DSH[12]	Dec 2010	3	365	21,849	1,992	**23,841**
NASA DRM 5.0 Transit Habitat[13]	Jul 2009	6	360	28,100	13,240	**41,340**
DPT Mars Transit Habitat[14]	Jul 2007	6	400	20,437	12,192	**32,629**
DPT Short-Stay Habitat[14]	Jul 2007	4	365	15,748	5,200	**20,948**
ISS TransHab Concept[15]	Jun 2003	6	-	13,200	-	**13,200**
NASA DRM 3.0 Lander Habitat[16]	Jun 1998	6	680	-	-	**24,268**
NASA DRM 3.0 ERV Habitat[16]	Jun 1998	6	680	-	-	**21,915**
Mars Direct ERV[17]	Nov 1997	4	200	25,200	3,400	**28,600**
Mars Direct Hab[17]	Nov 1997	4	800	18,200	7,000	**25,200**

- **Long Duration Habitat masses vary significantly**
  - Different assumptions are made regarding crew size and mission duration
  - Different studies assume different levels of technology, particular in the area of open loop, partially closed, and fully closed life support systems
  - Different studies assume different capabilities of the habitat. In some Mars studies the in-space habitat also serves as the surface habitat and includes landing systems and structure.

Image Source: http://upload.wikimedia.org/wikipedia/commons/8/87/Transhab-cutaway.jpg

Payload Mass: Long Duration Habitat (1 of 2)





* This dataset excludes the ISS TransHab concept because the ISS TransHab leverages the equipment and supplies available on the station.

- **Some correlation between crew size and habitat mass**
  - Habitat volume scales directly with crew size, and mass scales directly with volume
- **Little correlation between mission duration and habitat mass**
  - For mission durations exceeding 100 days, habitat volume is typically insensitive to mission duration.
  - Food requirement scales directly with duration, but different studies make different assumptions on food requirements and hydrated vs. dehydrated foods.
- **SpaceWorks recommends keeping the original estimate of 30,000 kg. This mass represents a conservative 4 crew case or an average 6 crew case.**

Original Estimate = 30,000 kg

SpaceWorks Recommendation = 30,000 kg

Payload Mass: Long Duration Habitat (2 of 2)

Source	Date	Number of Crew	Total Mass in LEO (kg)
FISO EELV Option[18]	Dec 2010	3	**16,000***
FISO HLV Option[19]	Dec 2005	4	**30,500**
DPT Gateway Habitat[20]	Oct 2001	4	**22,800**

* Excludes1,500 kg of outfitted mass to be brought to Earth-Moon L1,2 point by separate launch system.

- **Gateway Habitat is currently being studied by the Future In Space Operations (FISO) working group**
- **FISO EELV Option represents latest design iteration of Gateway Habitat**
  - Launched in two parts to LEO using Delta IV H
  - Assembled in LEO and launched to Earth-Moon L1,2 using Delta IV Upper Stage
- **SpaceWorks recommends using the original estimate of 16,000 kg to be consistent with the latest FISO design iteration**

Original Estimate = 16,000 kg

SpaceWorks Recommendation = 16,000 kg

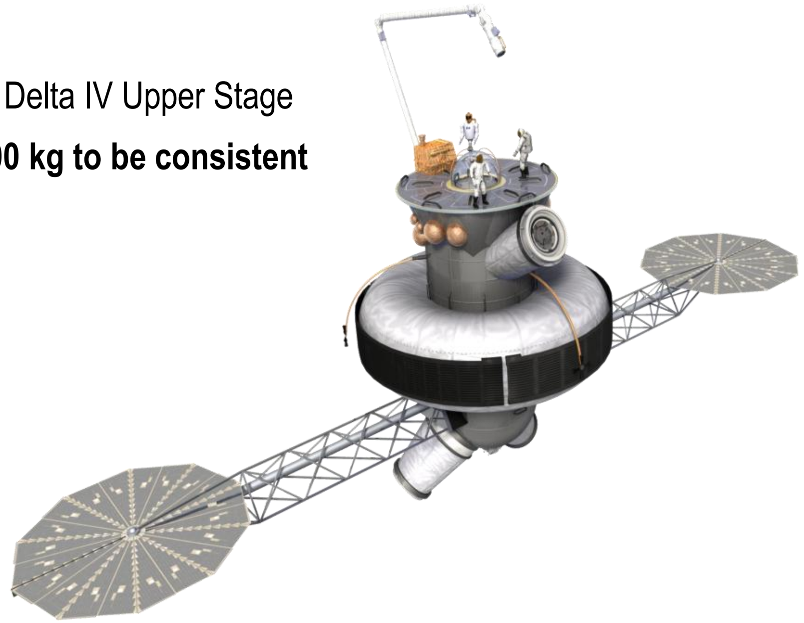
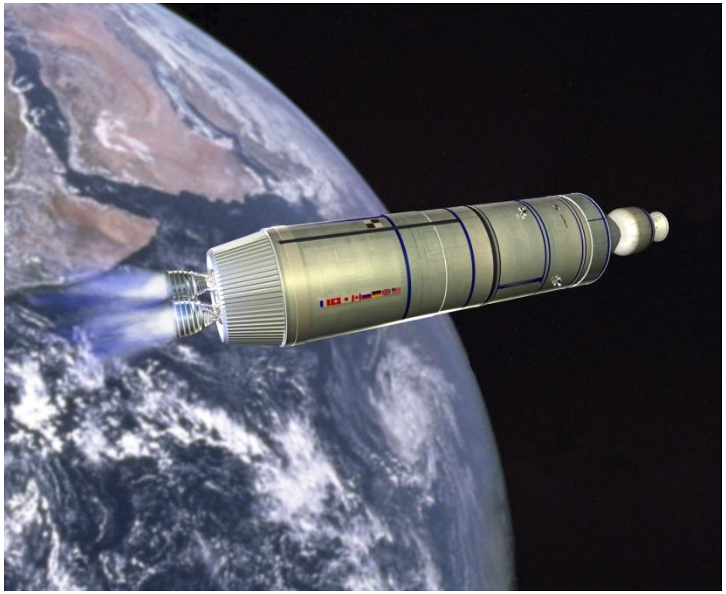


Image Source: http://www.futureinspaceoperations.com/papers/ISU_ISS_Pres2011v3.pdf

Payload Mass: Gateway Habitat

- **SpaceWorks recommends using the following masses for the four CPS payloads:**

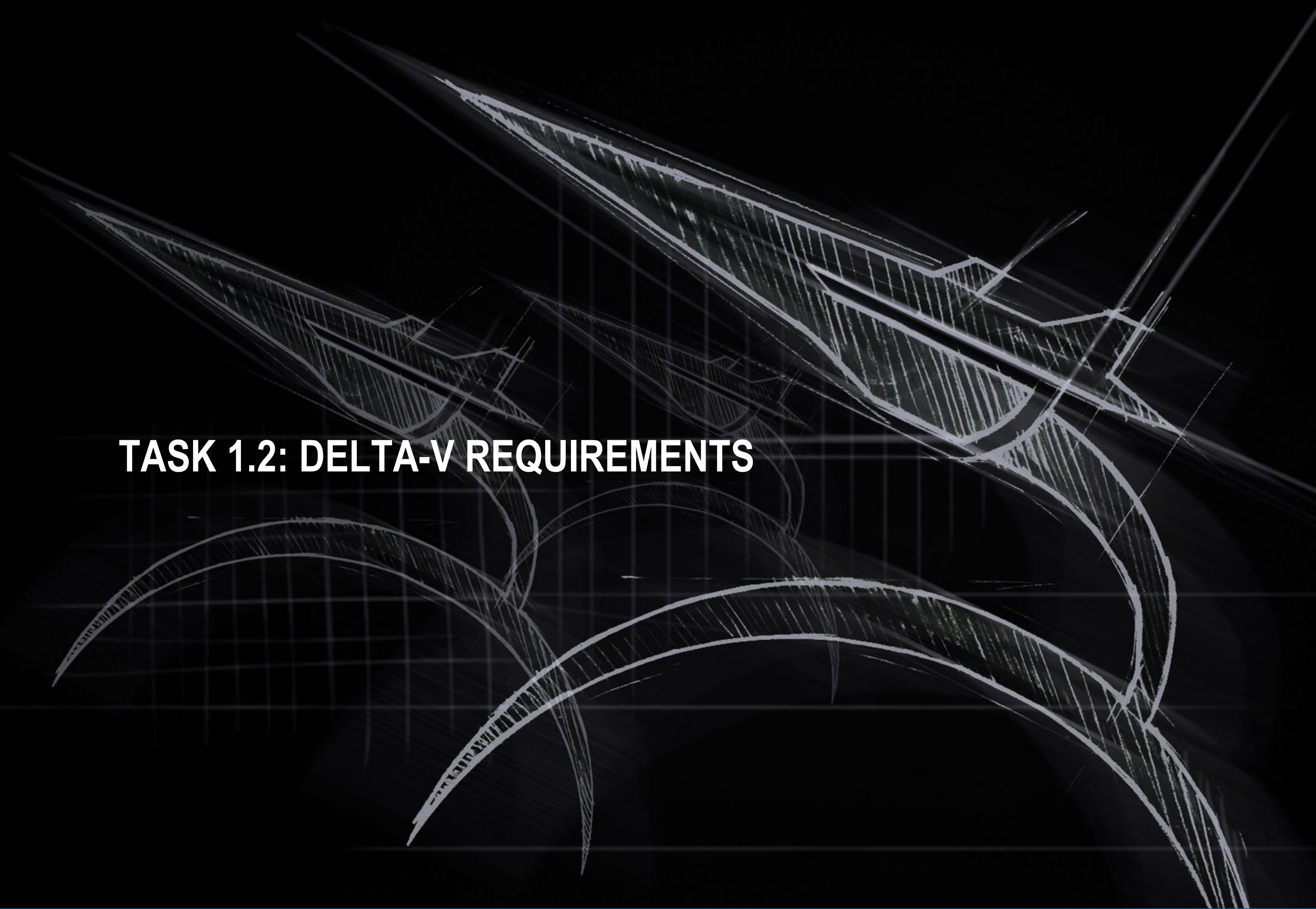
Payload	Original Mass Estimate (kg)	SpaceWorks Recommendation (kg)
Orion / MPCV	20,000	**21,250**
Lunar Lander	30,000	**30,000**
Long Duration Habitat	30,000	**30,000**
Gateway Habitat	16,000	**16,000**



Payload Mass: Summary

Image Credit: SpaceWorks Enterprises, Inc.

TASK 1.2: DELTA-V REQUIREMENTS



- **SpaceWorks was tasked with determining the required mission delta-Vs for the CPSs beginning in LEO for each of the following reference missions:**
  - Earth-Moon L1 Lagrange Point
  - Low Lunar Orbit (LLO)
  - Near Earth Object (NEO) Encounter
  - Mars Orbit Mission

- **SpaceWorks conducted a literature search of publicly available papers, presentations, and reports to investigate these delta-Vs and find examples of these reference missions**
  - As a part of the Task 1.0 deliverable, SpaceWorks will provide the contractor with all of the collected data along with all of the source documents for the data when applicable

- **SpaceWorks is currently modeling each reference mission with internal trajectory tools to determine the final delta-Vs to be used in the next phase of the study.**
  - The delta-Vs calculated will be cross-referenced with published data
  - As a part of the Task 1.0 deliverable, SpaceWorks will provide the full set of results from the trajectory modeling

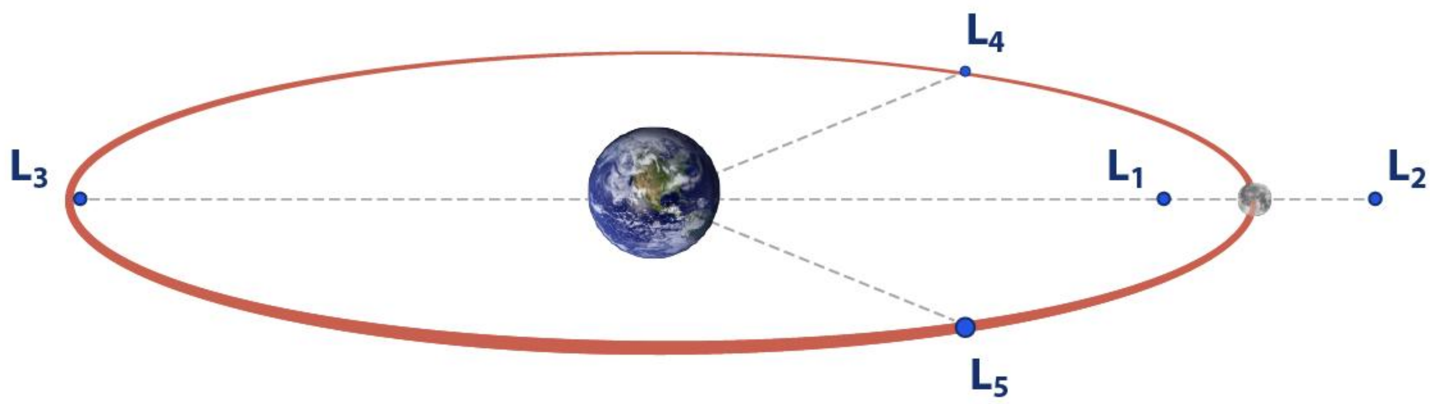
Delta-V Requirements: Overview

Mission 1: Earth-Moon L1 Lagrange Point

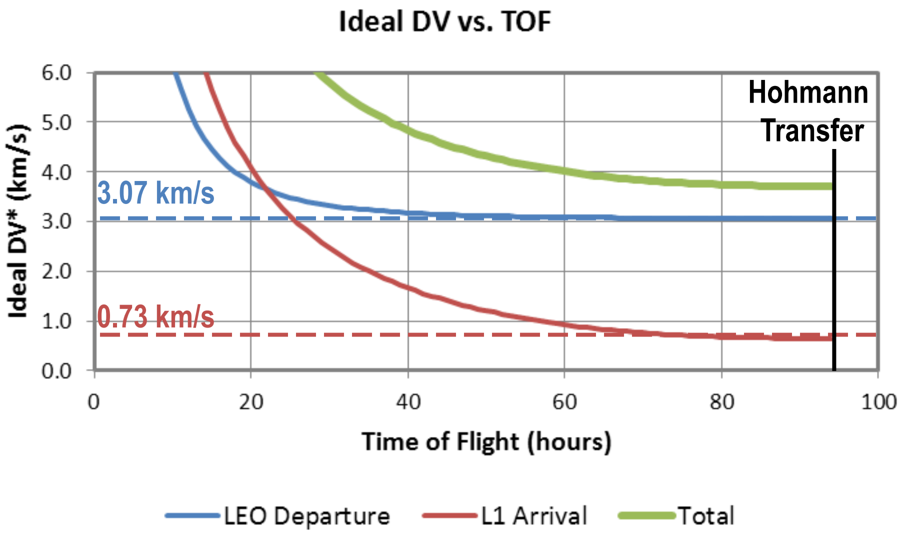
- SpaceWorks has collected delta-Vs from other mission examples for reference:

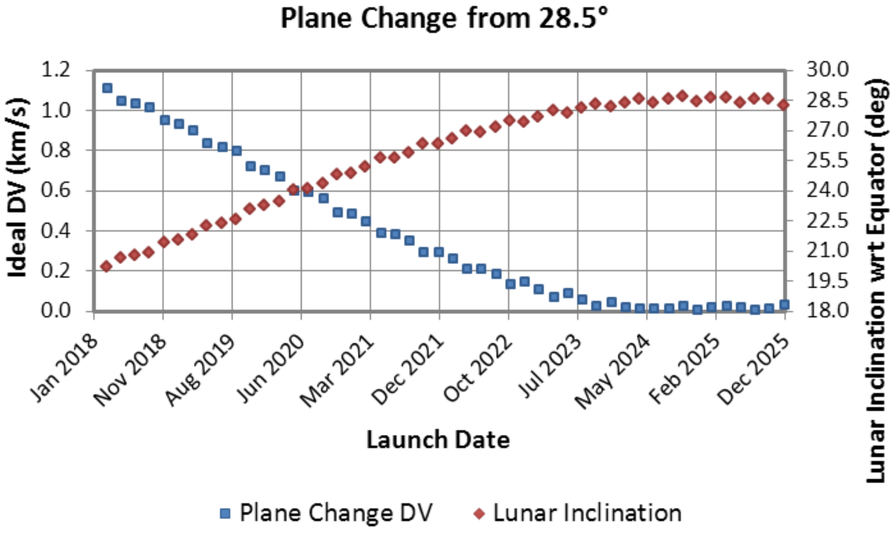
Source	Date	LEO Departure ΔV (m/s)	L1 Arrival ΔV (m/s)	Total ΔV (m/s)
FISO Working Group[18]	Dec 2010	3,074	630	**3,700**
FISO Working Group* [19]	Sept 2009	-	-	**3,800**
Princeton – Chow/Gralla[21]	May 2004	2,800**	600	**3,400**
UMD - Clarke Station[22]	May 2001	3,100	700	**3,800**
NASA White Paper[23]	1993	-	-	**3,770**

* Original source listed as NASA Decade Planning Team (2000)
** Calculated from C3 requirement



Delta-V Requirements: Earth-Moon L1 – Background





* Assume insertion into a circular geocentric orbit at L1 distance and lunar rotational velocity

- SpaceWorks used an internal Earth-Moon patched conic trajectory tool to determine ideal required LEO departure and L1 arrival delta-Vs as a function of Time of Flight (TOF) for Earth-Moon L1 missions
- Because the Moon's orbit inclination varies with respect to the Earth, a plane change maneuver may be required if launching from CCAFS (inclination = 28.5°) before performing the LEO departure maneuver

- **SpaceWorks recommends the following delta-Vs for LEO Departure and L1 Arrival based on this analysis. These represent the delta-V required for a minimum 72 hour time of flight from LEO to L1.**

LEO Departure ΔV (m/s)	L1 Arrival ΔV (m/s)	Total ΔV (m/s)
3,070	730	3,800

Delta-V Requirements: Earth-Moon L1 – Analysis

Mission 2: Low Lunar Orbit

- SpaceWorks has collected delta-Vs from other mission examples for reference, including Trans-Lunar Injection (TLI) and Lunar Orbit Insertion (LOI) values when available:

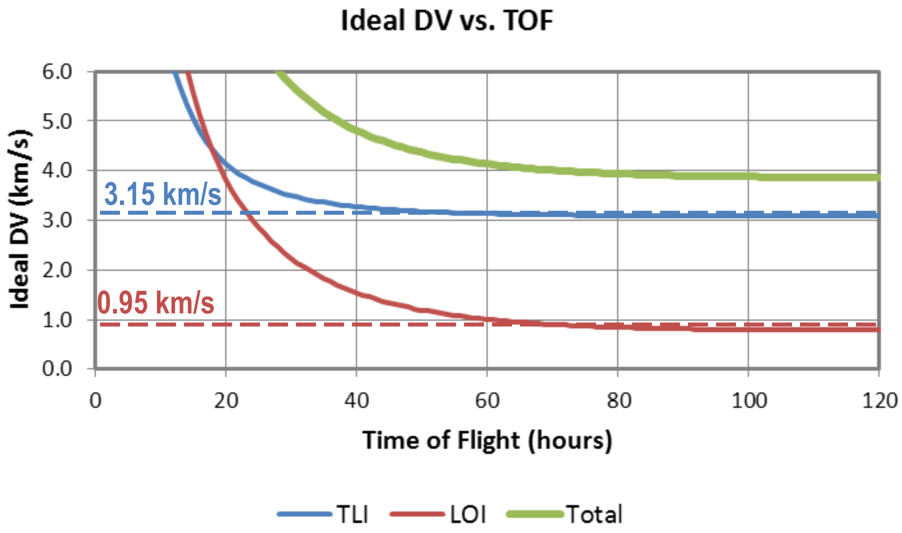
Source	Date	TLI ΔV (m/s)	LOI ΔV (m/s)	Total ΔV (m/s)
FISO Working Group[18]	Dec 2010	-	-	**4,040**
FISO Working Group* [19]	Sept 2009	-	-	**4,000**
NASA CEV SM Overview[24]	May 2007	3,120	795	**3,915**
ESAS Final Report[7]	Nov 2005	3,150	835	**3,985**
Farquhar Journal Article[25]	June 1972	3,139	914	**4,053**

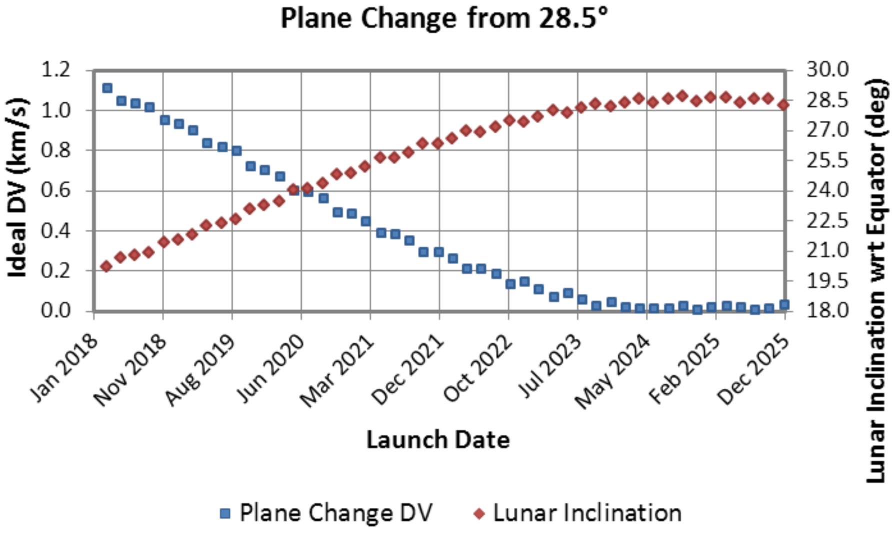
* Original source listed as NASA Decade Planning Team (2000)

