A MARS HEAVY TRANSPORT ARCHITECTURE
Increasing current payload capability to Mars with existing technology

by
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ABSTRACT

Payload mass delivered to Mars orbit using currently available launchers can be substantially increased by making use of low thrust (ion drive) propulsion for orbit raising around Earth. Trans-Mars injection is performed after orbit raise with the initial launcher's upper stage, modified to enable active cooling of cryogenic propellants. Payload masses at destination are calculated for fast (180 days) transfers from the Earth Departure Orbit, aerocaptured into an elliptical Mars orbit, or aerobraked and landed onto the martian surface. Only existing launchers, commercially available in 2008, and only existing technology are considered. Resulting masses in Martian orbit are calculated for transfers from 2008 up to 2039 using the Delta-IV, Ariane V, or Proton launchers. It is demonstrated that using an orbital space tug around Earth can dramatically increase the mass of payload delivered to Mars to within manned mission capability.

ABBREVIATIONS USED

LEO	Low Earth Orbit
EDO	Earth Departure Orbit
TMI	Trans-Mars Injection
ISS	International Space Station
MOC	Mars Orbit Capture
GCR	Galactic Cosmic Rays
OTT	Orbital Transfer Tug
EDL	Entry Descent and Landing
RCS	Reaction Control System
TMS	Trans-Mars Stack
EOL	End Of Life
SMA	Standard Mars Aeroshell
TPS	Thermal Protection System
RCS	Reaction Control System
TOF	Time of Flight
EDS	Earth Departure Stage

INTRODUCTION

This paper was written to further the cause of manned missions to Mars. Its aim is to increase awareness of the possibilities generated by combining technologies from different domains of existing space engineering, such as deep space probes and manned lunar missions. These existing technologies may render feasible manned missions to Mars within a short timeline without the need to develop new launchers or radically new propulsion systems. Though these latter, future technologies may be necessary to increase the efficiency of later missions to Mars, they currently contribute to the general skepticism about sending humans to Mars because of their uncertainties and high development costs. It is therefore in the interest of both the Mars Project and also the development of future technologies, to demonstrate the feasibility of going to Mars now, with only existing technologies.

The Trans-Mars Injection (TMI) "throw capacity" of currently available launchers is insufficient for planning a manned mission to Mars using a direct launch:

	Delta-IV Heavy	Ariane V	Proton / cryo[i]	Proton/ Briz[ii]	
Payload after TMI:	8,000[iii]	5,900[iv]	5’800[v]	4,580[vi]	kg

Table 1. Current launchers TMI throw capacity (C3 = 10 km2/s2)

Recent developments in ion-drive propulsion[vii], high-capacity solar arrays[viii], and active cooling of cryogenic propellants[ix], now allow us to consider substantial increases in useful payload to Mars orbit by using a solar-powered, ion-driven space tug to lift the Trans Mars Stack (TMS) from Low Earth Orbit (LEO) to a highly elliptical Earth Departure Orbit (EDO), thus substantially reducing the TMI burn. The massively increased payload capacity to Mars allows us to consider transferring habitable modules with a time of flight of 6 months, thereby potentially enabling manned missions to Mars using existing launchers:

	Delta-IV Heavy	Ariane V	Proton / cryo	Proton / Briz	
Payload after TMI:	17,600[x]	16,400	16’000	14,600	kg

Table 2. Current launchers TMI capacity using a space tug from LEO to EDO (C3 = 10 km2/s2)

Aiming for the 2018 favorable Earth-Mars conjunction (next would be 2031), it is decided that only *existing technology* will be considered in this study. In this context, the term is defined as englobing technology which has already been used (such as ion-drives) but also that which has been tested but not yet applied in the field (such as active propellant cooling) or is already funded and currently under development, with clearly predictable results and a short-term timeline (such as Stretched Lens Array Squarerigger technology). In the conclusion, a suggestion will be made for developing a new hypersonic braking scheme (ie new technology) for landing large (manned-mission sized) payloads on the surface.

MISSION SUMMARY

1. Payload is launched from Earth on an existing heavy launcher to a 400x400 km orbit (LEO).

2. The upper stage remains attached to the payload and contains sufficient propellant for the future TMI burn from EDO. It has been modified to enable active cooling of its propellant during the months-long orbit raising maneuver.

3. The ion-driven Orbital Transfer Tug (OTT) makes rendezvous with the TMS in LEO and connects to the forward end of the payload module.

