Incentivizing LEO Debris Removal with Electrodynamic Tethers

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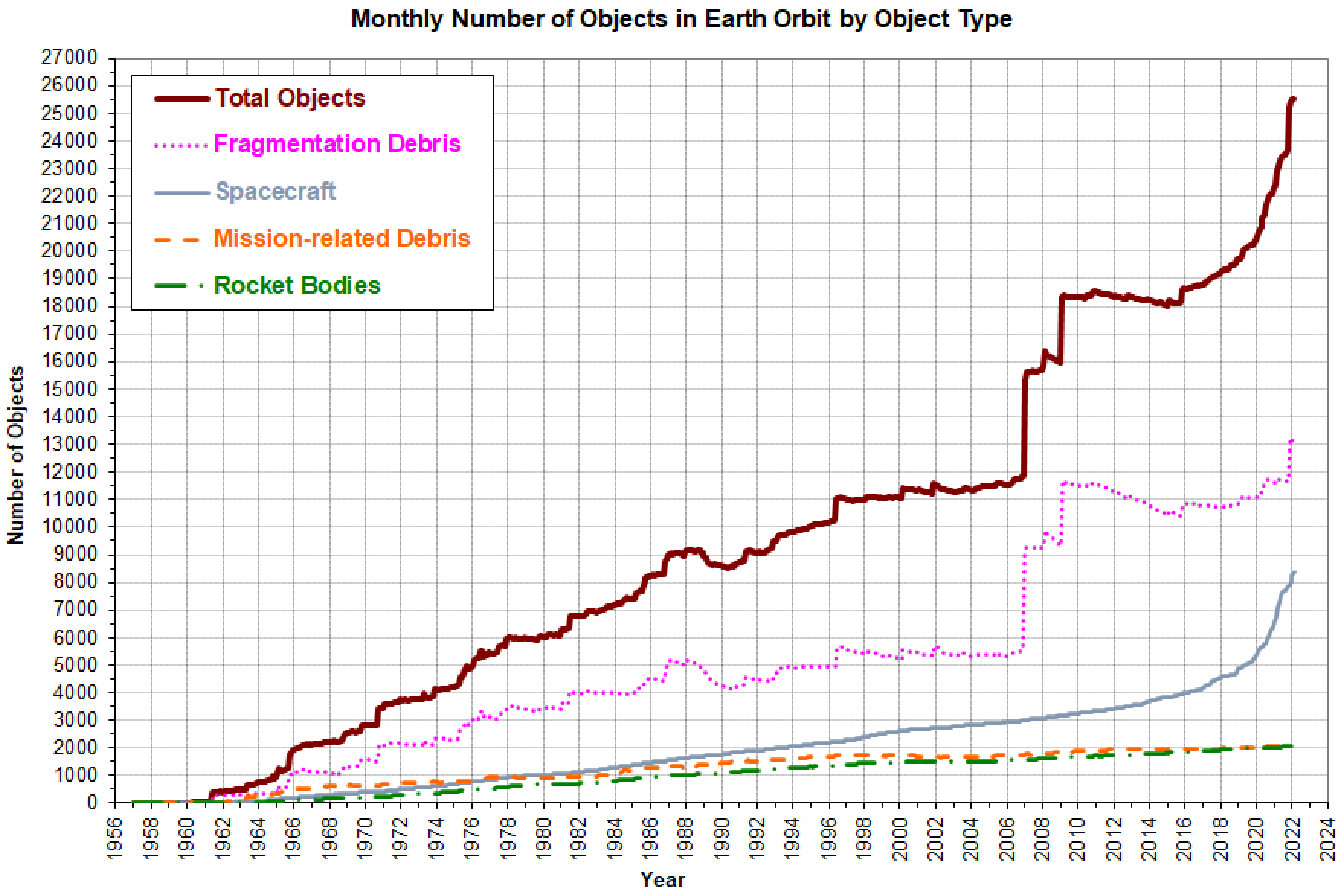
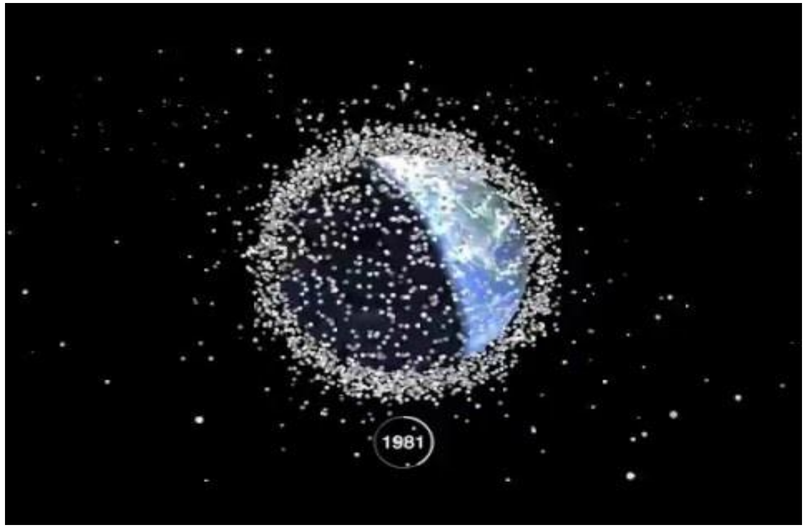
Paper No: 2024081