- SpaceWorks has also collected the delta-Vs from the Apollo missions:

Source	Date	TLI ΔV (m/s)	LOI ΔV (m/s)	Total ΔV (m/s)
Apollo 11 Mission Report[30]	Jul 16, 1969	3,183	944	**4,127**
Apollo 12 Mission Report[31]	Nov 14, 1969	3,205	950	**4,155**
Apollo 14 Mission Report[32]	Jan 31, 1971	3,160	967	**4,127**
Apollo 15 Mission Report[33]	Jul 26, 1971	3,187	939	**4,126**
Apollo 16 Mission Report[34]	Apr 16, 1972	3,167	883	**4,049**
Apollo 17 Mission Report[35]	Dec 7, 1972	3,163	935	**4,098**

Delta-V Requirements: Low Lunar Orbit – Background





- SpaceWorks used an internal Earth-Moon patched conic trajectory tool to determine ideal required TLI and LOI delta-Vs as a function of Time of Flight (TOF) for Low Lunar Orbit (LLO) missions
- Because the Moon's orbit inclination varies with respect to the Earth, a plane change maneuver may be required if launching from CCAFS (inclination = 28.5) before performing TLI

- **SpaceWorks recommends the following delta-Vs for TLI and LLO based on this analysis. These represent the delta-V required for a minimum 72 hour time of flight from LEO to LLO.**

TLI ΔV (m/s)	LOI ΔV (m/s)	Total ΔV (m/s)
3,150	950	4,100

Delta-V Requirements: Low Lunar Orbit – Analysis

Mission 3: Near Earth Object (NEO) Encounter

- SpaceWorks identified 27 candidate NEOs for human missions that have been considered in previous studies[2,26, 27,28]

- SpaceWorks narrowed down this list based on launch date and total required delta-V to 7 candidates for further study

Candidate NEO	Launch Date	Earth Departure ΔV (m/s)	Post Escape ΔV (m/s)	Total ΔV (m/s)	Mission Duration (days)
2006 FH36	2/16/2016	3,849	4,165	8,014	123
2008 HU4	4/5/2016	3,280	1,980	5,260	180
2004 JN1	11/26/2016	3,232	5,260	8,492	79
1991 VG	7/21/2017	3,360	2,290	5,650	180
2001 CQ36	5/6/2018	3,244	6,213	9,457	115
2001 GP2	11/6/2019	3,360	1,570	4,930	365
2008 EA9	11/28/2019	3,420	2,120	5,540	180
1998 KY26	5/21/2020	3,362	4,164	7,526	54
2007 UN12	5/22/2020	3,300	1,450	4,750	365
2001 QJ142	4/24/2024	3,490	3,400	6,890	180
1999 AO10	9/19/2025	3,320	3,740	7,060	150
2003 LN6	12/21/2025	3,330	3,690	7,020	180
2000 SG344	4/27/2028	3,340	3,220	6,560	180
2006 UQ216	8/15/2028	3,710	3,550	7,260	180
Apophis	4/13/2029	3,448	6,601	10,049	201
2006 DQ14	8/27/2030	3,770	2,100	5,870	180
1999 CG9	8/18/2033	3,530	3,080	6,610	180
2000 LG6	1/2/2036	3,270	3,210	6,480	180
2001 FR85	9/24/2039	3,610	1,790	5,400	180
2005 LC	12/13/2039	3,310	3,310	6,620	180
1999 VX25	6/12/2040	3,360	3,870	7,230	150
1997 YM9	12/27/2044	4,060	2,360	6,420	180
2006 UB17	4/22/2045	3,290	3,450	6,740	180
2006 WB	6/1/2050	3,450	3,250	6,700	180
2006 QQ56	3/7/2051	3,450	2,470	5,920	180
2006 HC	4/20/2054	3,950	3,190	7,140	180
1992 JD	11/4/2054	5,320	1,650	6,970	180

Delta-V Requirements: NEO Encounter – Introduction

- SpaceWorks narrowed the original list of 27 NEOs down to 7 NEOs with launch windows in the 2020-2030 timeframe for use as representative NEOs:

Candidate NEO	Launch Date	Earth Departure ΔV (m/s)	Post Escape* ΔV (m/s)	Total ΔV (m/s)	Time of Flight (days)	Semi-major Axis (AU)	Eccentricity	Inclination to Ecliptic (deg)
1998 KY26	5/21/2020	3,362	4,164	7,526	54	1.252	0.212	1.267
2007 UN12	5/22/2020	3,300	1,450	4,750	365	1.054	0.063	0.238
2001 QJ142	4/24/2024	3,490	3,400	6,890	180	1.062	0.087	3.104
1999 AO10	9/19/2025	3,320	3,740	7,060	150	0.911	0.113	2.624
2003 LN6	12/21/2025	3,330	3,690	7,020	180	0.855	0.213	0.673
2000 SG344	4/27/2028	3,340	3,220	6,560	180	0.983	0.065	0.108
2006 UQ216	8/15/2028	3,710	3,550	7,260	180	1.112	0.166	0.483

* Post escape delta-V includes NEO arrival, near-NEO operations, and NEO departure maneuvers

- SpaceWorks collected orbital parameters and ephemeris data for these 7 NEOs from the JPL HORIZONS[36] online database to analyze missions to these asteroids
  - Mass and diameter information is unavailable for these NEOs currently
  - Breakouts of post escape delta-Vs were not available from literature for the selected NEOs

Delta-V Requirements: NEO Encounter – NEO Down-Select

- SpaceWorks used Bullseye, their commercially available interplanetary trajectory tool, to independently check the delta-Vs found in literature based on the latest ephemeris data
  - This ephemeris used in Bullseye to generate these results was the latest available from JPL and may be different than the ephemeris used in the literature references

Candidate NEO	Launch Date	ΔV From Literature (m/s)			SpaceWorks Calculated ΔV (m/s)				
		Earth Departure	Post Escape	Total	Earth Departure	NEO Arrival	NEO Departure	Post Escape	Total
1998 KY26*	5/21/2020	3,362	4,164	7,526	-	-	-	-	-
2007 UN12	5/22/2020	3,300	1,450	4,750	3,250	390	560	950	4,200
2001 QJ142	4/24/2024	3,490	3,400	6,890	3,350	1,220	1,990	3,210	6,560
1999 AO10	9/19/2025	3,320	3,740	7,060	3,230	2,020	2,110	4,130	7,360
2003 LN6	12/21/2025	3,330	3,690	7,020	3,230	2,230	2,210	4,440	7,670
2000 SG344	4/27/2028	3,340	3,220	6,560	3,300	960	2,150	3,110	6,410
2006 UQ216**	8/15/2028	3,710	3,550	7,260	3,350	1,800	1,790	3,590	6,940

* SpaceWorks unable to find feasible mission around reference launch date
** SpaceWorks unable to reasonable match reference value for 180 mission. Delta-Vs shown are for 360 day mission.

- SpaceWorks recommends the following delta-Vs based on this analysis:

Earth Departure ΔV (m/s)	Post Escape ΔV (m/s)	Total ΔV (m/s)
3,350	**4,150**	**7,500**

Delta-V Requirements: NEO Encounter – Analysis

Mission 4: Mars Orbit

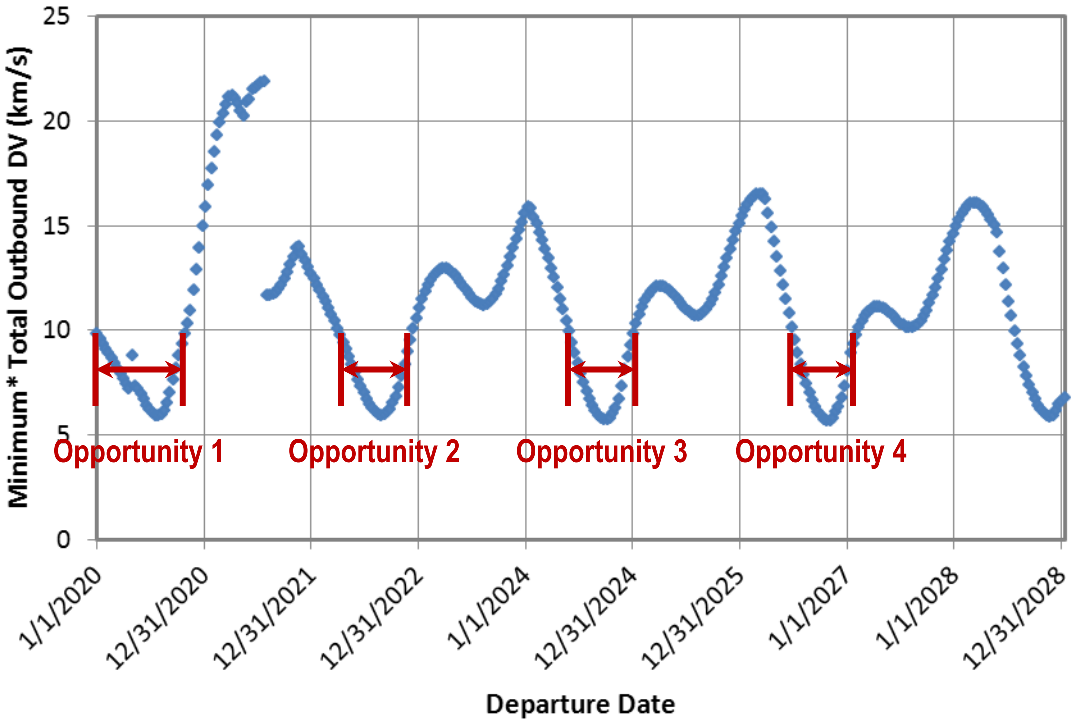
- SpaceWorks has collected delta-Vs from other Mars mission examples for reference:

Source	Date	Mission Type	TMI ΔV (km/s)	MOI ΔV (km/s)	TEI ΔV (km/s)	Total ΔV (km/s)	Mission Duration (months)
Mars Rapid Round Trip[29]	Sept 2010	Opposition	7.9	13.0	7.5	**28.4**	**14**
NASA DRM 5.0[13]	Jul 2009	Conjunction	3.7-4.1	1.0-1.8	1.5-1.6	**6.2-7.5**	**30**
DPT Mars Study[14]	Jul 2007	Short Duration	-	-	-	**21.7-31.2**	**12**
DPT Mars Study[14]	Jul 2007	Opposition	-	-	-	**14.8-25.8**	**15-22**
DPT Mars Study[14]	Jul 2007	Conjunction	-	-	-	**5.6-6.7**	**30-32**

- Mars missions are typically classified as either short duration (1 year or less), opposition, or conjunction class missions
- SpaceWorks recommends focusing on conjunction class missions for this study. The long mission duration for these missions is offset by significantly reduced delta-V requirements.

Delta-V Requirements: Mars Orbit Mission – Background

- SpaceWorks used Bullseye, their commercially available interplanetary trajectory tool, to perform a sparse sweep of conjunction class Earth-to-Mars trajectories in the 2020-2030 timeframe and identify 4 mission opportunities for drill-down investigation.

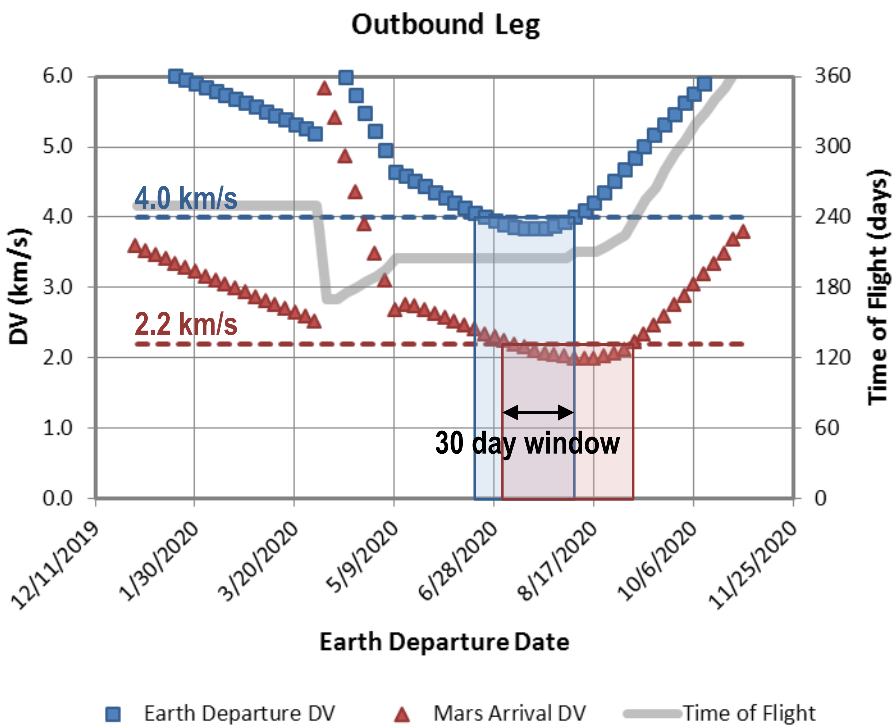


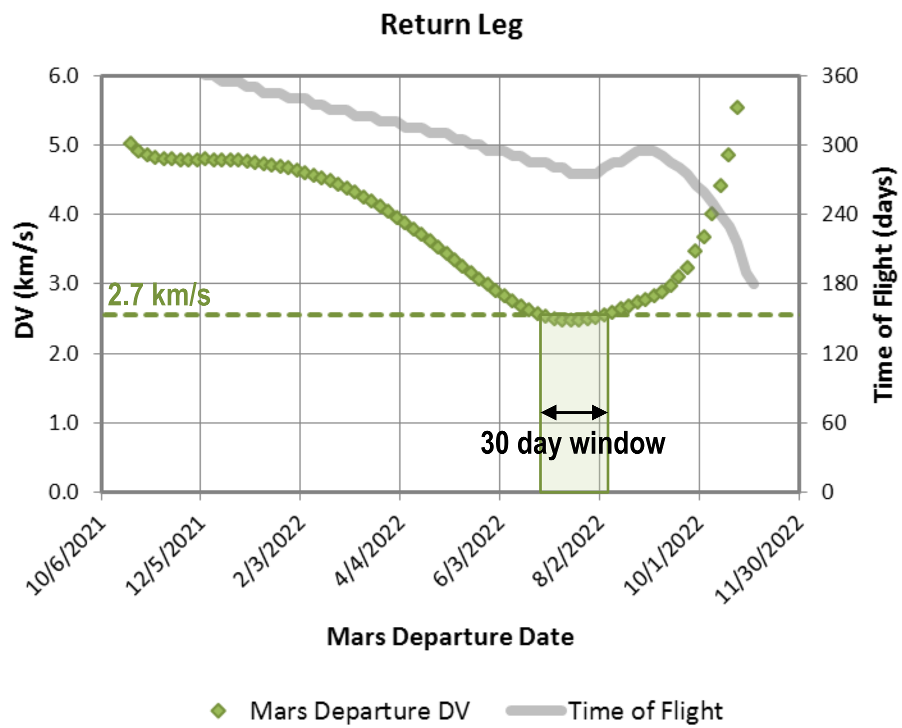
Opportunity	Start Date	End Date
1	1/1/2020	11/6/2020
2	3/11/2022	12/16/2022
3	5/19/2024	1/14/2025
4	6/28/2026	2/13/2027

* Subject to maximum time-of-flight constraint of 365 days

- Within these opportunities, SpaceWorks then determined appropriate launch windows and performed a detailed analysis of each window using Bullseye

Delta-V Requirements: Mars Orbit Mission – Opportunities



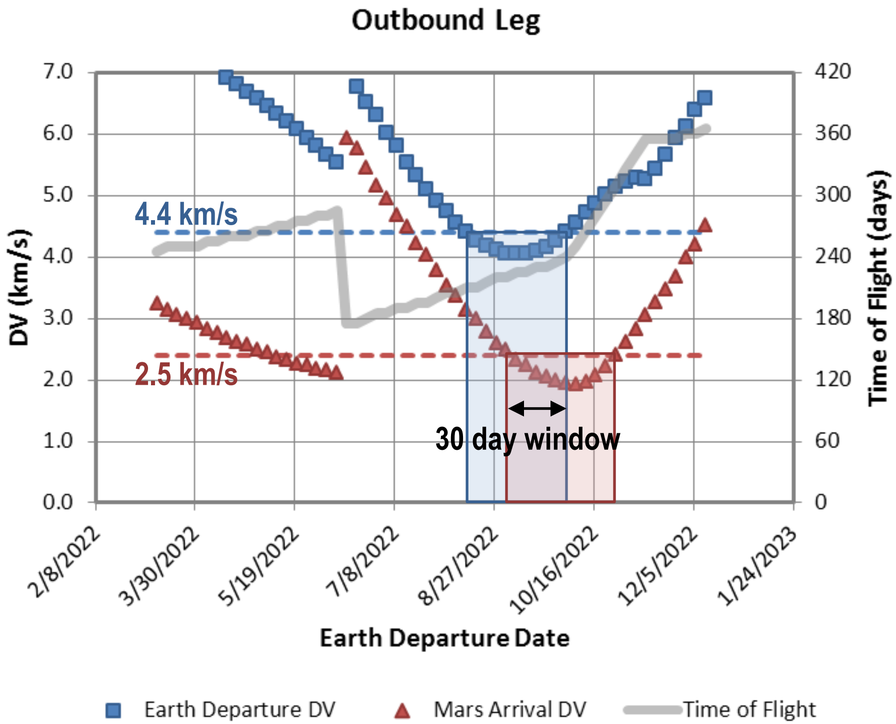


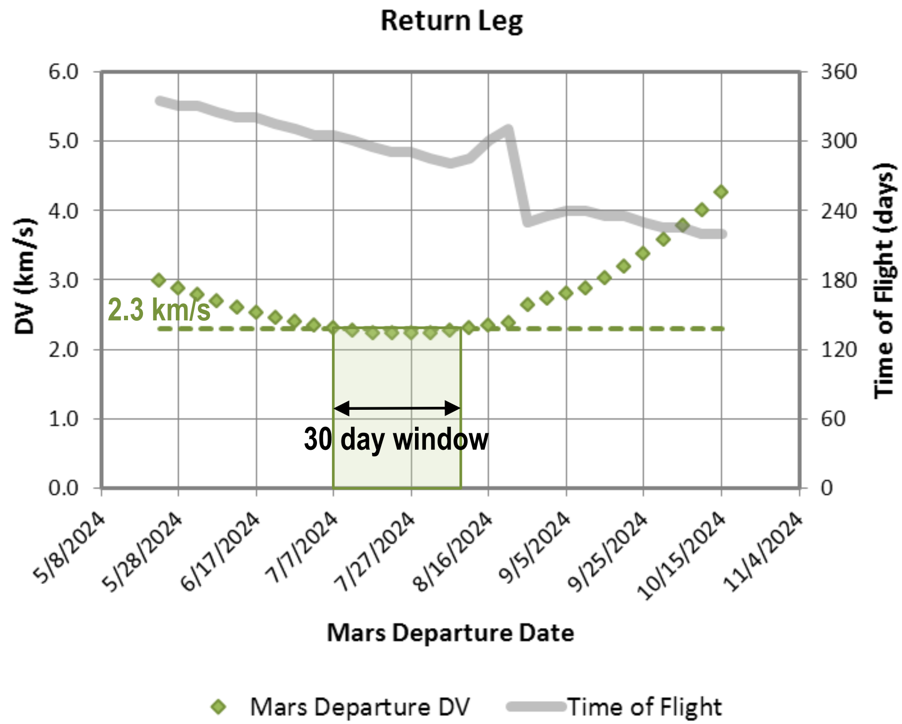
- Delta-Vs required to achieve 30 day launch windows for outbound and return legs:

Maneuver	ΔV or Velocity	Opening Date	Closing Date
Earth Departure (ΔV)	4,000 m/s	7/9/2020	8/8/2020
Mars Arrival (ΔV)	2,200 m/s	1/30/2021	3/6/2021
Mars Departure (ΔV)	2,550 m/s	7/1/2022	8/5/2022
Earth Entry (maximum velocity)	12.2 km/s	4/12/2023	5/12/2023