4. The OTT lifts the TMS to EDO (a 400x300'000 km orbit) during a slow, months-long maneuver with multiple burns at periapse. It also propels the TMS for the plane change at apoapse from launch inclination to ecliptic (0.12 km/s for the Delta IV from Cape Canaveral, 0 km/s for the Ariane V from Kourou , 0.23 km/s for the Proton from Baikonour).

5. If the mission is to be manned, a crew module is launched from Earth on a direct (3-day) transfer to EDO, where it docks with the TMS for crew transfer before the TMI burn.

6. In EDO, the OTT detaches itself from the TMS and lowers its orbit sequentially by multiple high-atmosphere passes at periapse (Mars Global Surveyor style[xi]) in order to return to LEO for the next payload stack.

7. The upper stage fires its engines a second time at EDO to inject the TMS onto a trans-Mars trajectory.

8. Once the OTT is back in LEO, it replaces its jettisoneable propellant tank with a new one delivered to LEO by a medium-capacity launcher (7.1 tons to LEO), and waits for the next payload to lift to EDO.

9. Before the TMS reaches MOC, the upper stage is jettisoned. The SMA aerocaptures into a 250x33'793 km orbit around Mars (1 martian day period) or initiates a direct Entry Descent and Landing (EDL) to the martian surface.

ORBITAL TRANSFER TUG

The OTT is composed of two modules, a self-maneuverable, 7.1-ton, jettisoneable tank, and an 11.3-ton Propulsion Module. The propulsion module is propelled by 77 NEXT ion-drives[xii], for a total thrust of 28 N. Energy is delivered by 3,500 m2 of SLASR solar arrays[xiii], producing a total of 860 kW of electrical energy at End of Life (EOL). The OTT can lift a maximum 30.5-ton payload from LEO to EDO during a total propulsion duration of 73 days at 0.0006 m/s/s average acceleration. As the thrust is only applied at periapse, the actual trip duration is much longer, around 300 days.

The 7.1-ton jettisoneable tank is equipped with RCS and avionics such that it can maneuver to rendezvous with the propulsion module and, once empty, de-orbit itself for a controlled destructive entry into the Earth's atmosphere.

Because the OTT has a very low thrust and a very high surface area, it is strongly affected by air drag at low altitudes. At 250 km altitude, the air drag is around 28 N on its 3,500 m2 solar arrays, which is equivalent to its thrust. This is therefore the limiting altitude under which the OTT cannot maintain orbital velocity. At 400 km altitude the air drag is around 0.6N, or 2% of its thrust. The OTT only goes below 400 km altitude during the periapse passes of its orbital lowering phase.

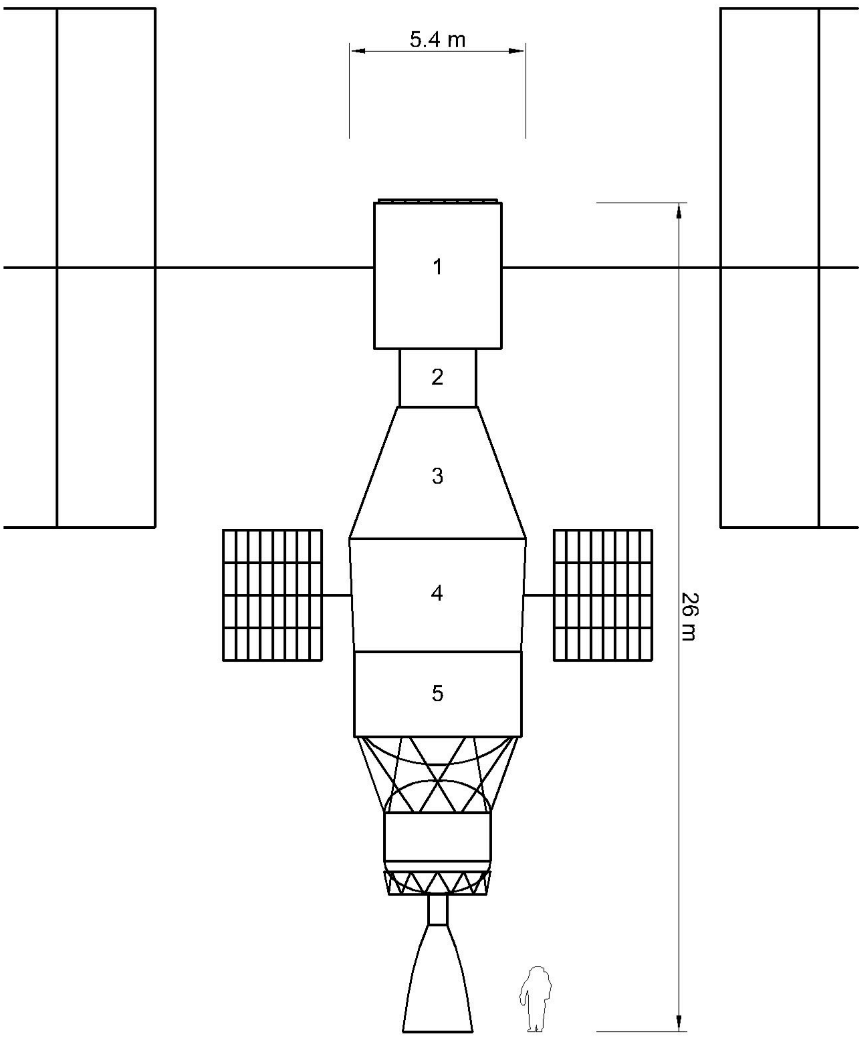

Fig 1. OTT and TMS assembled in LEO. From top to bottom:

1 - OTT propulsion module
2 - OTT jettisoneable tank
3 – SMA with payload
4 - Payload adapter with cryogenic cooling system and solar panels / radiators
5 - Upper stage (Delta IV shown here)

UPPER STAGE

The heavy launcher's upper stage has been modified to enable active cooling of its cryogenic propellants, except in the case of the Proton / Briz which uses storable propellants. Active cooling is necessary to reduce propellant loss via boil-off during the 300-day orbit raise by the OTT. The refrigeration system[xiv], solar arrays, batteries, and inter-stage adapter increase the upper stage's dry mass by some 22-32%.

TRANS-MARS PAYLOAD

The Standard Mars Aeroshell (SMA) will have a standard diameter of 5.4 meters, and a fixed structure and Thermal Protection System (TPS) across the whole range of launchers and launch dates, in order to optimize design, testing and production. It is an entry capsule with a 70° conical forward shield and spherical nosecap, as has been the tradition with previous Mars entry capsules since the Viking probe.

The capsule does a first aerocapture pass to slow itself into its 250 x 33'793 km parking orbit with entry velocities varying between 5.42 km/s (2035 mission) and 8.59 km/s (2028 mission). Aeroshell mass for a total entry mass of 18.0 tons (2018 highest mass scenario) and an entry velocity of 8.59 km/s (2028 worst case scenario) is calculated and shown at the end of this paper. The thermal protection system used is the traditional Mars atmospheric entry SLA-561 ablative.

MANNED MISSIONS

For manned missions the empty TMS is lifted to EDO like any other payload using the OTT. The crew, on the other hand, is transferred to EDO by an adequate crew transfer vehicle such as a Soyuz capsule (or an Orion capsule, a Dragon capsule, or a Shenzou module) propelled by an adequate, high-thrust booster, for a fast 3-day transfer, in order to avoid exposing the crew to zero-G and GCRs for more than the 6-month trans-Mars flight.

180-DAY TRANSFER OPPORTUNITIES

Lowest departure C3

Departure	400	x	300,000	km orbit
Arrival	250	x	33,793	km orbit
Departure gravity loss	5.0%			

Date	Departure C3 (km2/s2)	Arrival C3 (km2/s2)	Departure dV (km/s)	Arrival dV (km/s)	Entry V (km/s)	TOF (days)
17.03.2001	8.65	22.27	0.53	1.90	6.53	180.00
05.06.2003	9.20	9.00	0.56	0.84	5.47	180.00
09.08.2005	16.52	13.80	0.90	1.24	5.87	180.00
28.09.2007	20.80	26.50	1.09	2.21	6.84	180.00
28.10.2009	20.20	44.38	1.06	3.38	8.00	180.00
02.12.2011	15.91	49.85	0.87	3.70	8.33	180.00
06.01.2014	11.09	44.16	0.65	3.36	7.99	180.00
20.02.2016	8.88	28.45	0.55	2.34	6.97	180.00
06.05.2018	8.13	10.85	0.51	1.00	5.63	180.00
19.07.2020	13.63	10.85	0.77	1.00	5.63	180.00
08.09.2022	19.69	22.09	1.04	1.89	6.52	180.00
17.10.2024	21.57	40.90	1.12	3.16	7.79	180.00
17.11.2026	17.92	50.47	0.96	3.74	8.37	180.00
21.12.2028	13.50	54.37	0.76	3.96	8.59	180.00
31.01.2031	9.28	35.98	0.56	2.85	7.48	180.00
06.04.2033	8.42	14.92	0.52	1.34	5.97	180.00
25.06.2035	10.47	8.48	0.62	0.79	5.42	180.00
24.08.2037	17.76	15.47	0.95	1.38	6.01	180.00
03.10.2039	21.00	31.83	1.10	2.57	7.20	180.00