- Surface stay duration is 480 days to 550 days

Delta-V Requirements: Mars Orbit Mission – Opportunity 1



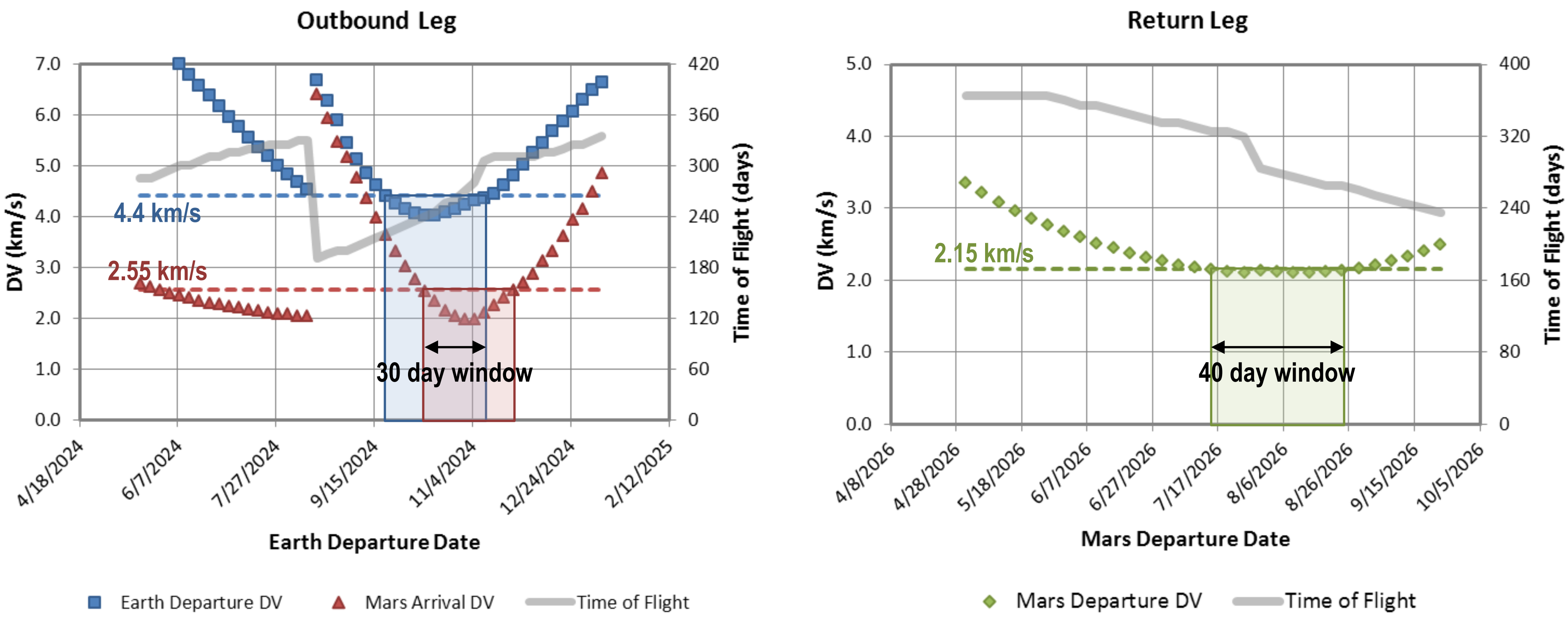


- Delta-Vs required to achieve 30 day launch windows for outbound and return legs:

Maneuver	ΔV or Velocity	Opening Date	Closing Date
Earth Departure (ΔV)	4,400 m/s	9/2/2022	10/2/2020
Mars Arrival (ΔV)	2,500 m/s	4/10/2023	5/30/2023
Mars Departure (ΔV)	2,300 m/s	7/7/2024	8/11/2024
Earth Entry (maximum velocity)	11.5 km/s	5/8/2025	5/23/2025

- Surface stay duration is 400 days to 490 days

Delta-V Requirements: Mars Orbit Mission – Opportunity 2

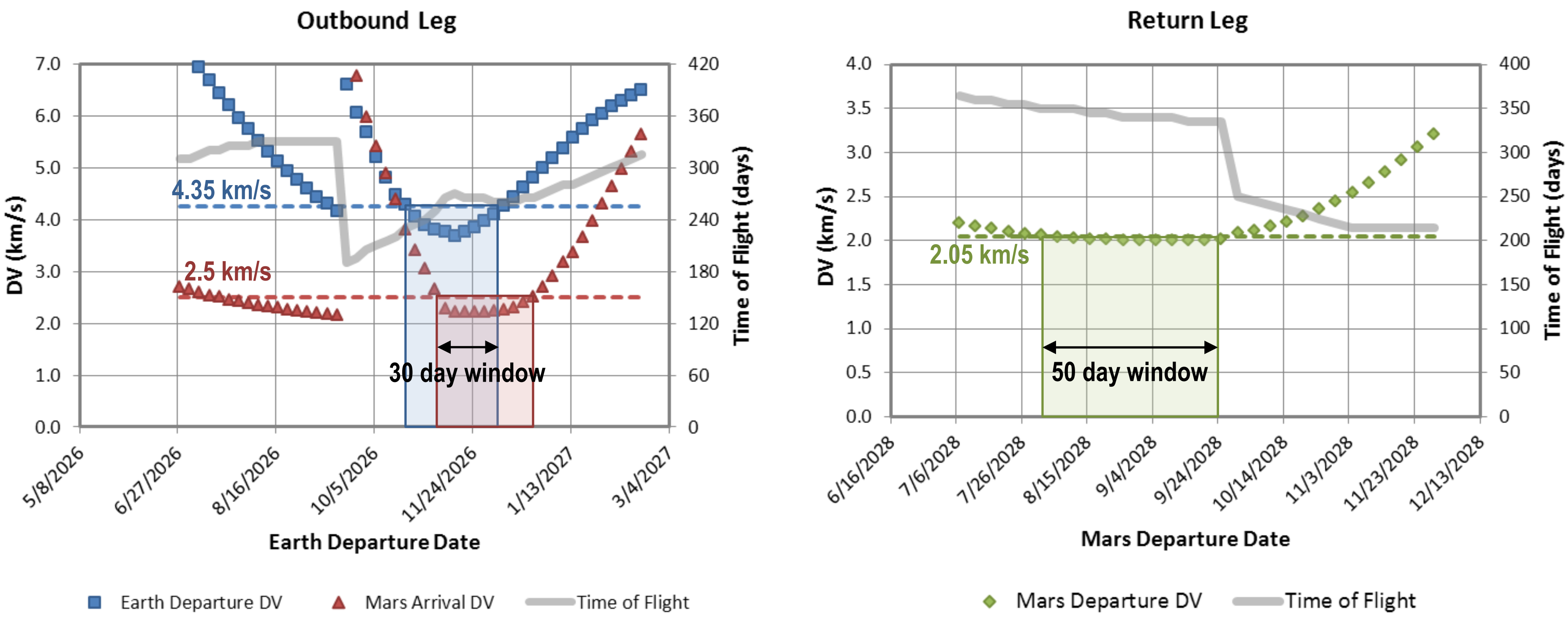


- Delta-Vs required to achieve 30 day launch windows for outbound and return legs:

Maneuver	ΔV or Velocity	Opening Date	Closing Date
Earth Departure (ΔV)	4,400 m/s	10/11/2024	11/10/2/24
Mars Arrival (ΔV)	2,550 m/s	6/8/2025	9/11/2025
Mars Departure (ΔV)	2,150 m/s	7/15/2026	8/24/2026
Earth Entry (maximum velocity)	12.0 km/s	5/11/2027	6/10/2027

- Surface stay duration is 300 days to 440 days

Delta-V Requirements: Mars Orbit Mission – Opportunity 3



- Delta-Vs required to achieve 30 day launch windows for outbound and return legs:

Maneuver	ΔV or Velocity	Opening Date	Closing Date
Earth Departure (ΔV)	4,250 m/s	11/10/2026	12/10/2026
Mars Arrival (ΔV)	2,500 m/s	8/2/2027	8/27/2027
Mars Departure (ΔV)	2,050 m/s	8/6/2028	9/25/2028
Earth Entry (maximum velocity)	12.4 km/s	7/22/2029	8/26/2029

- Surface stay duration is 340 days to 420 days

Delta-V Requirements: Mars Orbit Mission – Opportunity 4

- SpaceWorks recommends the following delta-Vs for Mars orbit missions
  - The recommended delta-Vs will allow for 30 day minimum launch windows for Earth departure and Mars departure in the 2020-2030 timeframe
  - The split between Mars arrival and Mars departure delta-V changes with each mission, but the sum of these two delta-Vs stays relatively constant. Both delta-Vs would be performed by the same CPS in this study.

Data Point	Earth Departure ΔV (m/s)	Mars Arrival ΔV (m/s)	Mars Departure ΔV (m/s)	Total Mars ΔV (m/s)
Opportunity 1	4,000	2,200	2,550	4,750
Opportunity 2	4,400	2,500	2,300	4,700
Opportunity 3	4,400	2,550	2,150	4,650
Opportunity 4	4,250	2,500	2,050	4,550
SpaceWorks Recommendation	**4,400**	**2,200**	**2,550***	**4,750**

* Maximum second burn delta-V represents a conservative estimate for boil-off

- Surface stay times for these missions range from 300 days to 550 days
- In-space time of flight varies from 210 days to 350 days
- Earth re-entry velocities vary from 11.5 to 12.4 km/s. This represents at maximum a 10% increase over entry velocities seen during a typical Apollo mission.[32]

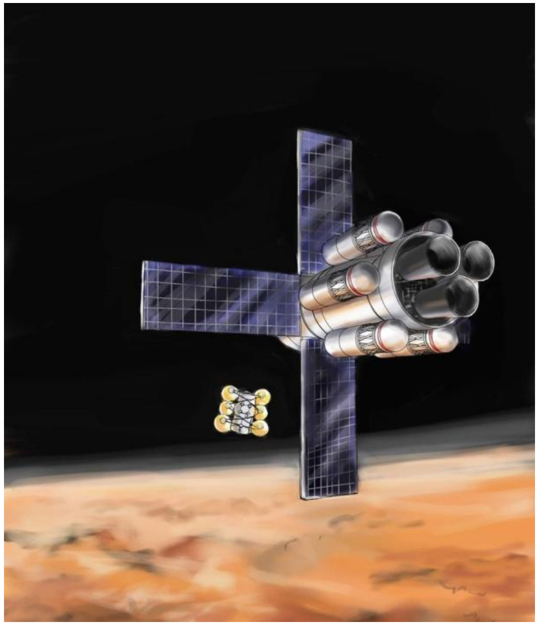
Delta-V Requirements: Mars Orbit Mission – Summary

Reference Missions Summary

- **SpaceWorks recommends using the following delta-Vs for each mission for the CPSs:**

Mission	CPS 1		CPS 2	
	ΔV 1 (m/s)	ΔV 2 (m/s)	ΔV 1 (m/s)	ΔV 2 (m/s)
Earth-Moon L1	3,070	730	-	-
Low Lunar Orbit	3,150	950	-	-
NEO Encounter	3,350	-	2,000	2,150*
Mars Orbit	4,400	-	2,200	2,550*

* Maximum second burn delta-V represents a conservative estimate for boil-off



Delta-V Requirements: Summary

Image Credit: SpaceWorks Enterprises, Inc.

TASK 2.0 Mission Scenario Analysis Tool (MSAT) Modeling

- SpaceWorks assumed the following masses for the four CPS payloads:

Payload	Mass (kg)	Applicable Mission
Orion / MPCV	21,250	1, 2, 3, 4
Lunar Lander	30,000	2
Long Duration Habitat	30,000	3, 4
Gateway Habitat	16,000	1

- SpaceWorks assumed the following delta-Vs for each mission for the CPSs:

Mission	CPS 1		CPS 2	
	ΔV 1 (m/s)	ΔV 2 (m/s)	ΔV 1 (m/s)	ΔV 2 (m/s)
Earth-Moon L1	3,070	730	-	-
Low Lunar Orbit	3,150	950	-	-
NEO Encounter	3,350	-	2,000	2,150*
Mars Orbit	4,400	-	2,200	2,550*

* Maximum second burn delta-V represents a conservative estimate for boil-off

MSAT Modeling: Payload and Delta-V Requirements

- Four design variables are identified in the statement of work:

Variable A: Propellant Mass Fraction	
0.75	Human Exploration Framework Team (HEFT) assumed low-end CPS mass fraction
0.85	Ares V EDS-like (Earth Departure Stage) mass fraction
0.90	Centaur-like mass fraction
0.95	High-end possible mass fraction

Variable B: Engine Specific Impulse (Isp)	
448 sec	J2-X
451 sec	RL10-A4-2
465 sec	RL10-B2 or Next Generation Engine

Variable C: Boil-off Rate	
0.001%/day	Requires active cooling
0.01%/day	Aggressive boil-off rate with passive thermal protection
0.05%/day	Reasonable near-term boil-off rate with passive thermal protection
0.1%/day	Centaur boil-off rate achievable via already reviewed modifications

Variable D: LEO Duration	
1 day	Constellation approach requiring same day launch
1 month	One month centers provides reasonable time to enable launch of two vehicles
6 months	Provides time to launch multiple vehicles

MSAT Modeling: Design Variables

- Mission Scenario and Analysis Tool (MSAT) was developed in 2004 by SpaceWorks Engineering as an architecture-level modeling tool that combines various disciplinary design models into one overarching simulation

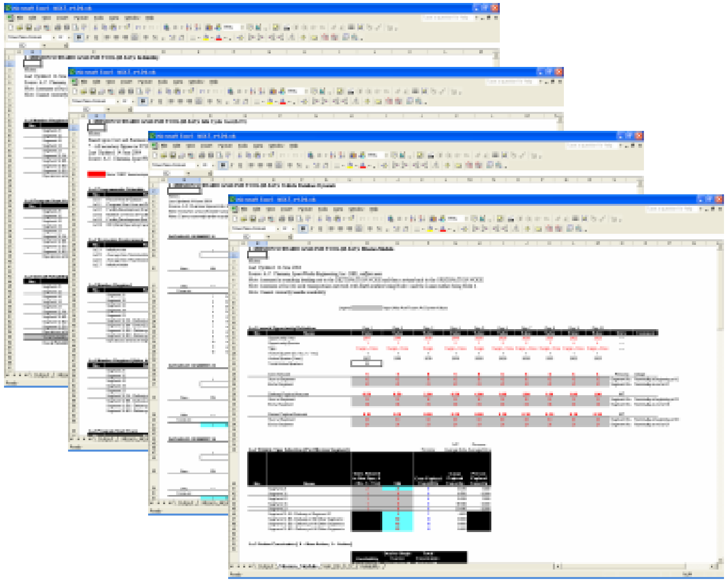
- MSAT models can be used to measure architecture-level impacts of design variables on multi-disciplinary figures of merit including weights, life cycle costs, and reliability metrics
    - These models allow the architecture to be analyzed probabilistically; probability distributions function can associated with the input variables to generate output variable distributions and confidence levels
    - Using optimization software, the architecture can be optimized to any of the figures of merit, for example the lowest cost mission design

- MSAT models are typically built in ModelCenter in order to directly incorporate disciplinary models in trajectory, aerodynamics, etc. though simpler implementations can be made in Excel or other design environments

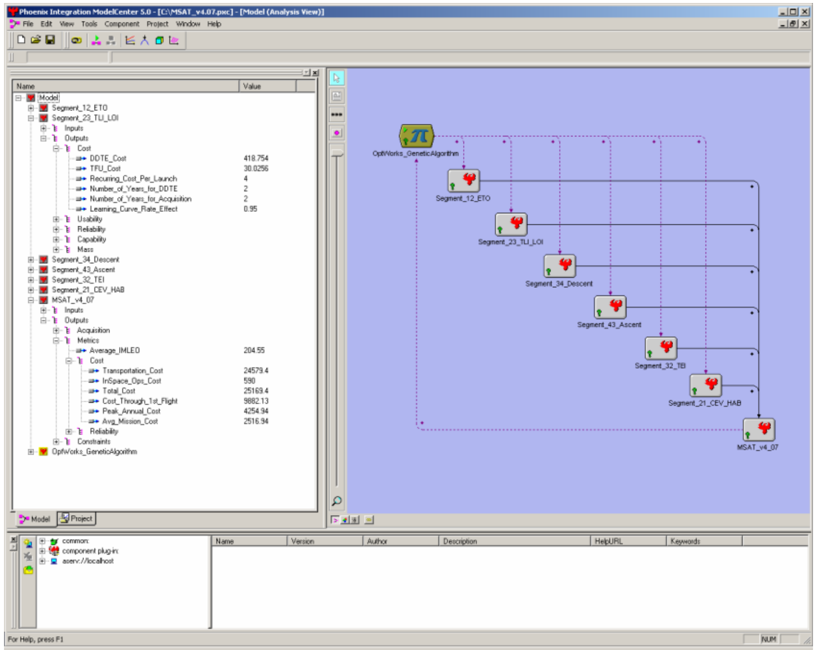
MSAT Modeling: Overview

- Below is a model of a typical MSAT implementation of ModelCenter:

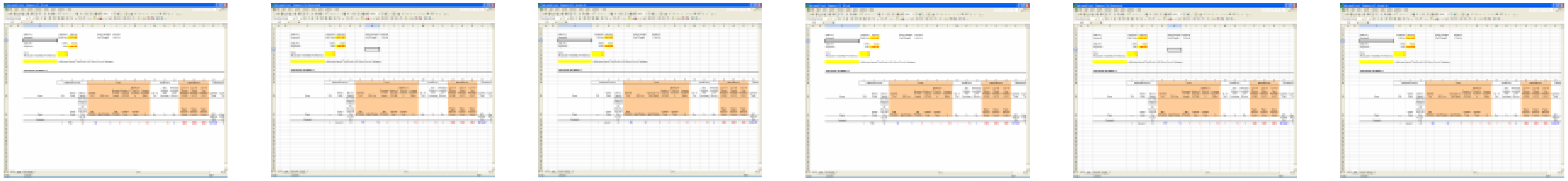
MSAT Core



ModelCenter© Implementation



Vehicle Databases



MSAT Modeling: ModelCenter Implementation

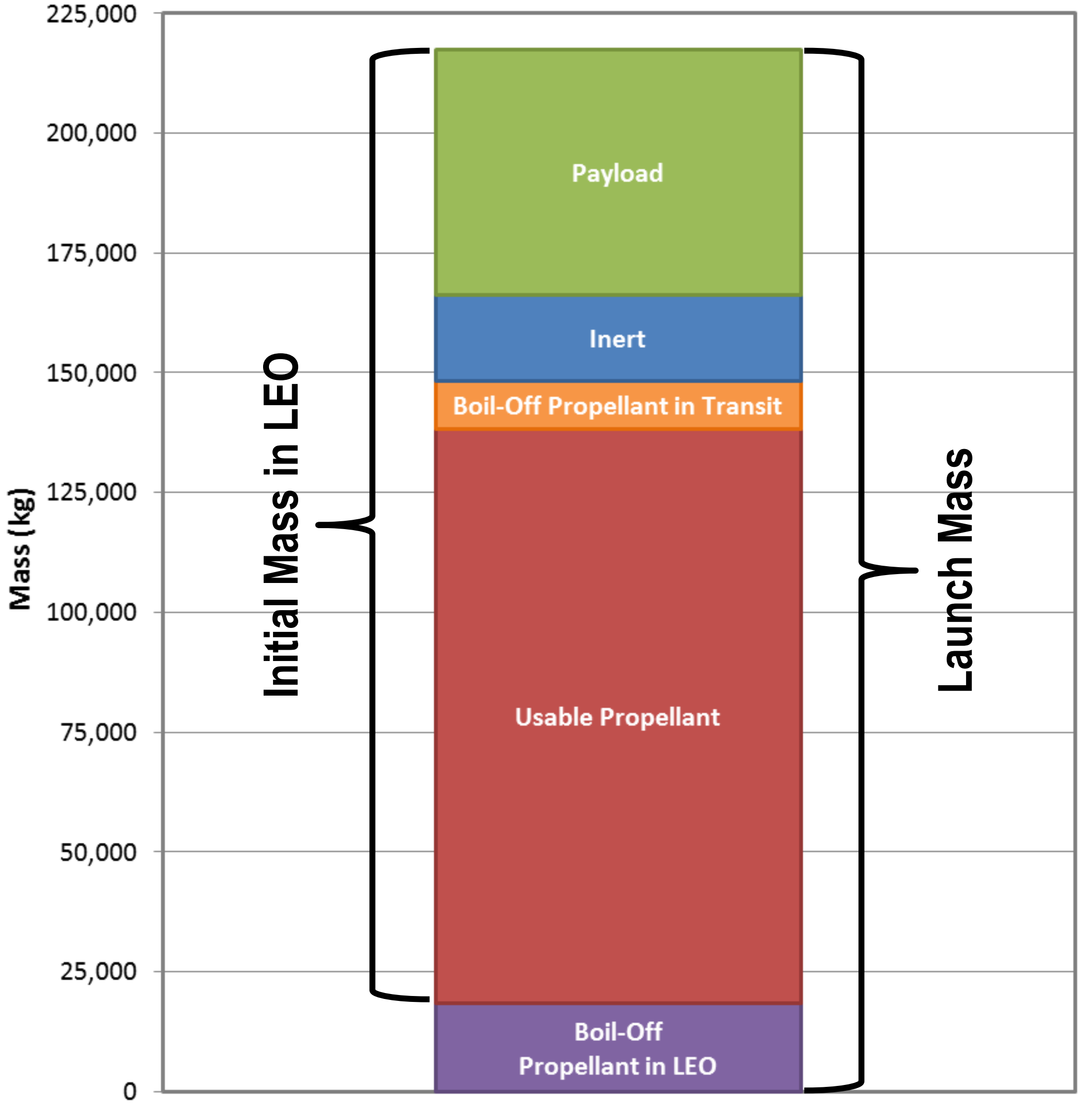
- SpaceWorks has developed a MSAT model to size the CPSs required for each mission