- Entry V for equatorial prograde orbits (0.236 km/s planetary rotation bonus wrt polar orbital entry included)

RESULTS

Gross Payload Mass to MOC (including aeroshell):

Date	Delta-IV H	Ariane V	Proton/cryo	Proton/Briz	
17.03.2001	17'896	16'731	16'307	14'922	kg
05.06.2003	17'766	16'599	16'193	14'775	kg
09.08.2005	16'128	14'942	14'769	12'955	kg
28.09.2007	15'245	14'050	14'001	11'998	kg
28.10.2009	15'366	14'172	14'106	12'128	kg
02.12.2011	16'258	15'073	14'882	13'098	kg
06.01.2014	17'327	16'154	15'811	14'282	kg
20.02.2016	17'841	16'676	16'259	14'861	kg
06.05.2018	18'020	16'856	16'414	15'063	kg
19.07.2020	16'755	15'575	15'314	13'645	kg
08.09.2022	15'469	14'276	14'196	12'239	kg
17.10.2024	15'092	13'895	13'868	11'833	kg
17.11.2026	15'834	14'644	14'513	12'634	kg
21.12.2028	16'783	15'604	15'339	13'677	kg
31.01.2031	17'747	16'580	16'177	14'754	kg
06.04.2033	17'950	16'786	16'354	14'984	kg
25.06.2035	17'469	16'299	15'936	14'442	kg
24.08.2037	15'867	14'678	14'542	12'670	kg
03.10.2039	15'205	14'009	13'966	11'955	kg

- Payload mass is reduced for additional air drag during launch due to increased payload diameter for Delta-IV (5.13m) and Proton (4.35m).
- Refrigeration system, solar panels and payload adapter subtracted from total mass.
- 100 m/s TMI mid-course corrective burn factored in.

Net Mass to MOC by aerobraking:

Date	Delta-IV H	Ariane V	Proton/cryo	Proton/Briz	
17.03.2001	14'297	13'179	12'771	11'442	kg
05.06.2003	14'286	13'158	12'766	11'396	kg
09.08.2005	12'664	11'520	11'354	9'605	kg
28.09.2007	11'724	10'579	10'532	8'613	kg
28.10.2009	11'733	10'597	10'535	8'653	kg
02.12.2011	12'551	11'426	11'245	9'550	kg
06.01.2014	13'600	12'484	12'158	10'703	kg
20.02.2016	14'197	13'082	12'683	11'344	kg
06.05.2018	14'514	13'391	12'964	11'659	kg
19.07.2020	13'292	12'154	11'902	10'290	kg
08.09.2022	11'969	10'823	10'746	8'867	kg
17.10.2024	11'492	10'352	10'326	8'388	kg
17.11.2026	12'144	11'015	10'891	9'107	kg
21.12.2028	13'024	11'906	11'655	10'079	kg
31.01.2031	14'054	12'940	12'556	11'197	kg
06.04.2033	14'410	13'289	12'873	11'553	kg
25.06.2035	14'004	12'873	12'521	11'077	kg
24.08.2037	12'399	11'254	11'123	9'320	kg
03.10.2039	11'653	10'510	10'468	8'545	kg

- Assumes using a 2'884 kg SMA (Standard Mars Aeroshell) for aerobraking at Mars (see section 12 for details).
  - Aeroshell, RCS, RCS propellant and avionics subtracted from total mass
  - RCS propellant varies between 480 and 875 kg depending on total entry mass.

DISCUSSION OF RESULTS

Modifying existing launchers' upper stages and using continuous low thrust for orbit raising around Earth may multiply payloads sent to Mars by as much as three times. Final net payloads in Martian orbit may as high as 14.5 tons.