- This MSAT model is implemented in Microsoft Excel and uses the built-in circular reference functionality to accomplish vehicle closure automatically whenever a new set of design variables is selected

- This simplified model estimates the inert and propellant mass for each stage using the following closure process:
    1. Determine required mass ratios from rocket equation, required mission delta-Vs, and selected Isp values assuming instantaneous delta-V application
    2. Guess an inert mass for each stage, then determine the available usable propellant mass from the selected Propellant Mass Fraction and required payload masses
    3. Determine additional unusable boil-off propellant mass required from Boil-Off Rates and required mission durations
    4. Determine the required usable propellant mass from the required mass ratios
    5. Iterate on the inert mass guess until the available total propellant mass equals the required total propellant mass
    6. Determine additional boil-off propellant in LEO from the total propellant mass and selected LEO Duration

- A VBA macro sweeps through all combinations of the design variables and records the inert and propellant masses, initial mass in LEO, and additional boil-off propellant calculated for each case

MSAT Modeling: Model Structure



- Initial Mass in LEO (IMLEO) is defined as the mass of the combined stack of CPS and payloads at the beginning of the first mission maneuver. IMLEO **does not include** the boil-off propellant lost during the selected LEO Duration period

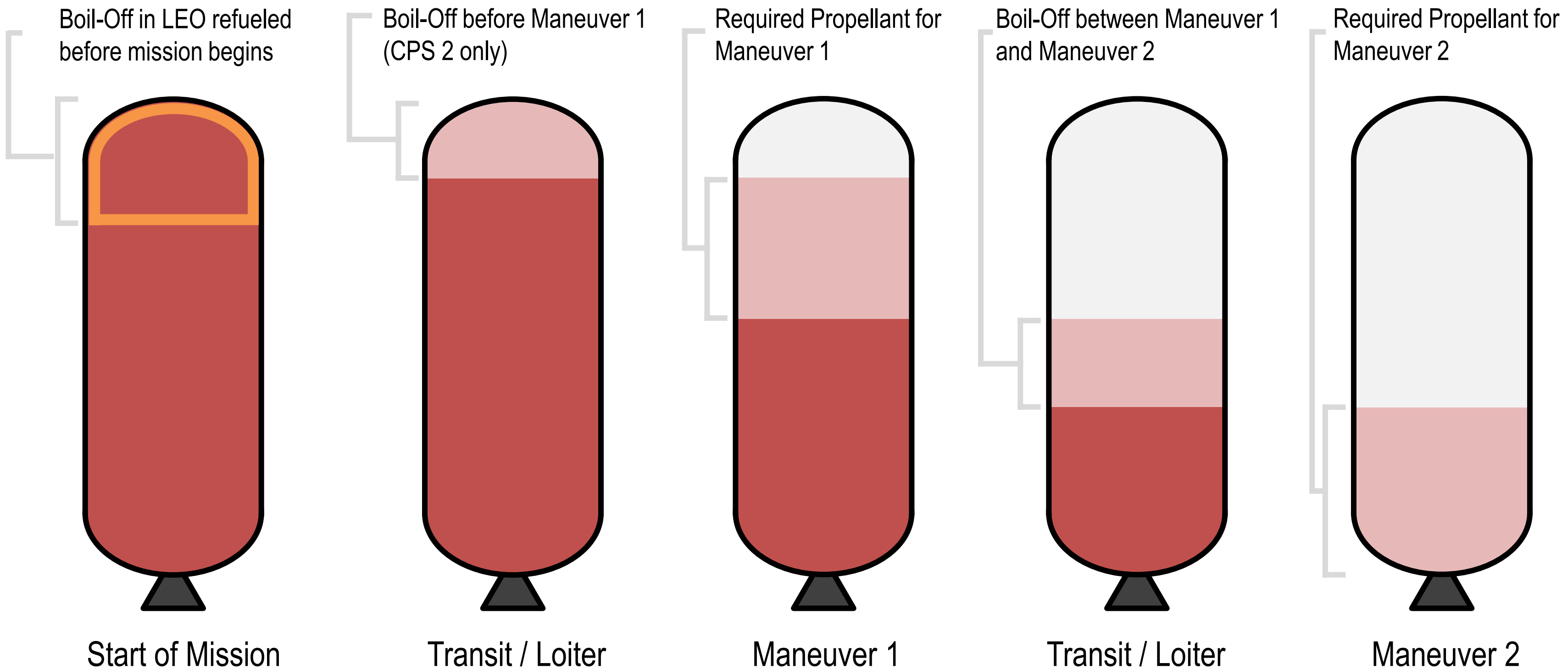
- Launch Mass is defined as the total mass that must be launched into orbit. Launch Mass **does include** the boil-off propellant lost during the selected LEO Duration period.

- The boil-off propellant lost during the LEO Duration period **is not included** in the CPS Propellant Mass Fraction when the inert mass of the CPS is being determined. It is assumed that this boil-off propellant is replenished from a propellant depot before the mission commences.

- The boil-off propellant lost during the mission away from LEO and in transit **is included** in the Propellant Mass Fraction when the inert mass is being determined.

MSAT Modeling: Mass Definitions and Boil-Off Assumptions

- Boil-off rates are measured as a percent of total propellant mass lost per day
- The mass loss rate is calculated at the beginning of the mission and held constant throughout the mission
  - The impact of this assumption was investigated during the study
  - Additional data was generated with the mass loss rate calculated as a percent of the current remaining propellant. This is an integral calculation over the mission time frame.
- It is assumed that the boil-off propellant mass generated before any particular maneuver is vented before that maneuver occurs



MSAT Modeling: Boil-Off Methodology

Mission 1: Earth-Moon L1

- SpaceWorks assumed the following masses for the CPS payloads:

Payload	Mass (kg)	Application Mission
Orion / MPCV	**21,250**	**1, 2, 3, 4**
Lunar Lander	30,000	2
Long Duration Habitat	30,000	3, 4
Gateway Habitat	**16,000**	**1**

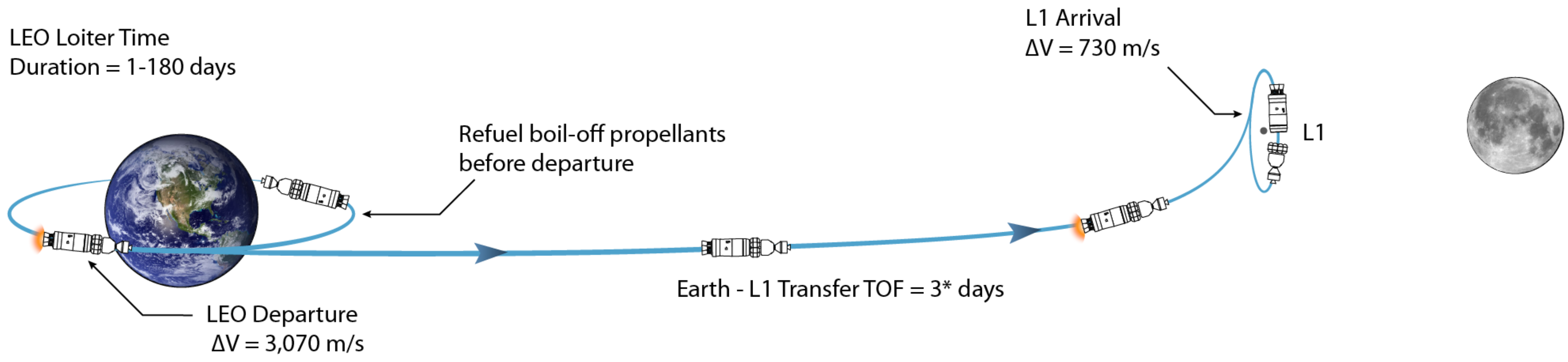
- SpaceWorks assumed the following delta-Vs for this mission for the CPS:

Mission	CPS 1		CPS 2	
	ΔV 1 (m/s)	ΔV 2 (m/s)	ΔV 1 (m/s)	ΔV 2 (m/s)
Earth-Moon L1	**3,070**	**730**	**-**	**-**
Low Lunar Orbit	3,150	950	-	-
NEO Encounter	3,350	-	2,000	2,150*
Mars Orbit	4,400	-	2,200	2,550*

* Maximum second burn delta-V represents a conservative estimate for boil-off

- SpaceWorks assumed 4 days (96 hours) time of flight for the purposes of calculating propellant boil-off in transit

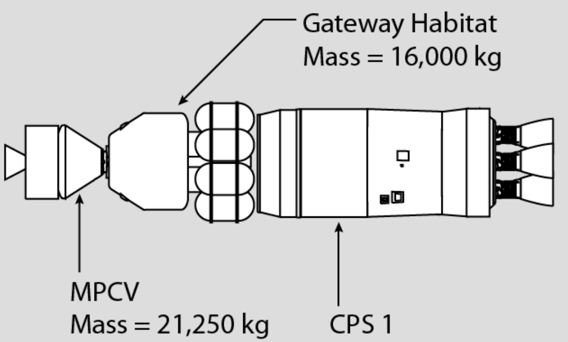
L1 Mission: Payload, Delta-V, and Time of Flight Assumptions



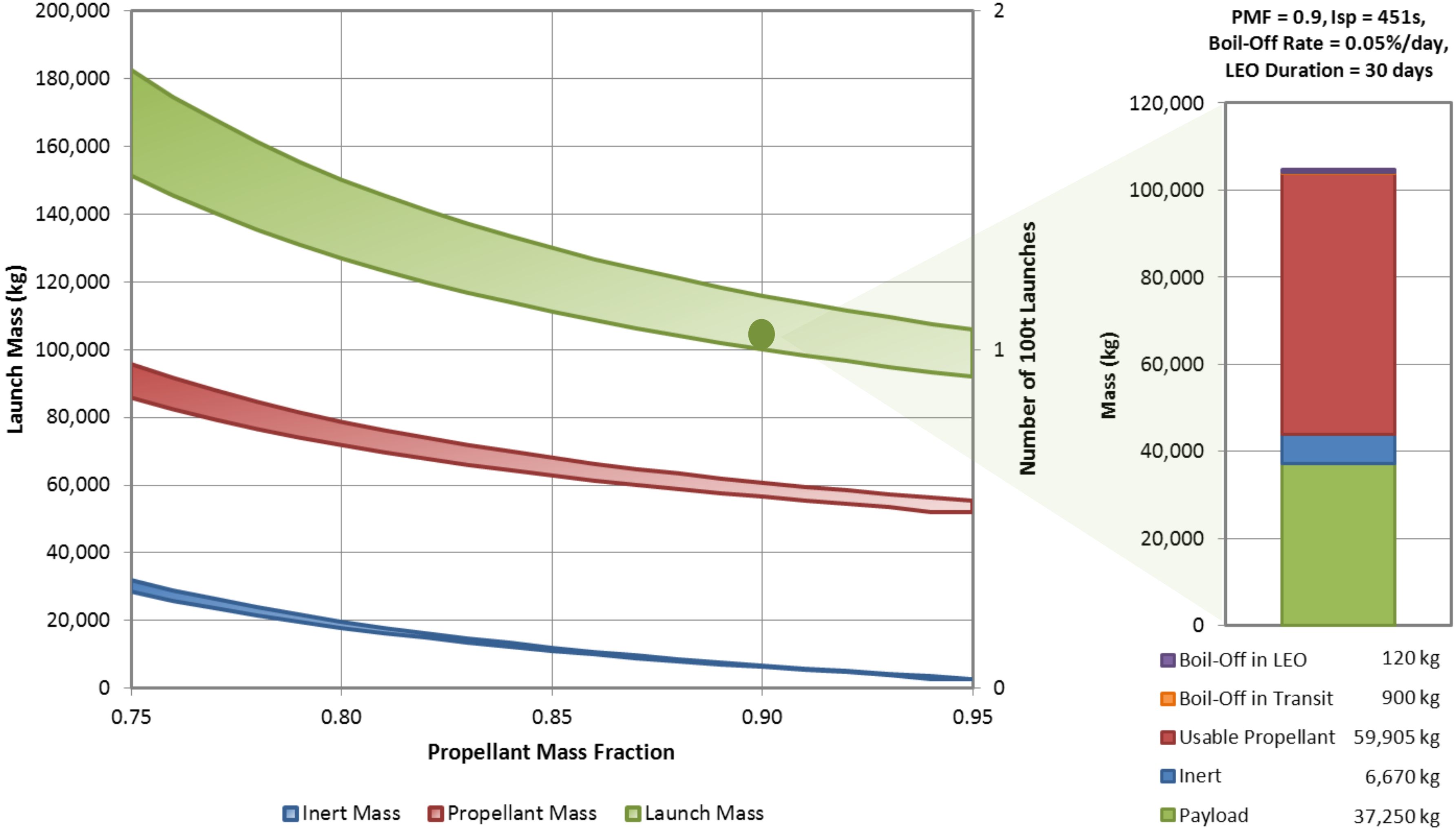
Parameter	Value
Trans-Lunar Injection (TLI) ΔV	3,070 m/s
Lunar Orbit Insertion (LOI) ΔV	730 m/s
Time of Flight (for ΔV determination)	3 days
Time of Flight (for boil-off calculations)	4 days
Multi-Purpose Crew Vehicle (MPCV) Mass	21,250 kg
Gateway Habitat Mass	16,000 kg

* Transfer TOF conservatively assumes 3 days for ΔV calculation but 4 days for boil-off calculations

Vehicle Configuration

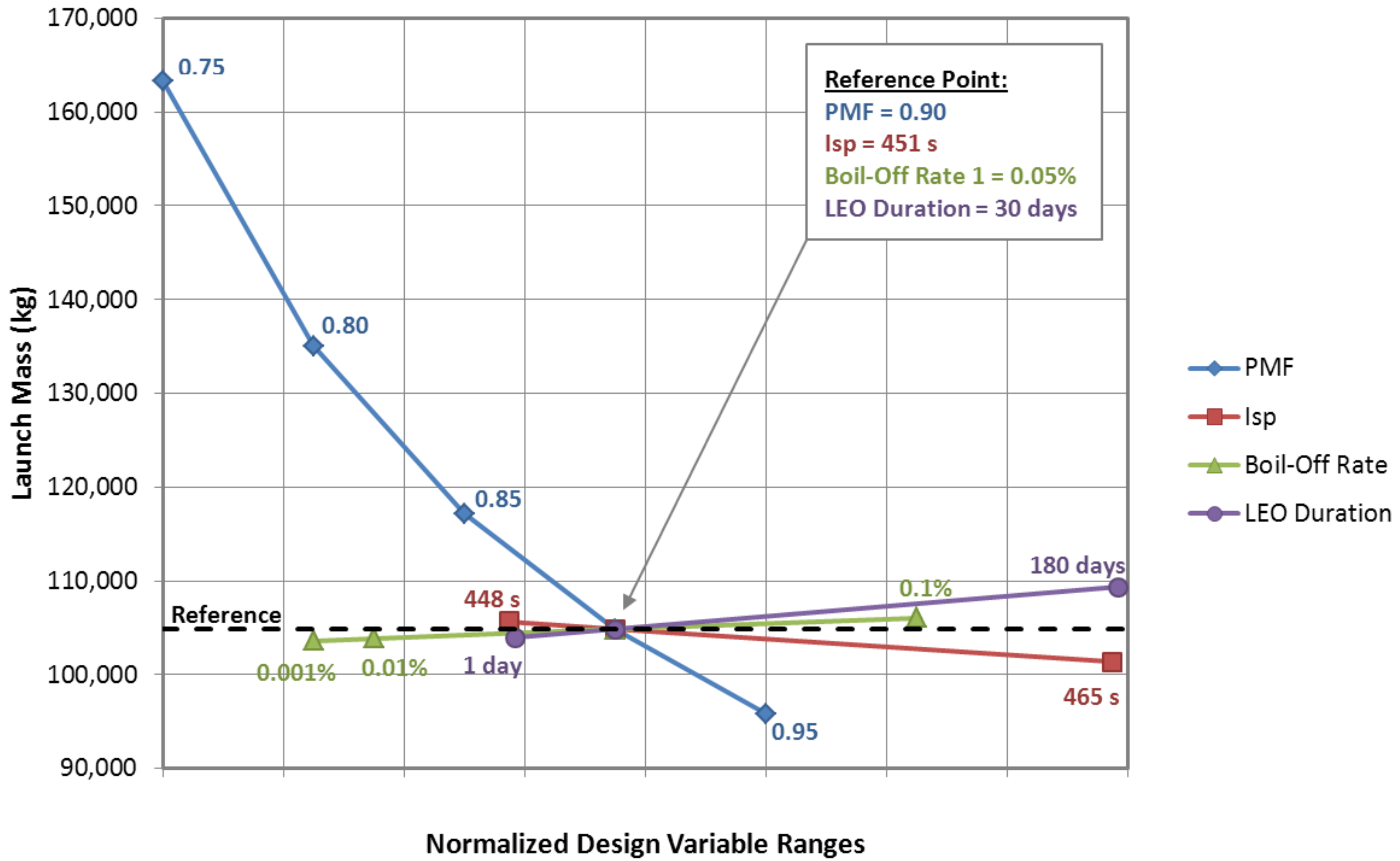


L1 Mission: Concept of Operations



- Results for all combinations of design variables shown above

L1 Mission: Design Variable Sweep Results (1 of 2)

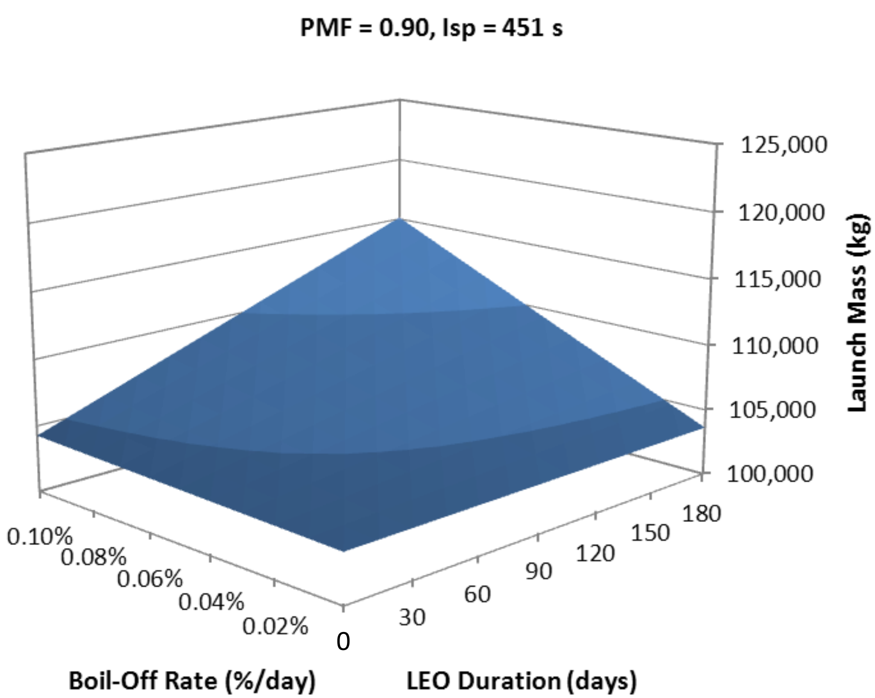


- Propellant Mass Fraction has the largest impact on Launch Mass and is the dominant design variable for this mission
- All other design variables have a much smaller impact

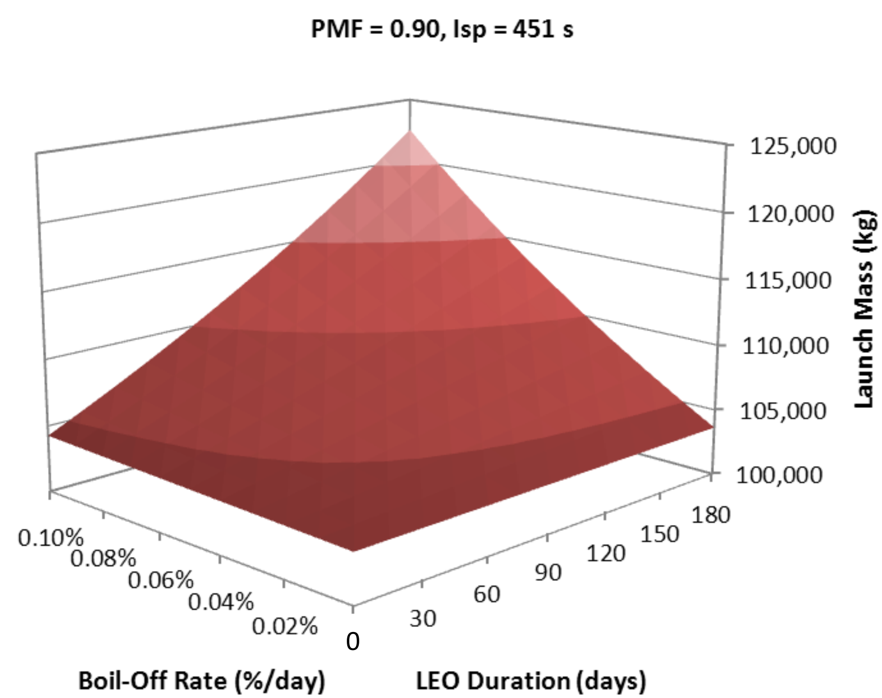
L1 Mission: Design Variable Sweep Results (2 of 2)