Net payload mass to Mars orbit (excluding the aeroshell) can vary from

- 8.4 to 11.6 tons using the Proton/Briz launcher
- 10.3 to 13.0 tons using the Proton with a new cryogenic upper stage
- 10.3 to 13.4 tons using the Ariane V launcher
- 11.5 to 14.5 tons using the Delta IV Heavy

These masses are comparable to existing habitable modules of the ISS such as Columbus (10.3 tons), Destiny (14.1 tons), Kibo (14.8 tons) or Svezda (19.1 tons). Note these latter masses include structural, insulation and power components, parts of which are accounted for in the SMA.

Assuming development of a low-mass, high-atmosphere braking scheme, net payload masses to the martian surface (excluding lander) can vary from:

- 3.5 to 6.4 tons using the Proton/Briz launcher
- 5.2 to 7.5 tons using the Proton with a new cryogenic upper stage
- 5.2 to 6.8 tons using the Ariane V launcher
- 6.2 to 8.9 tons using the Delta IV Heavy

For comparison's sake, note the dry mass of the Apollo LM ascent stage was 2.2 tons (including astronauts), the lunar rover massed 0.2 tons, a pressurized 4.5 x 6.7m module (about 80 m3 inner volume) such as the ISS Multi-Purpose Logistics Module masses 4.1 tons, and the airlock section of the quest module masses 0.9 tons.

Further Research

<u>A Hypersonic Braking System</u>

Because the Ballistic Coefficient of the SMA is much too high to enable effective slowdown in the martian atmosphere to landing-gear deployment velocities, a hypersonic braking method must be developed if we wish to land large payloads on the surface. This currently goes beyond existing (Viking legacy) technology and therefore beyond the scope of this paper. At present, the most promising hypersonic braking scheme in terms of minimizing additional mass constraints is the hypersonic parachute[xv]. This could be made from existing high-resistance, high-temperature carbon cloth and would be the object of a new area of research.

<u>Increasing Payload to MOC</u>

Another scheme may be considered whereby a 22-ton payload (maximum Delta-IV Heavy payload to LEO) is launched to LEO where it is mated with a 22-ton Earth Departure Stage (EDS). This improvement was suggested by R. Zubrin. Both are then lifted to EDO using the OTT equipped with two jettisoneable propellant tanks. Once the stack is at EDO the crew module is launched from Earth on a direct transfer to EDO and docks with the stack. The EDS propels the stack to Mars and also captures it into Mars orbit. 570 days later it propels the same stack back to Earth. This scheme has the disadvantage of relying on an extra orbital docking maneuver but payload to Mars is increased from 18 to 22 tons and atmospheric entry risk is eliminated by using propulsive capture at Mars.

<u>Key Items to be Designed</u>

To further the goal of landing humans on Mars using existing launchers and returning them safely to Earth, the following elements must be further researched and designed:

1. Ion-driven Space Tug for lifting 27 tons from 400x400 to 400x300,000 km orbit
2. Upper stage modifications to enable cryogenic cooling of propellants
3. Man-rating of an existing heavy launcher for direct transfer to EDO of a crew module
4. < 22-ton Space habitat (ideally < 18 tons)
5. 5.4m Mars entry capsule
6. Carbon-fiber based Hypersonic Parachute
7. < 22-ton Descent-Ascent vehicle with in-situ propellant production, fitting in a 5.4m aeroshell (ideally < 18 tons)
8. < 10-ton Surface habitat (ideally < 8 tons)
9. 3-4 person surface pressurized rover
10. In Situ Propellant Production system capable of producing and liquifying 40 kg of LOX/LH per day

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APPENDIX: VEHICLE SPECIFICATIONS AND MASS BREAKDOWNS

Orbital Transfer Tug

DESIGN PARAMETERS

Engine model	NEXT	
Fuel/oxidizer type	Xe	
isp	3'640	s
Power requirements	866'000	W
Engine thrust-to-weight	0.00751	N/kg
Tank ullage factor	3	% volume
Residual propellant	2	% mass
Active cooling	yes	
Max payload mass	30'500	kg
Thrust	28	N
Required Delta-v	3'670	m/s
Required Delta-v margin	3	%