L1 Mission: Refueling in LEO before Departure

CPS Refueled Prior to Departure



No Refueling after Launch



- Clear coupling between Boil-Off Rate and LEO Duration with short mission times
- LEO Refueling Can Save up to ~10% in Launch Mass

Mission 2: Low Lunar Orbit (LLO)

- SpaceWorks assumed the following masses for the CPS payloads:

Payload	Mass (kg)	Application Mission
Orion / MPCV	**21,250**	**1, 2, 3, 4**
Lunar Lander	**30,000**	**2**
Long Duration Habitat	30,000	3, 4
Gateway Habitat	16,000	1

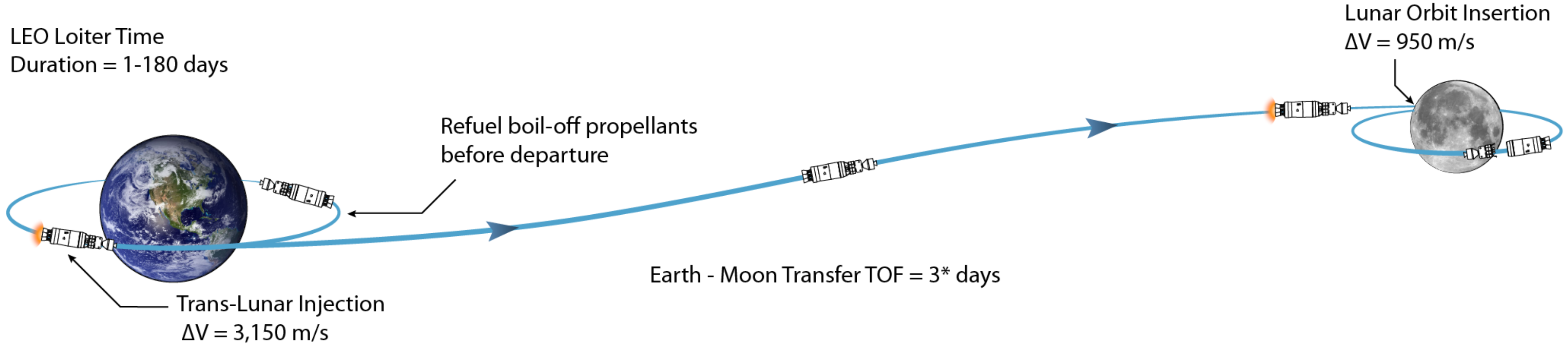
- SpaceWorks assumed the following delta-Vs for this mission for the CPS:

Mission	CPS 1		CPS 2	
	ΔV 1 (m/s)	ΔV 2 (m/s)	ΔV 1 (m/s)	ΔV 2 (m/s)
Earth-Moon L1	3,070	730	-	-
Low Lunar Orbit	**3,150**	**950**	**-**	**-**
NEO Encounter	3,350	-	2,000	2,150*
Mars Orbit	4,400	-	2,200	2,550*

* Maximum second burn delta-V represents a conservative estimate for boil-off

- SpaceWorks assumed 4 days (96 hours) time of flight for the purposes of calculating propellant boil-off in transit

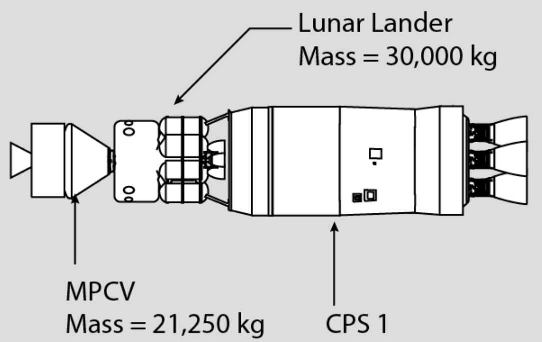
LLO Mission: Payload, Delta-V, and Time of Flight Assumptions



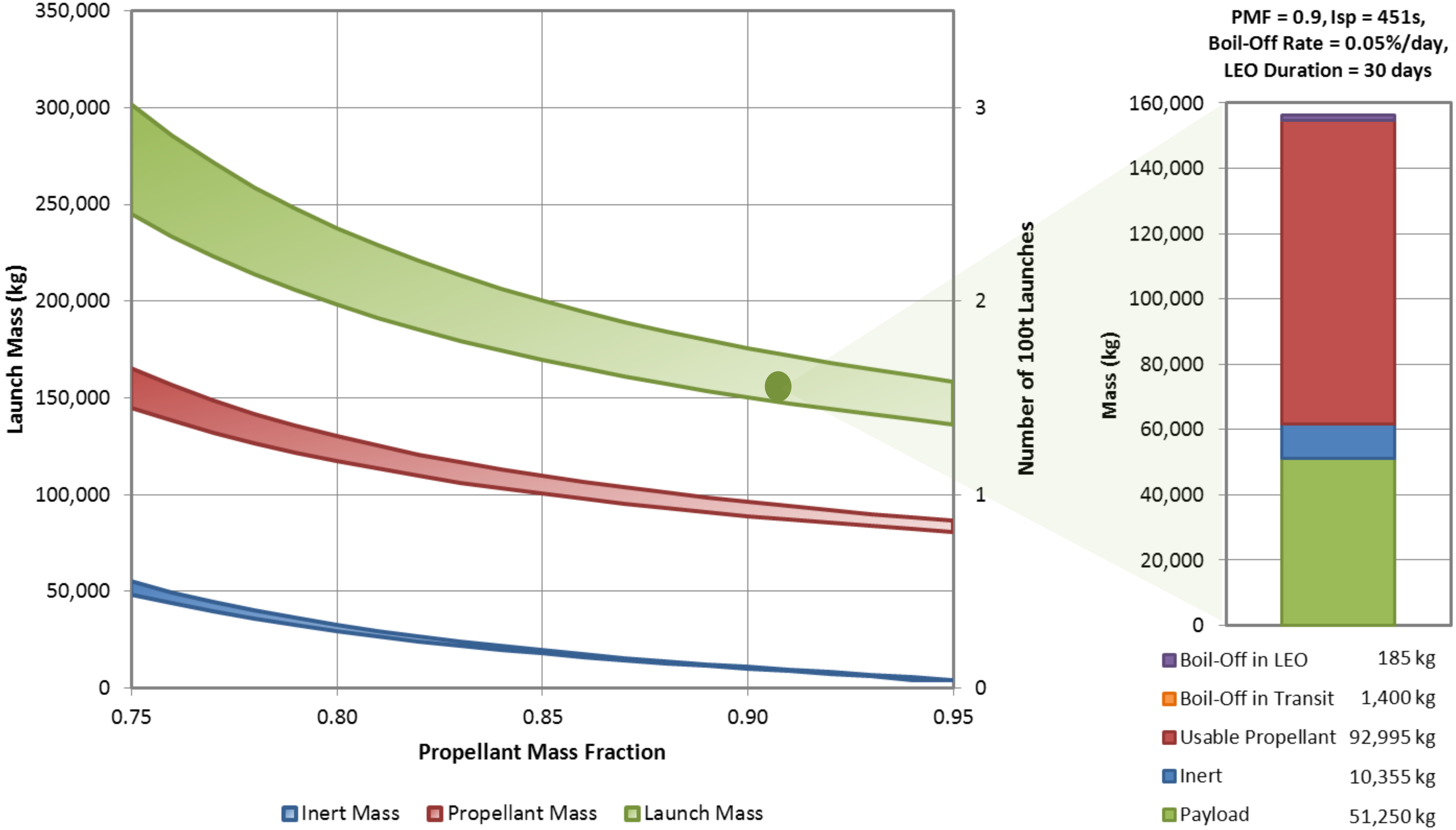
Parameter	Value
Trans-Lunar Injection (TLI) ΔV	3,150 m/s
Lunar Orbit Insertion (LOI) ΔV	950 m/s
Time of Flight (for ΔV determination)	3 days
Time of Flight (for boil-off calculations)	4 days
Multi-Purpose Crew Vehicle (MPCV) Mass	21,250 kg
Lunar Lander Mass	30,000 kg

* Transfer TOF conservatively assumes 3 days for ΔV calculation but 4 days for boil-off calculations

Vehicle Configuration

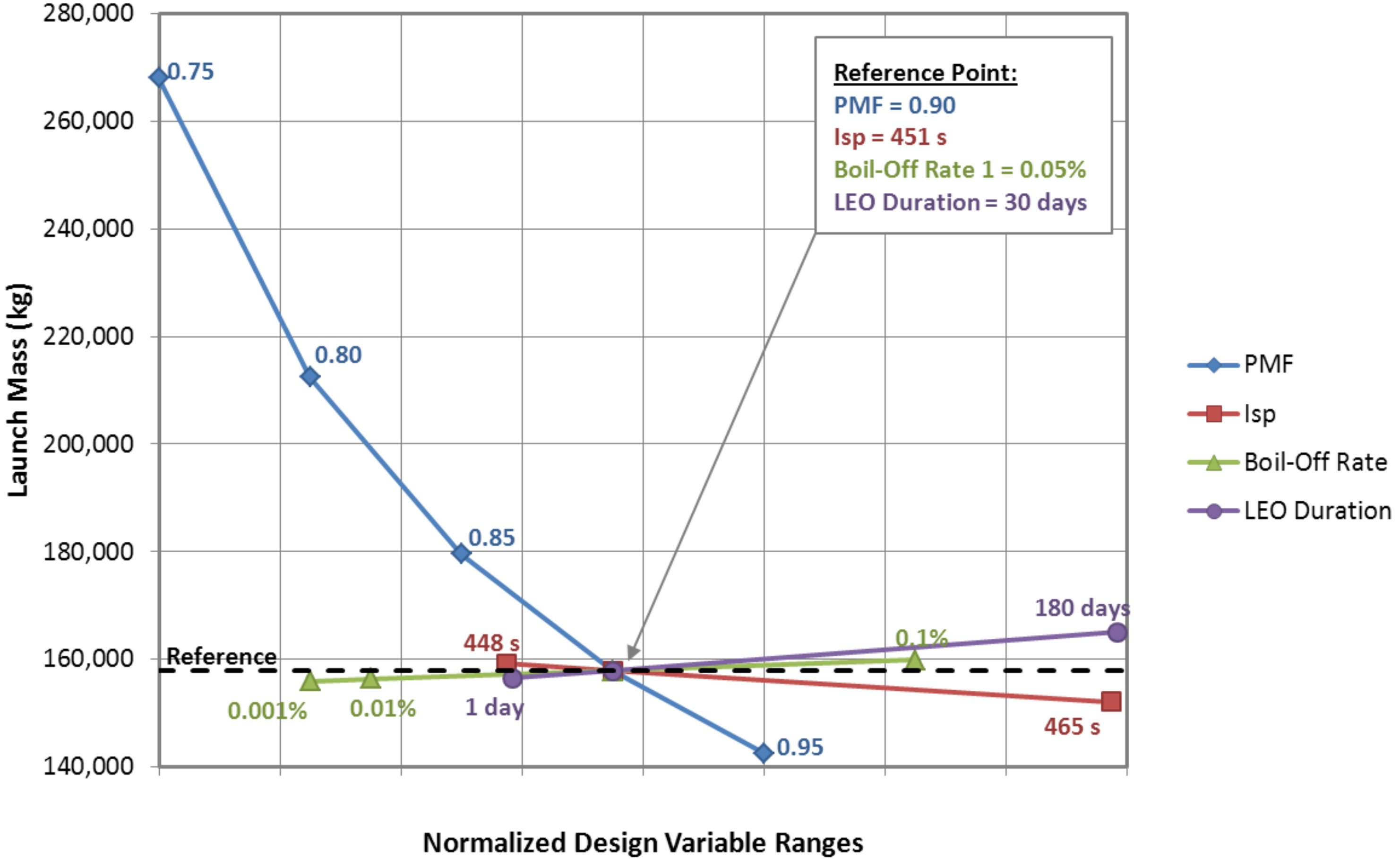


LLO Mission: Concept of Operations



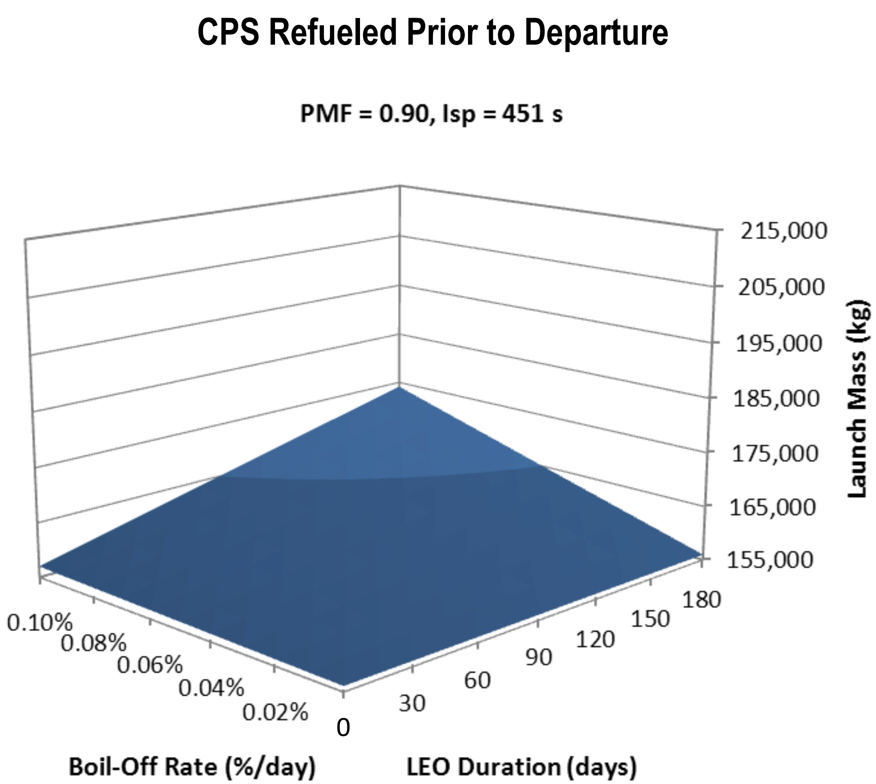
- Results for all combinations of design variables shown above

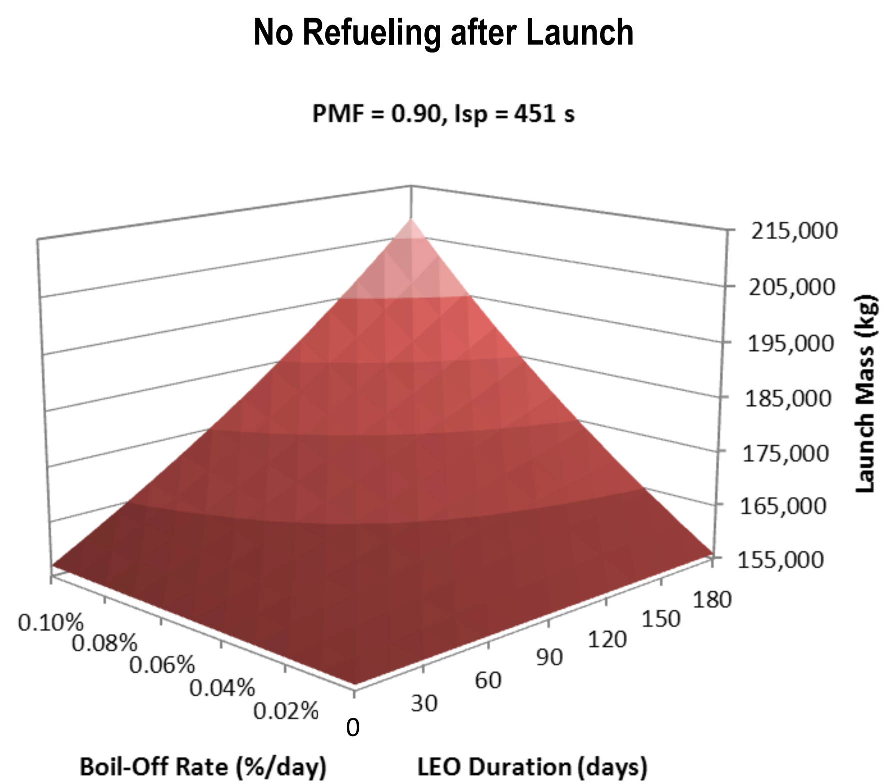
LLO Mission: Design Variable Sweep Results (1 of 2)



- Propellant Mass Fraction has the largest impact on Launch Mass and is the dominant design variable for this mission
- All other design variables have a much smaller impact

LLO Mission: Design Variable Sweep Results (2 of 2)





- Clear coupling between Boil-Off Rate and LEO Duration with short mission times
- LEO Refueling Can Save up to ~15% in Launch Mass

LLO Mission: Refueling in LEO before Departure

Mission 3: Near Earth Object (NEO) Encounter

- SpaceWorks assumed the following masses for the CPS payloads:

Payload	Mass (kg)	Application Mission
Orion / MPCV	**21,250**	**1, 2, 3, 4**
Lunar Lander	30,000	2
Long Duration Habitat	**30,000**	**3, 4**
Gateway Habitat	16,000	1

- SpaceWorks assumed the following delta-Vs for this mission for the CPS:

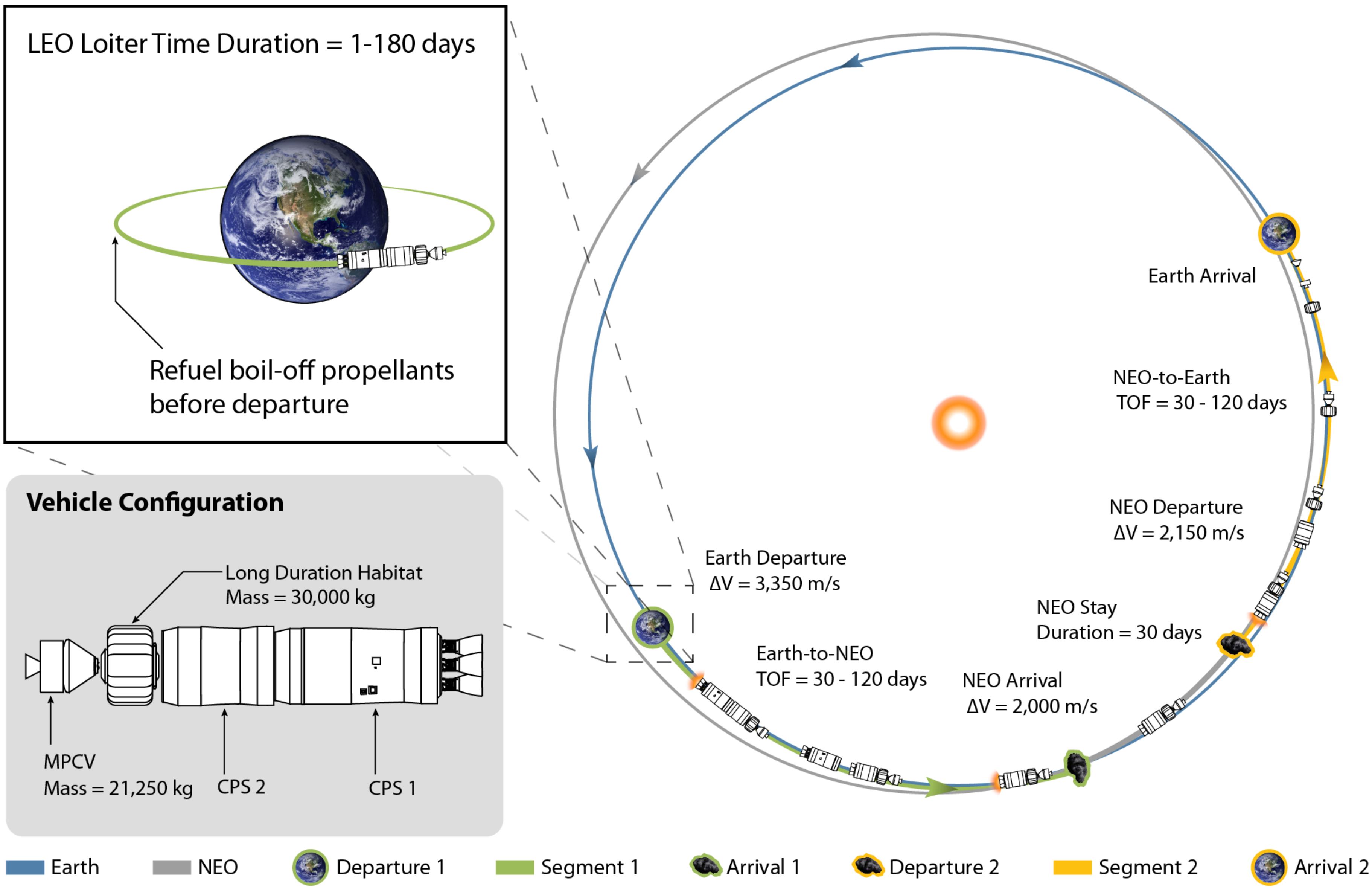
Mission	CPS 1		CPS 2	
	ΔV 1 (m/s)	ΔV 2 (m/s)	ΔV 1 (m/s)	ΔV 2 (m/s)
Earth-Moon L1	3,070	730	-	-
Low Lunar Orbit	3,150	950	-	-
NEO Encounter	**3,350**	**-**	**2,000**	**2,150***
Mars Orbit	4,400	-	2,200	2,550*

* Maximum second burn delta-V represents a conservative estimate for boil-off

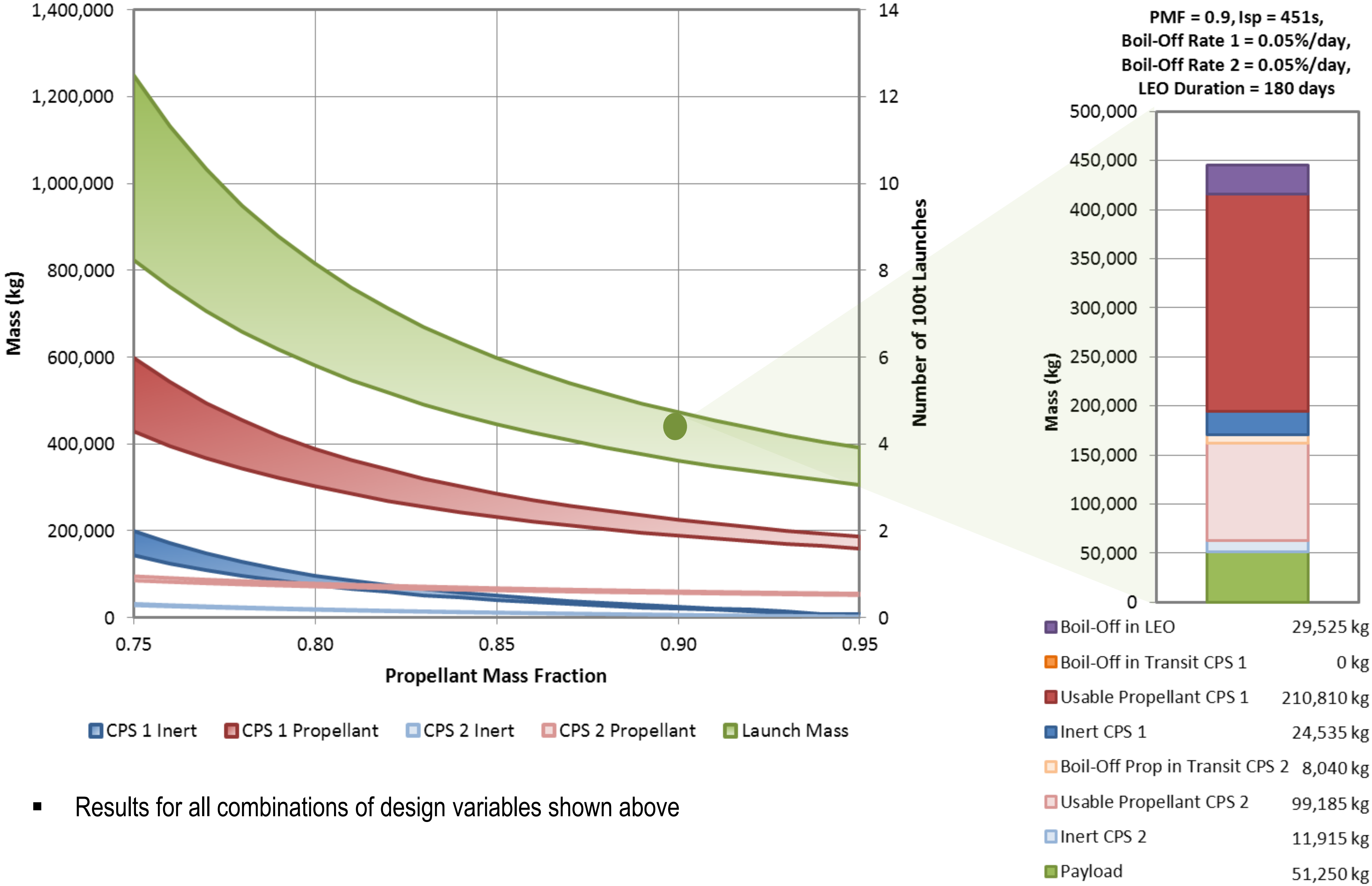
- SpaceWorks assumed the following mission times and durations for the purposes of calculation boil-off propellants:

Earth to NEO Transit (days)	NEO Stay (days)
120	**30**

NEO Mission: Payload, Delta-V, and Time of Flight Assumptions

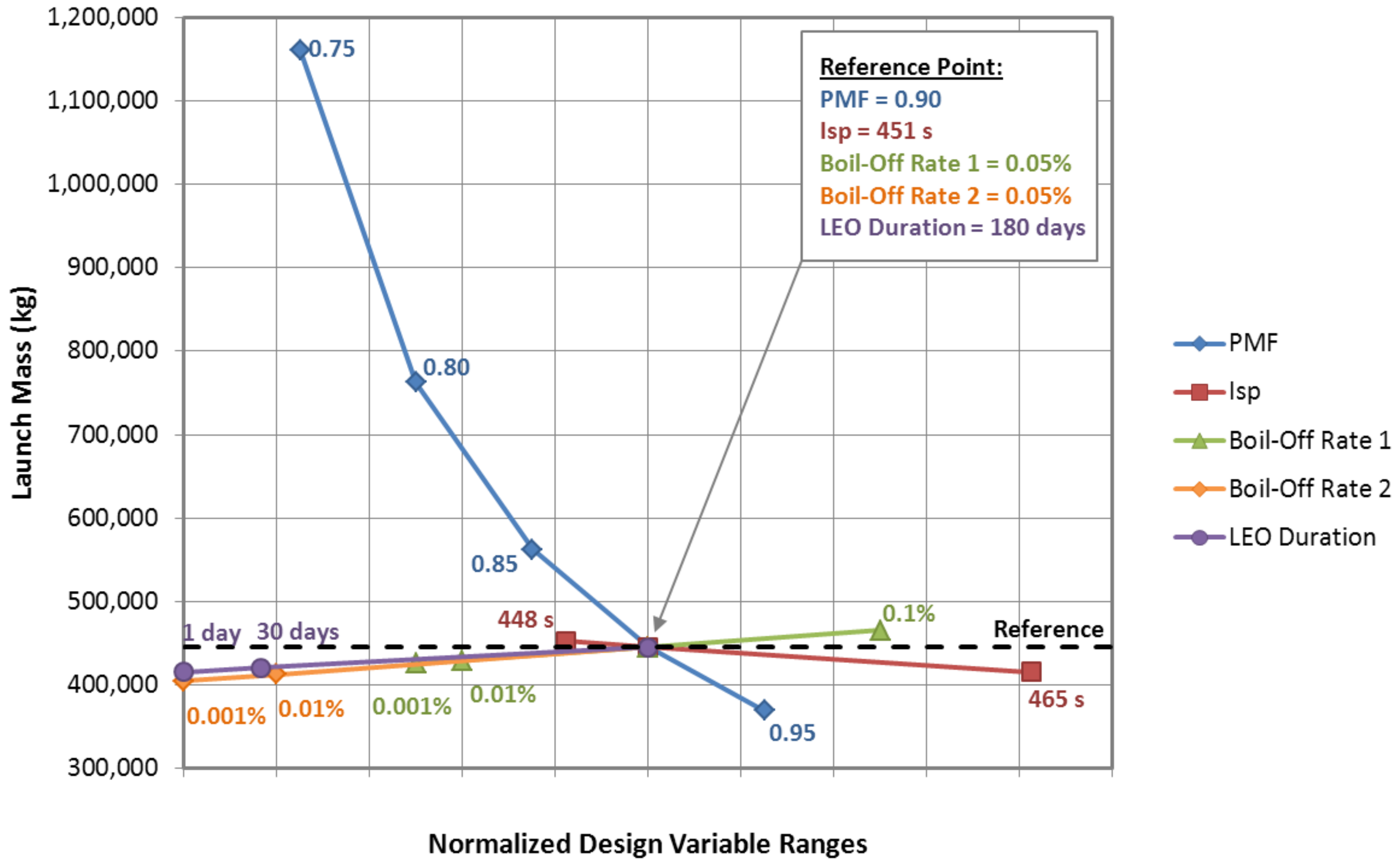


NEO Mission: Concept of Operations



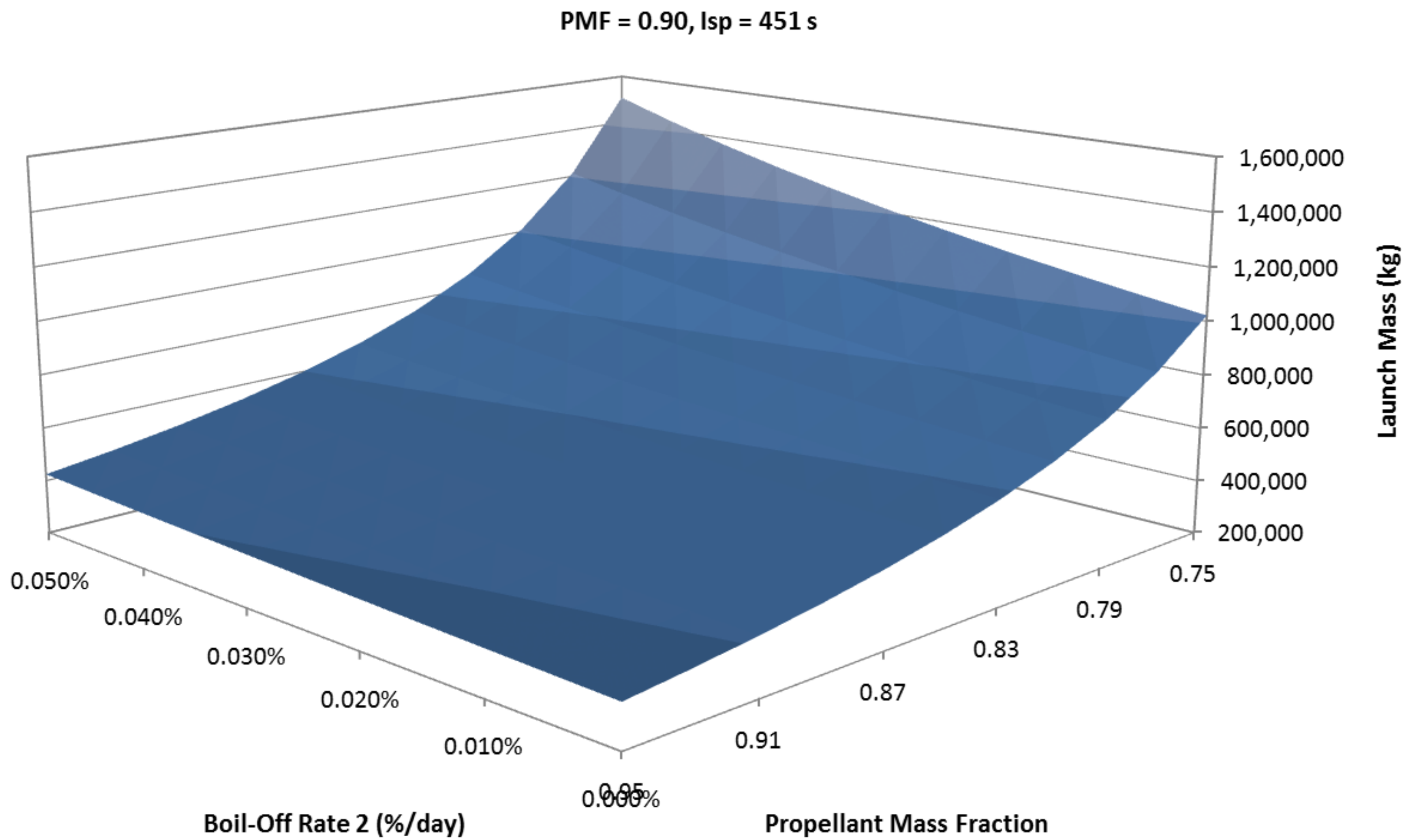
- Results for all combinations of design variables shown above

NEO Mission: Design Variable Sweep Results (1 of 2)



- Propellant Mass Fraction has the largest impact on Launch Mass and is the dominant design variable for this mission
- All other design variables have a much smaller impact

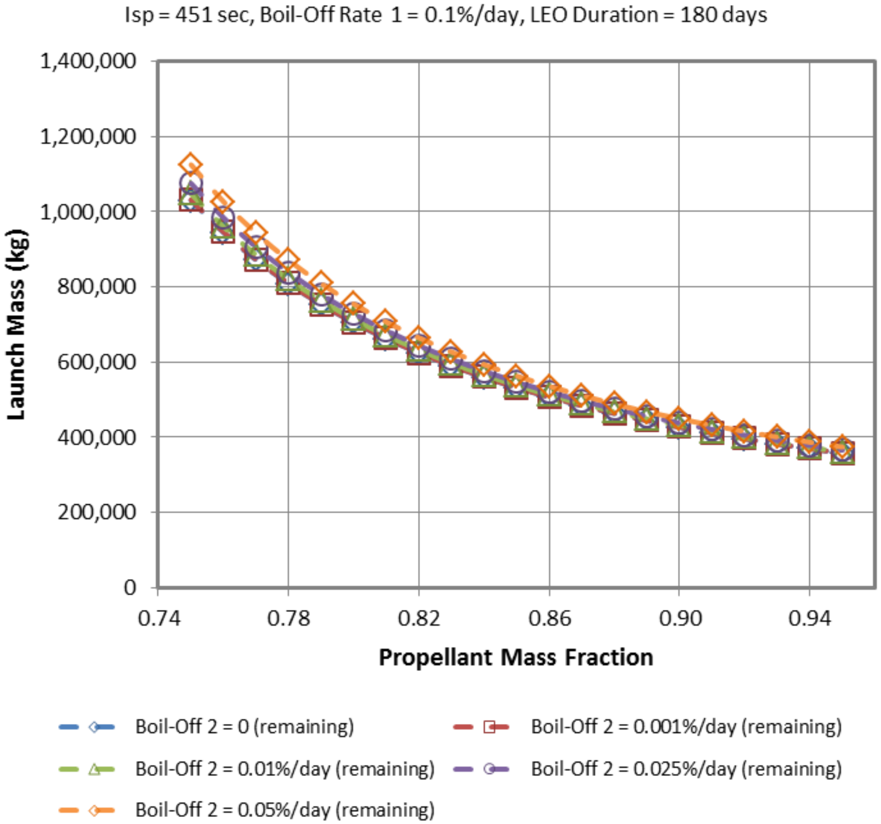
NEO Mission: Design Variable Sweep Results (2 of 2)



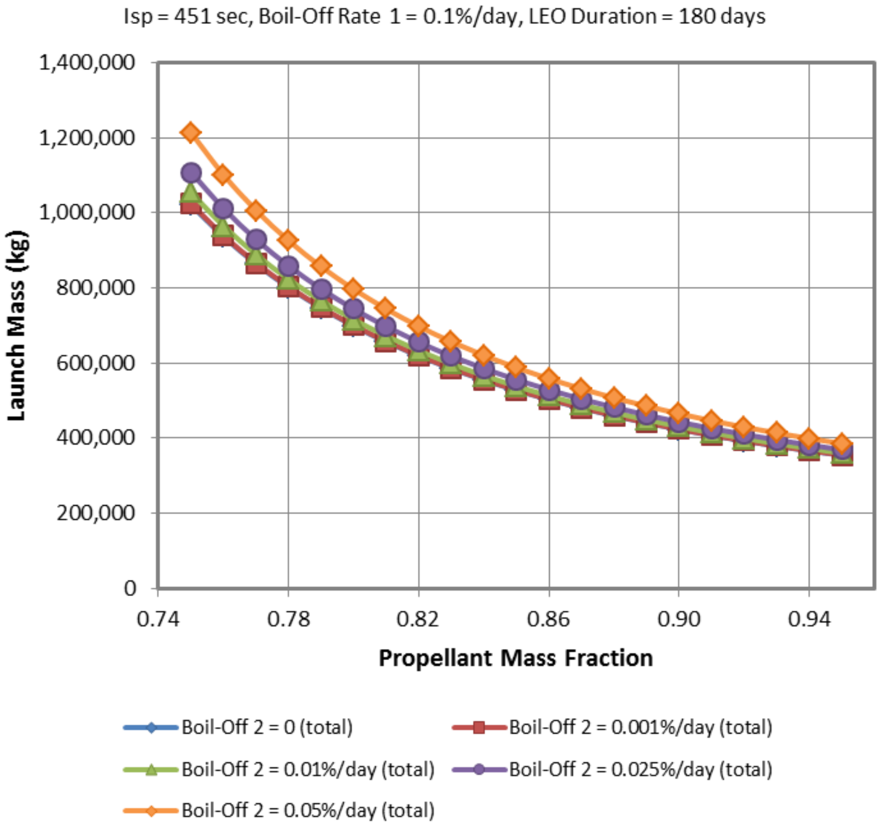
- Weak coupling between Boil-Off Rate for CPS 2 and Propellant Mass Fraction

NEO Mission: Design Variable Coupling – Boil-Off 2 and PMF

Remaining Method
Calculate boil-off losses based on current remaining propellant mass



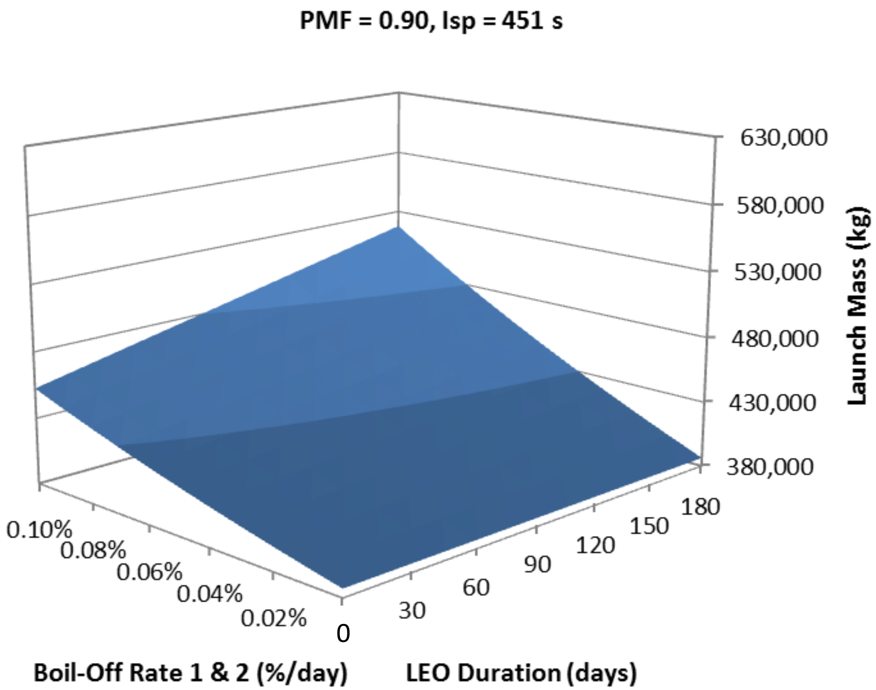
Total Method
Calculate boil-off losses based on total propellant mass at start of mission



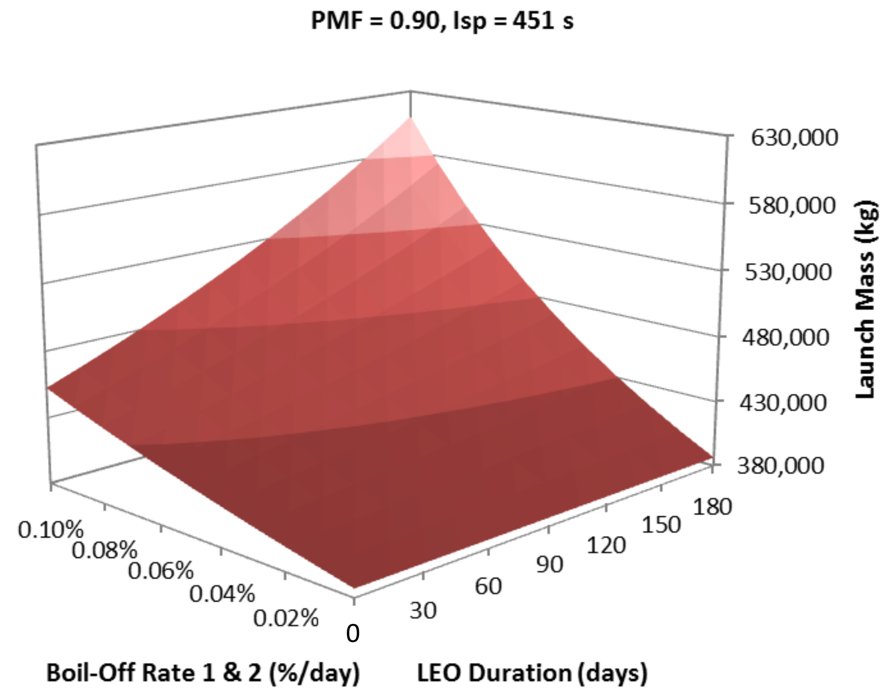
- Comparison of two methods for calculating impact of boil-off:
- For NEO mission with only 30 days at the NEO with a partially full CPS 2, there is little difference between the two methods