MASS BREAKDOWN - PROPULSION MODULE

Solar arrays (SLASR)	5'732	kg
Batteries (Li-ion)	4	kg
Engine mass	3'696	kg
RCS	205	kg
Avionics + auxilliary power	256	kg
Structure	330	kg
15% contingency	1'072	kg
Total mass	11'295	kg
Total tank + propulsion	18'393	kg

MASS BREAKDOWN - JETTISONEABLE TANK

Total propellant mass	5'010	kg
Propellant tank mass	532	kg
Pressurant and tank	50	kg
Payload adapter	305	kg
Refrigeration system	22	kg
Solar arrays (SLASR)	8	kg
RCS	129	kg
Avionics + auxilliary power	161	kg
Structure	208	kg
15% contingency	674	kg
Total mass	7'098	kg

Standard Mars Aeroshell

DESIGN PARAMETERS

Heat-shield diameter:	5.4	m
Max entry mass:	18'020	kg
Max entry velocity:	8.59	km/s
L/D ratio:	0.30	
Entry angle:	1.60	°
TPS type:	SLA-561	
Heat of ablation:	54	MJ/kg
Total heat-load:	42'974	J/cm2
Peak heat load:	96	W/cm2
Peak temperature:	1'752	°C
Peak deceleration:	4.99	Gs

MASS BREAKDOWN

Hypersonic parachute:	60	kg
Ablating material:	160	kg
Steel substructure:	604	kg
Insulation:	101	kg
Honeycomb aluminum shell:	1'153	kg
RCS:	90	kg
RCS propellant:	-	(varies with entry mass and delta-v)
Avionics:	200	kg
Power system:	200	kg
15% contingency:	376	kg
Total dry aeroshell mass:	2'944	kg

Modified Upper Stages

	Delta-IV H	Ariane V	Proton/cryo	Proton/Briz	
Upper stage isp:	462	446	462	326	s
Upper stage wet mass:	30'710	19'440	22'170	22'170	kg
Upper stage dry mass:	3'490	4'540	2'370	2'370	kg
Residual propellant (2%):	544	298	396	396	kg
Regular diameter:	5.13	5.40	4.35	4.35	m/s
Dv penalty for 5.4m diameter:	66	-	330	330	m/s
Regular payload to 407x407 km:	22'560	21'000	21'600	21'600	kg
Payload penalty for 5.4m diam.:	326	-	1'517	2'118	kg
New payload to 407x407 km:	22'234	21'000	20'083	19'482	kg
Mass to 407x407 incl. stage:	26'269	25'838	22'849	22'248	kg

MODIFIED UPPER STAGE MASS BREAKDOWN

Regular dry mass:	3'490	4'540	2'370	2'370	kg
Refrigeration system:	283	189	229	0	kg
Payload adapter:	589	557	532	516	kg
New dry mass:	4'362	5'286	3'131	2’886	kg

Possible Lander

DESIGN PARAMETERS

Max total mass:	14'914 kg	includes avionics and power system shared with aeroshell
Delta-v:	0.632 km/s	
Engine type:	LOX/LH	
Mixture ratio:	5.85 LOX/LH	
Tank ullage factor	3 % volume	
Residual propellant	2 % mass	
Engine isp:	462 s	
Max thrust:	110'864 N	
Parachute:	158 kg	
Engine mass:	305 kg	
Structure mass:	597 kg	
Propellant mass:	1'980 kg	includes 2% residual
LH tank mass:	241 kg	
LOX tank mass:	86 kg	
Landing system:	1'766 kg	
Avionics:	200 kg	included in aeroshell mass
Power system:	200 kg	included in aeroshell mass
15% contingency:	533 kg	
Max payload:	8'850 kg	
Dry mass:	4'084 kg	

FOOTNOTES

[i] Assumes developing a new cryogenic upper stage for the Proton, with same dry mass as the Briz-M.
[ii] Using the 327s isp Briz-M upper stage
[iii] Delta IV Heavy Demo brochure
[iv] Derived from GTO payload data
[v] Derived from Briz-M payload data
[vi] Proton Launch System Mission Planner's Guide, page 2-23
[vii] Oleson et al. 2002
[viii] Mark O'Neil et al. 2007
[ix] Plachta 1998
[x] Additional air drag due to increased payload fairing to 5.4m diameter factored in
[xi] Johnston et al. 1998
[xii] Oleson et al. 2002
[xiii] Mark O'Neil et al. 2007
[xiv] Borowski et al. 2002
[xv] Bolonkin 2006