NEO Mission: Boil-Off Method Comparison

CPS Refueled Prior to Departure



No Refueling after Launch



- Coupling between Boil-Off Rate and LEO Duration less clear with multiple stages and longer mission times
- LEO Refueling Can Save up to ~15% in Launch Mass

NEO Mission: LEO Refuel Assumption Comparison

Mission 4: Mars Orbit

- SpaceWorks assumed the following masses for the CPS payloads:

Payload	Mass (kg)	Application Mission
Orion / MPCV	**21,250**	**1, 2, 3, 4**
Lunar Lander	30,000	2
Long Duration Habitat	**30,000**	**3, 4**
Gateway Habitat	16,000	1

- SpaceWorks assumed the following delta-Vs for this mission for the CPS:

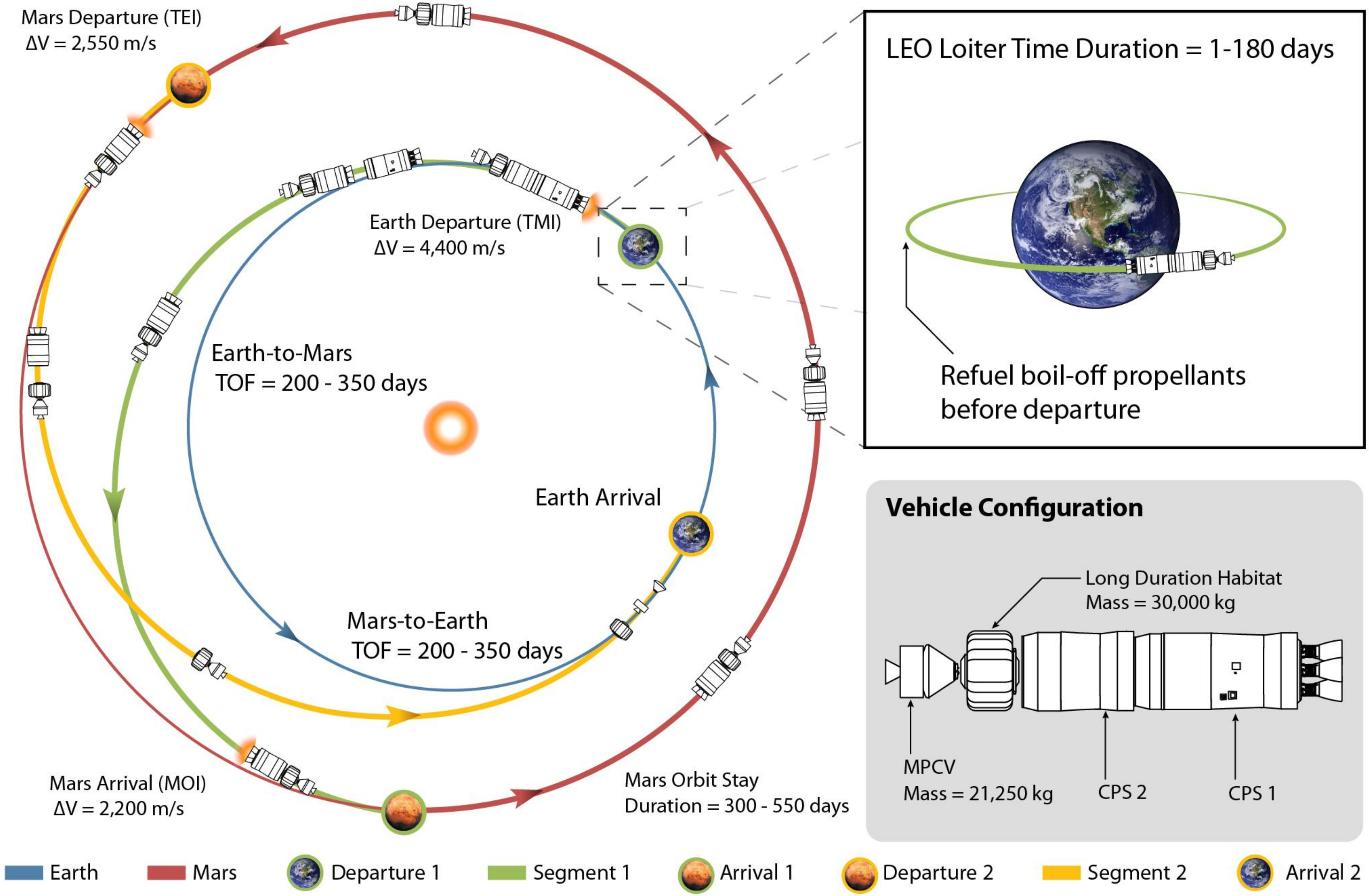
Mission	CPS 1		CPS 2	
	ΔV 1 (m/s)	ΔV 2 (m/s)	ΔV 1 (m/s)	ΔV 2 (m/s)
Earth-Moon L1	3,070	730	-	-
Low Lunar Orbit	3,150	950	-	-
NEO Encounter	3,350	-	2,000	2,150*
Mars Orbit	**4,400**	**-**	**2,200**	**2,550***

* Maximum second burn delta-V represents a conservative estimate for boil-off

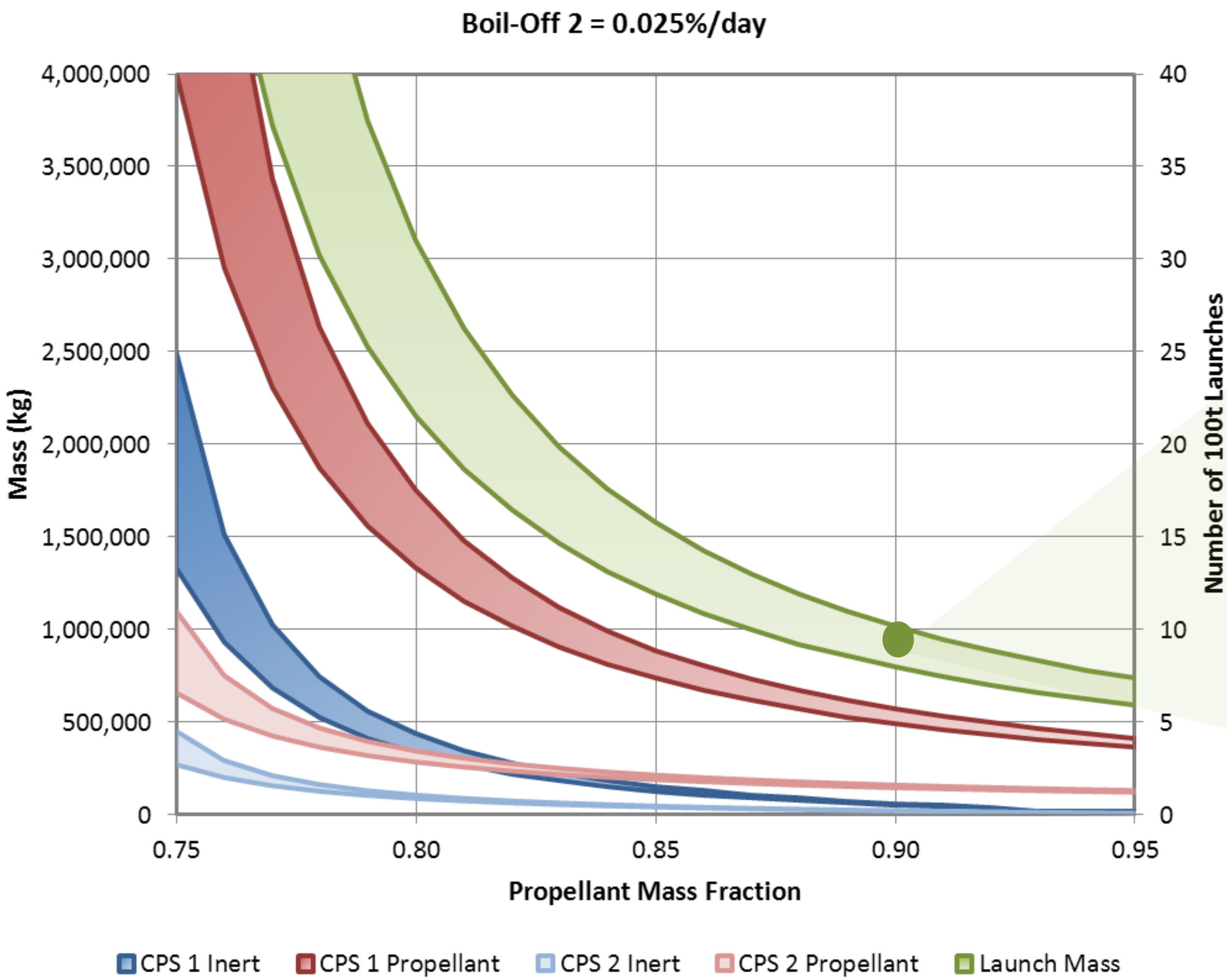
- SpaceWorks assumed the following mission times and durations for the purposes of calculation boil-off propellants:

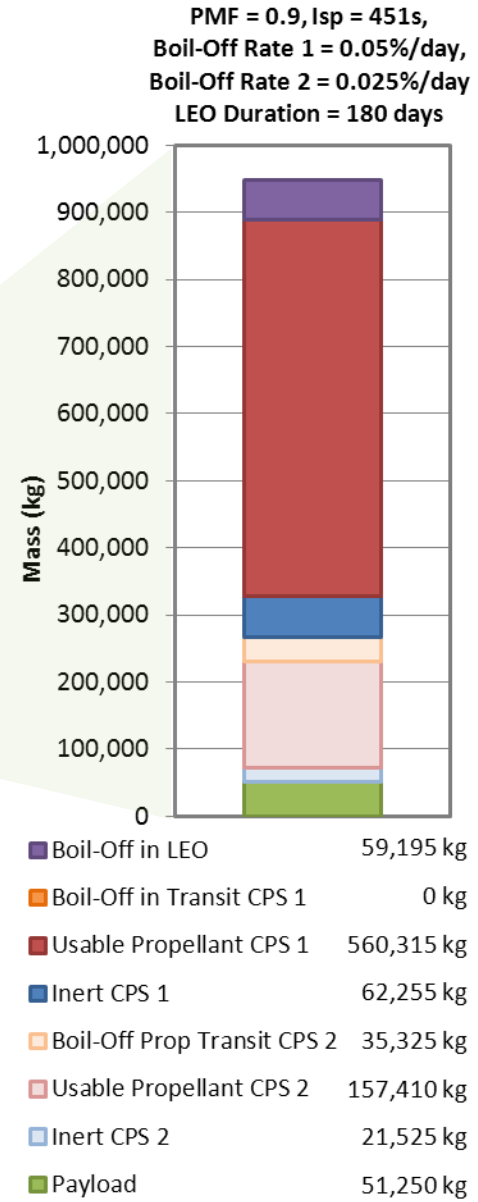
Earth to Mars Transit (days)	Mars Orbit Stay (days)
350	**500**

Mars Mission: Payload, Delta-V, and Time of Flight Assumptions



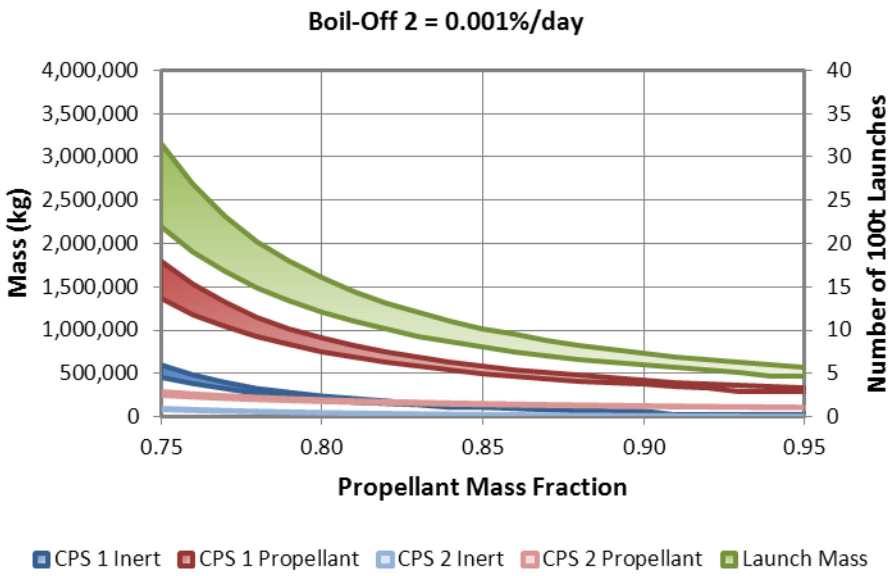
Mars Mission: Concept of Operations

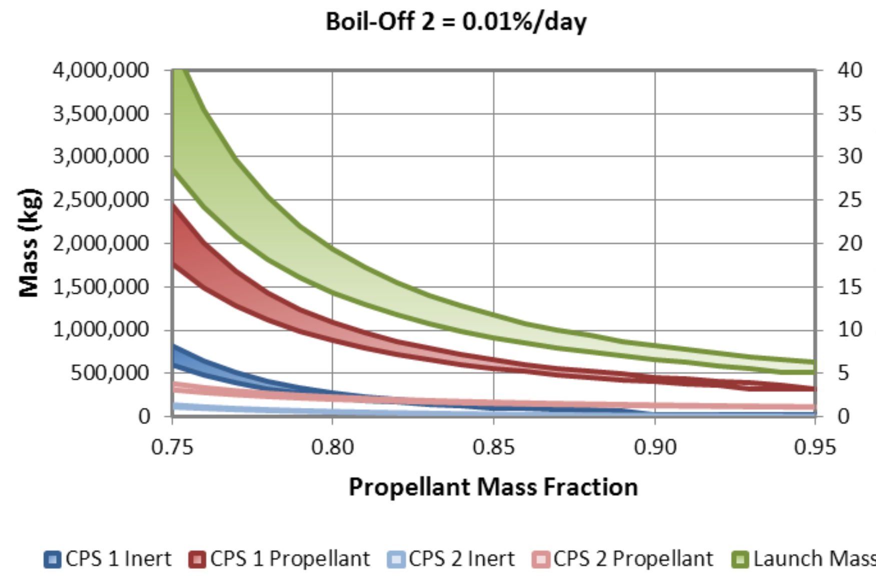


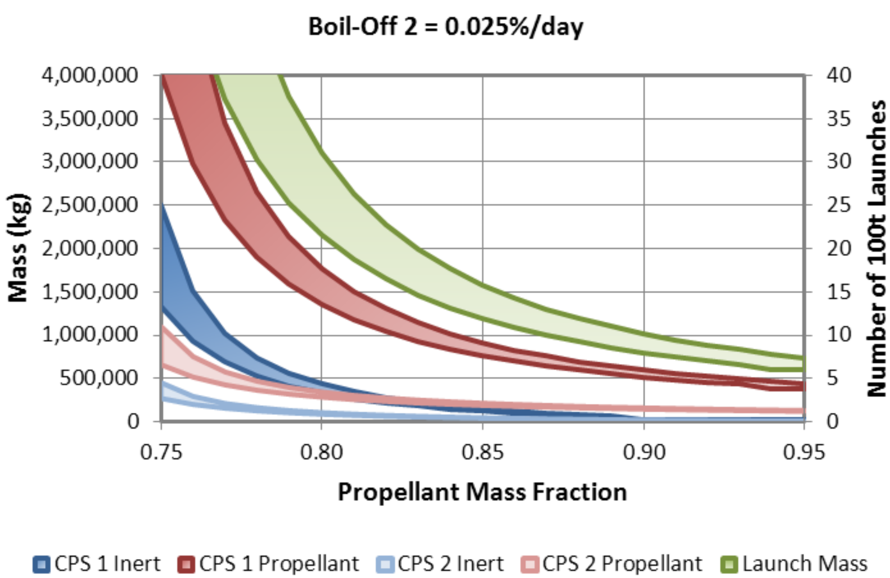


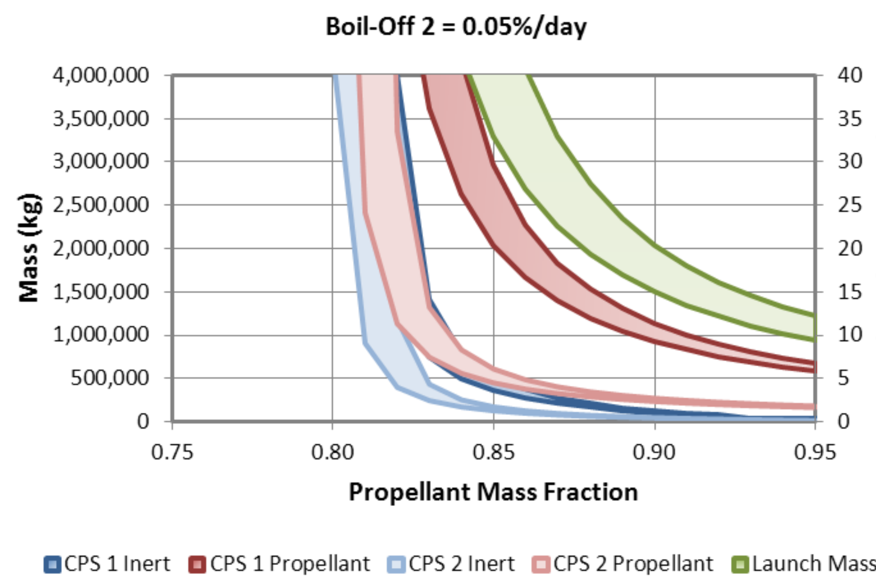
- Results for all combinations of design variables with Boil-Off Rate for CPS 2 = 0.025% shown above

Mars Mission: Design Variable Sweep Results (1 of 3)

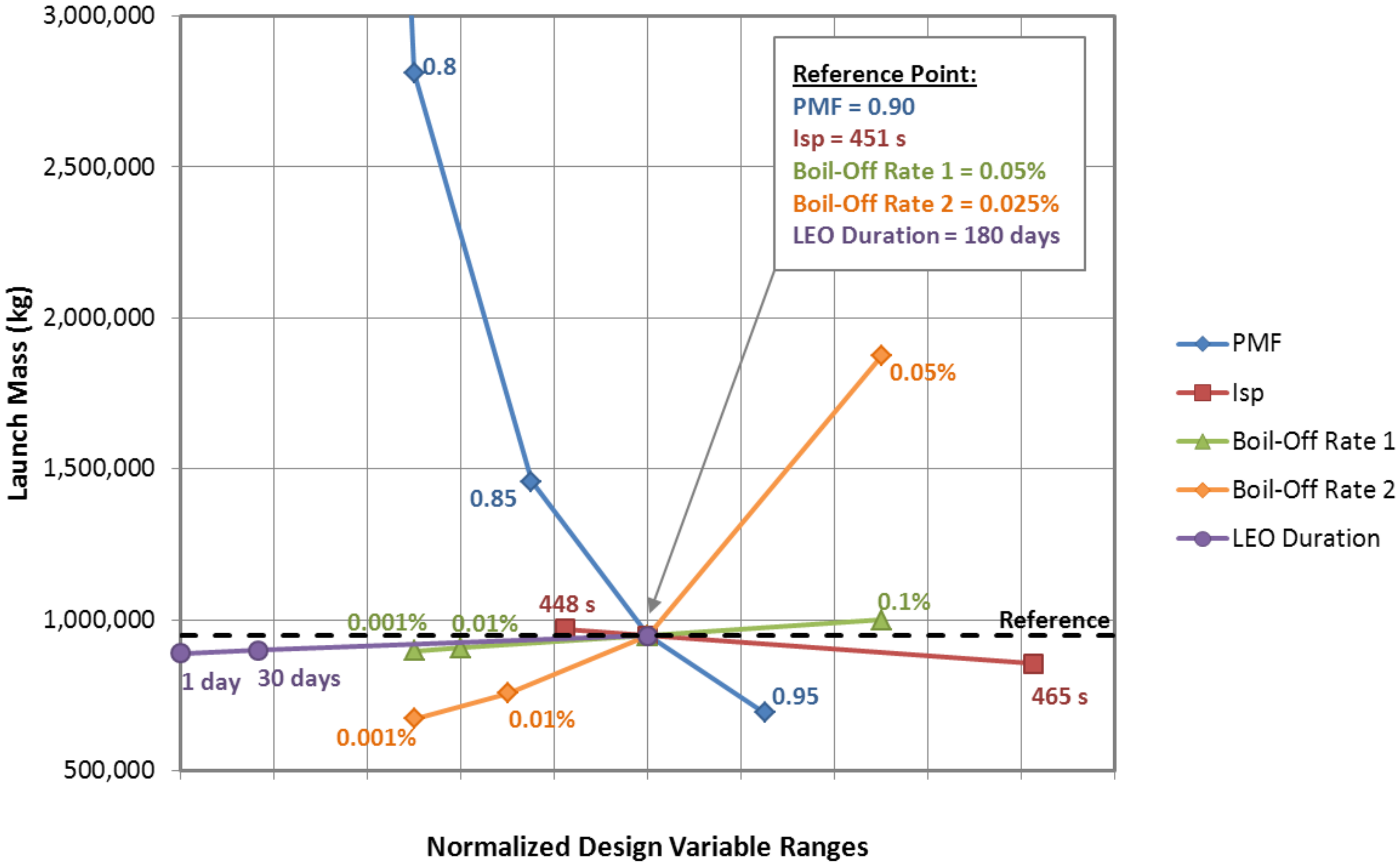






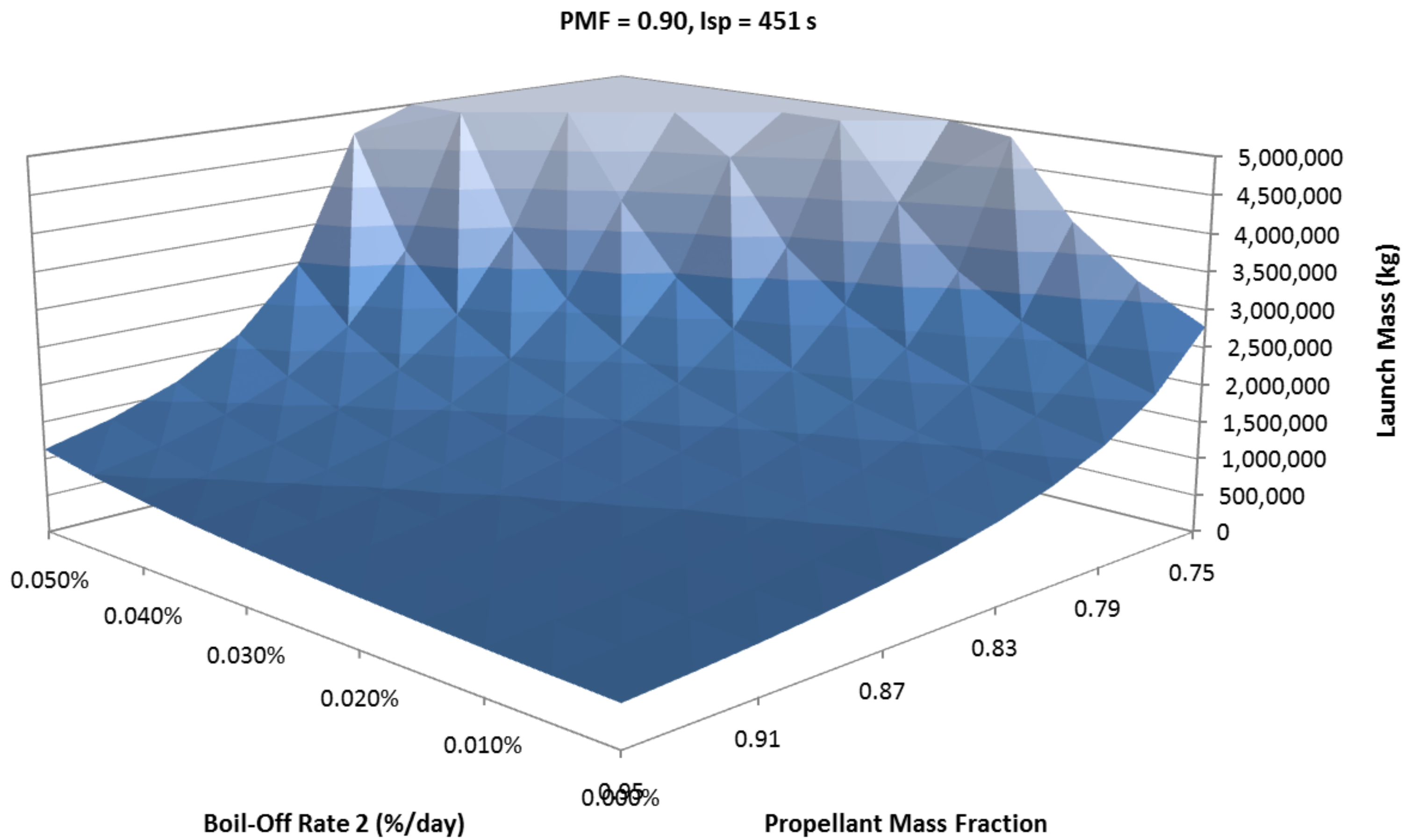


Mars Mission: Design Variable Sweep Results (2 of 3)



- Propellant Mass Fraction has the largest impact on launch mass – with a low enough PMF the vehicle will not close
- Because of the long mission times, Boil-Off Rate for CPS 2 is also a significant design driver
- Specific impulse, Boil-Off Rate for CPS 1, and LEO Duration all have small impacts on the design

Mars Mission: Design Variable Sweep Results (3 of 3)

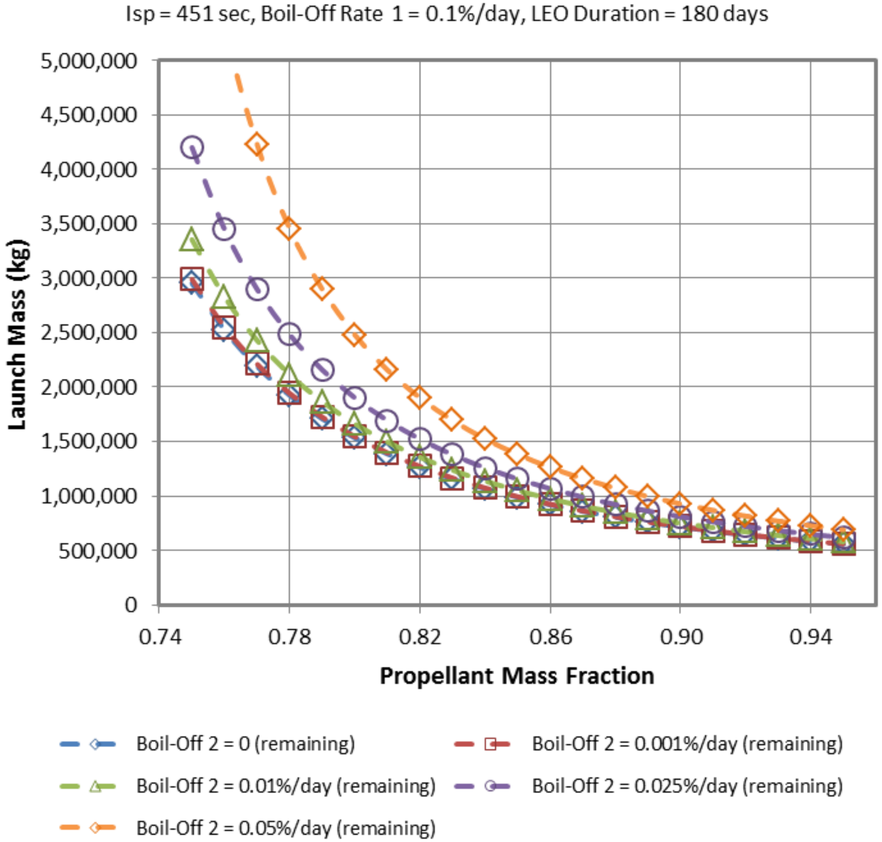


- Strong coupling between Boil-Off Rate for CPS 2 and Propellant Mass Fraction
- Conservative values for both variables can quickly lead to unclose-able cases

Mars Mission: Design Variable Coupling – Boil-Off 2 and PMF

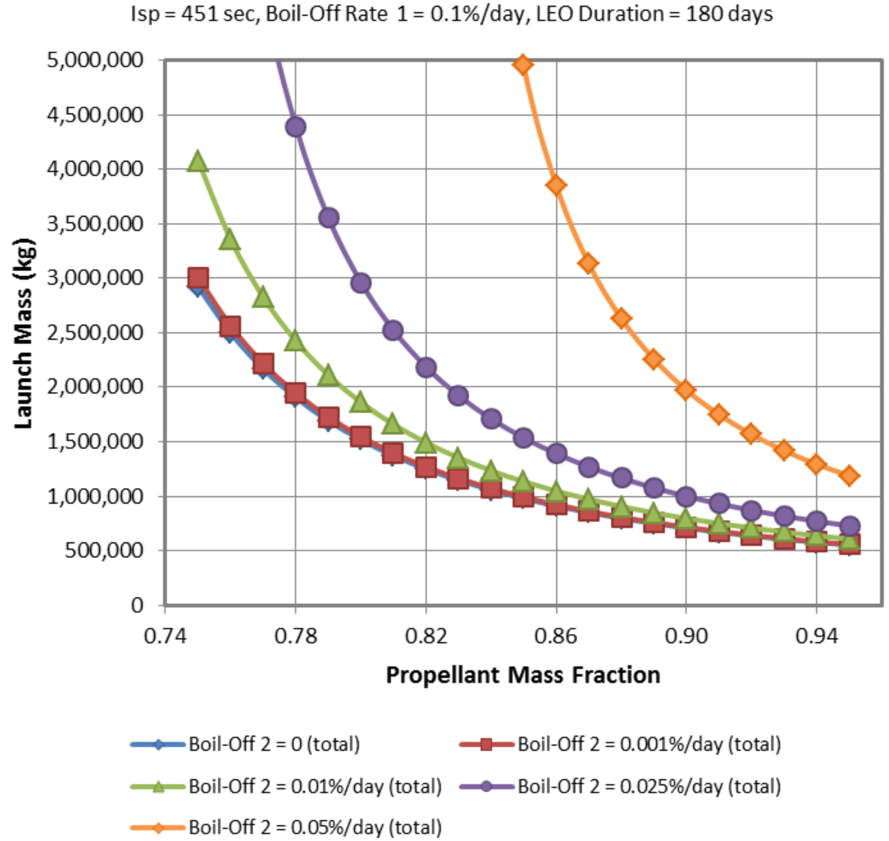
Remaining Method

Calculate boil-off losses based on current remaining propellant mass



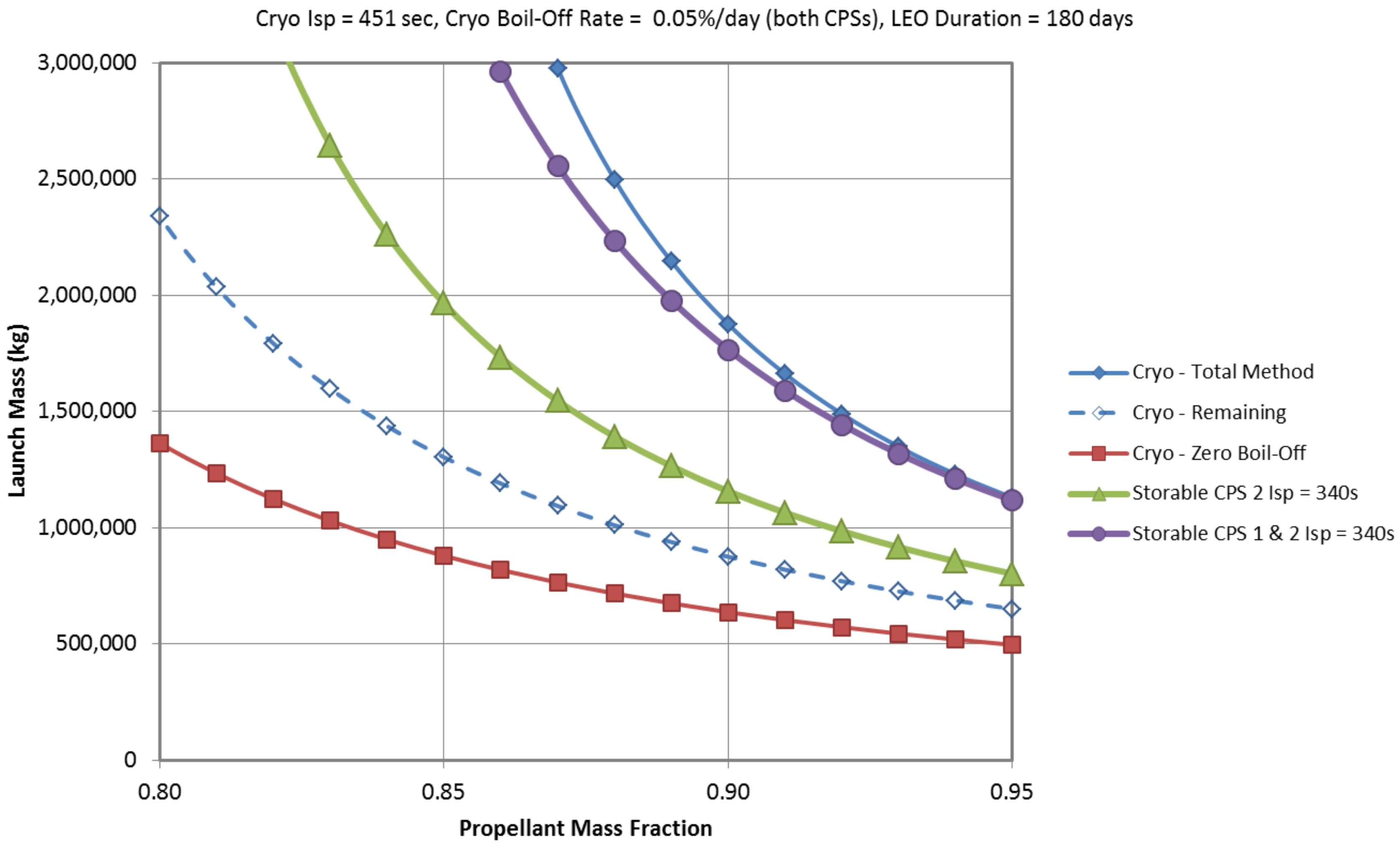
Total Method

Calculate boil-off losses based on total propellant mass at start of mission



- Comparison of two methods for calculating impact of boil-off
- For Mars mission with 550 days in Mars orbit with a partially full CPS 2, there is a significant difference between these two methods

Mars Mission: Boil-Off Method Comparison

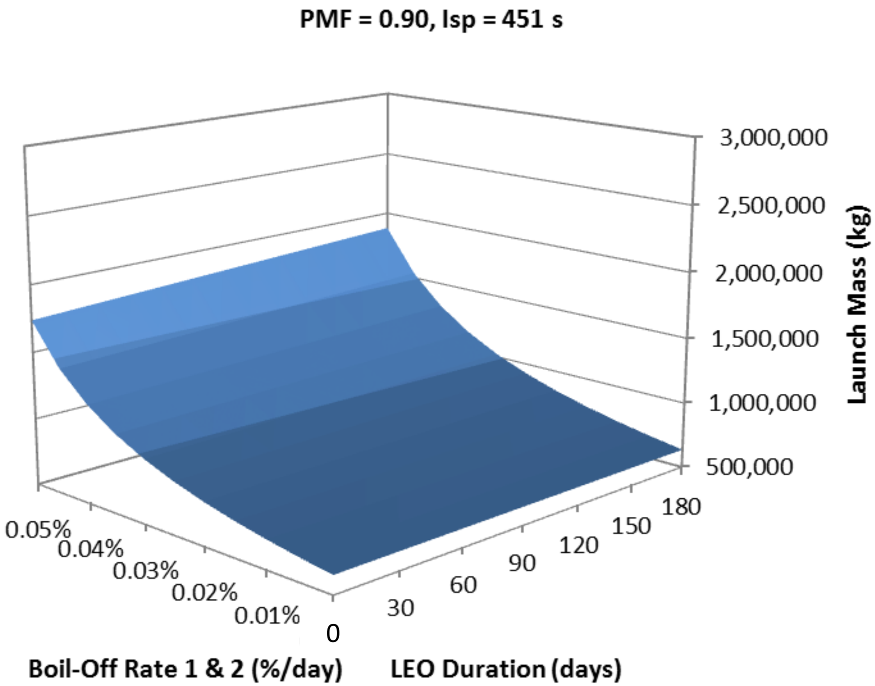


- Using storable propellants with a lower specific impulse can significantly reduce launch mass over cryogenic propellants with a higher specific impulse and high boil-off rate
- If low boil-off cannot be achieved for the second stage of a Mars mission, storable propellants are a viable alternative

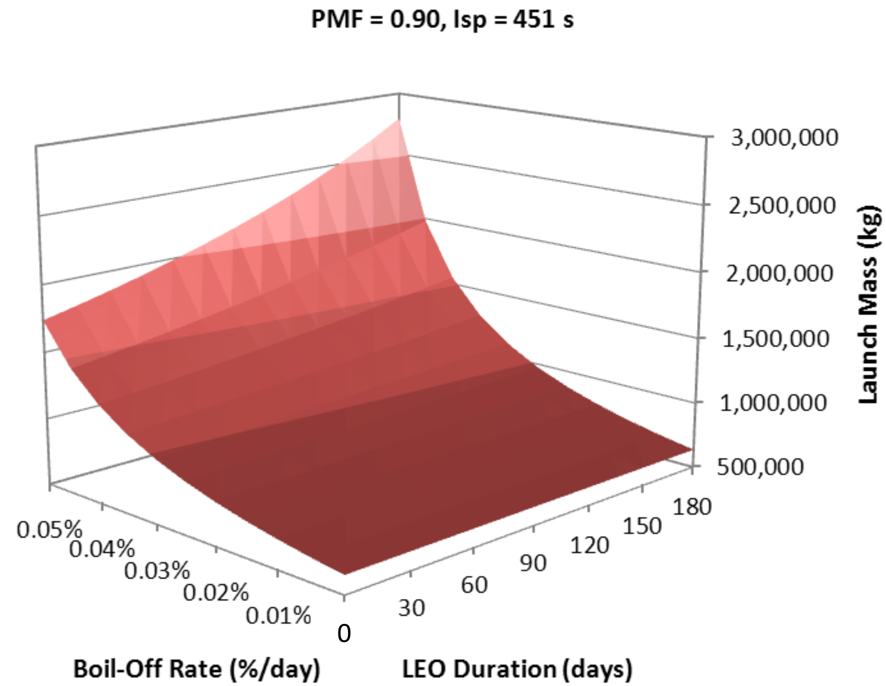
Mars Mission: Comparison of Cryogenic and Storable Propellants

Mars Mission: LEO Refuel Assumption Comparison

CPS Refueled Prior to Departure



No Refueling after Launch



- Coupling between Boil-Off Rate and LEO Duration less clear with multiple stages and longer mission times
- LEO Refueling Can Save up to ~35% in Launch Mass

OBSERVATIONS AND CONCLUSIONS

- Propellant Mass Fraction is the single largest design driver based on the range of values chosen for the design variables for all missions

- Specific Impulse has a much smaller impact than Propellant Mass Fraction but can still be a significant driver

- Boil-Off Rate has varying impact based on the mission requirements
  - For short, single stage missions (Earth-Moon L1 and LLO), boil-off rate has little impact on the system
  - For long, multi-stage missions (NEO and Mars), the boil-off rate on the second CPS stage can have a large impact on the mission as mission duration increases
  - The impact of boil-off rate is coupled with mission durations and the LEO Duration

- LEO Duration has a small impact on the total launch mass because the propellant lost to boil-off during this phase does not impact the size of the CPS stages

- The ability to refuel the CPSs in orbit can greatly reduce the total launch mass, even for short missions to L1 or LLO

- Calculating boil-off rate as either a percentage of current remaining propellant or total propellant at the beginning of the mission has a large impact on total launch mass for long duration missions with high boil-off rates

Observations: Design Variable, Assumptions, and Methodology Impacts

- The Earth-Moon L1 and LLO missions are very similar in time of flight and required Delta-V, so they show very similar results
    - Propellant Mass Fraction is the dominant design variable for these missions
    - Boil-Off Rate is only a factor over the LEO Duration time because of the short time of flight
    - At high boil-off rates, the ability to refuel the stage before launch can have a significant impact on the total launch mass, even for short missions

- The NEO Encounter mission introduces the second stage
    - Propellant Mass Fraction is still the dominant design variable, even at high boil-off rates
    - Boil-Off Rate on the second stage is a larger driver, but the total mission times are still under 180 days for this mission
    - Calculating boil-off rate based on current remaining propellant or total initial propellant does not have a significant impact on this mission design because CPS 2 is only partially full for 30 days

- The Mars Orbit mission introduces very long mission times
    - With the long mission times (750 days maximum outbound + stay time), Boil-Off Rates become a significant driver along with Propellant Mass Fraction
    - A combination of high Boil-Off Rates and low Propellant Mass Fractions can cause the vehicle to quickly grow and become unclose-able
    - Calculating boil-off rate based on current remaining propellant or total initial propellant has a significant impact on this mission design because CPS 2 is partially full for 550 days

Conclusions: Mission Conclusions

1. Invest in technologies that lead to **high propellant mass fraction** stages. These technologies will significantly benefit all four future human exploration mission categories.

2. Invest in technologies that **reduce propellant boil-off rates** to minimize propellant losses. This is particularly important for missions with long travel and stay times such as NEO and Mars missions.

3. Invest in on-orbit refueling technologies that **enable a refuelable CPS** in LEO. This will mitigate the impact of propellant losses for missions with long wait times before Earth departure.

These technology investments are interdependent.
Investment in one technology area may reduce the investment required in other areas for a particular mission.

Recommendations: Technology Investment

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SpaceWorks Enterprises, Inc.

SpaceWorks Enterprises, Inc. | 1040 Crown Pointe Parkway | Suite 950 | Atlanta, GA 30338 | 1+770-379-8000 (Office) | 1+ 770-379-8001 (Fax) | www.sei.aero | info@sei.aero

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