Houston, we have a problem…





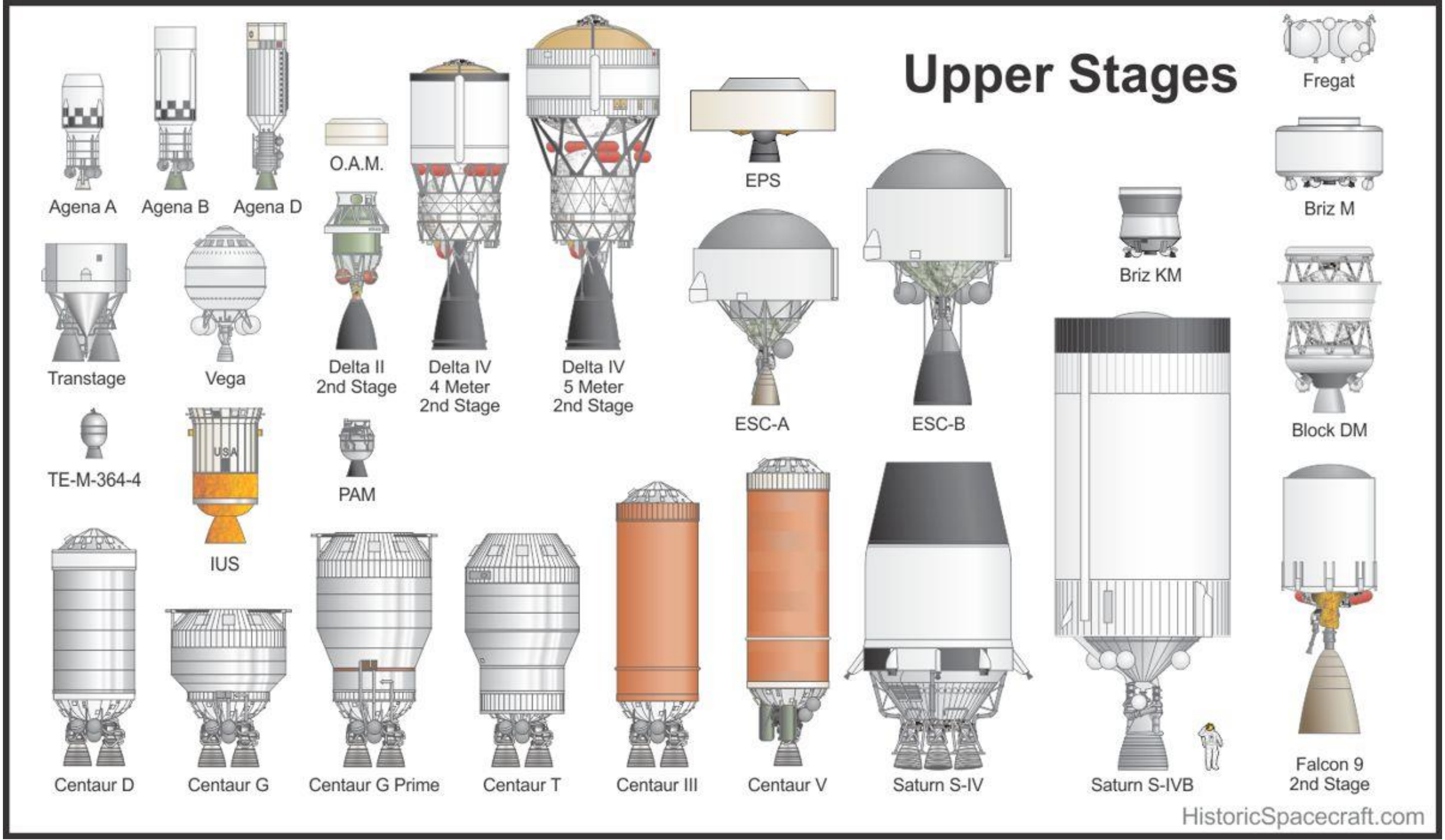
https://orbitaldebris.jsc.nasa.gov/modeling/legend.html

Ways to mitigate orbital debris

- Proactive
  - Mandate de-orbit at end-of-life (e.g., 5- or 25-year directive) for new systems

- Active
  - Active Debris Removal (ADR) systems

- Removing some of the largest debris could have a big effect on mitigating Kessler Syndrome



Upper Stages
Agena A
Agena B
Agena D
O.A.M.
Delta II 2nd Stage
Delta IV 4 Meter 2nd Stage
Delta IV 5 Meter 2nd Stage
EPS
ESC-A
ESC-B
Fregat
Briz M
Briz KM
Block DM
Transtage
Vega
TE-M-364-4
IUS
USA
PAM
Saturn S-IVB
Falcon 9 2nd Stage
Centaur D
Centaur G
Centaur G Prime
Centaur T
Centaur III
Centaur V
Saturn S-IV
HistoricSpacecraft.com

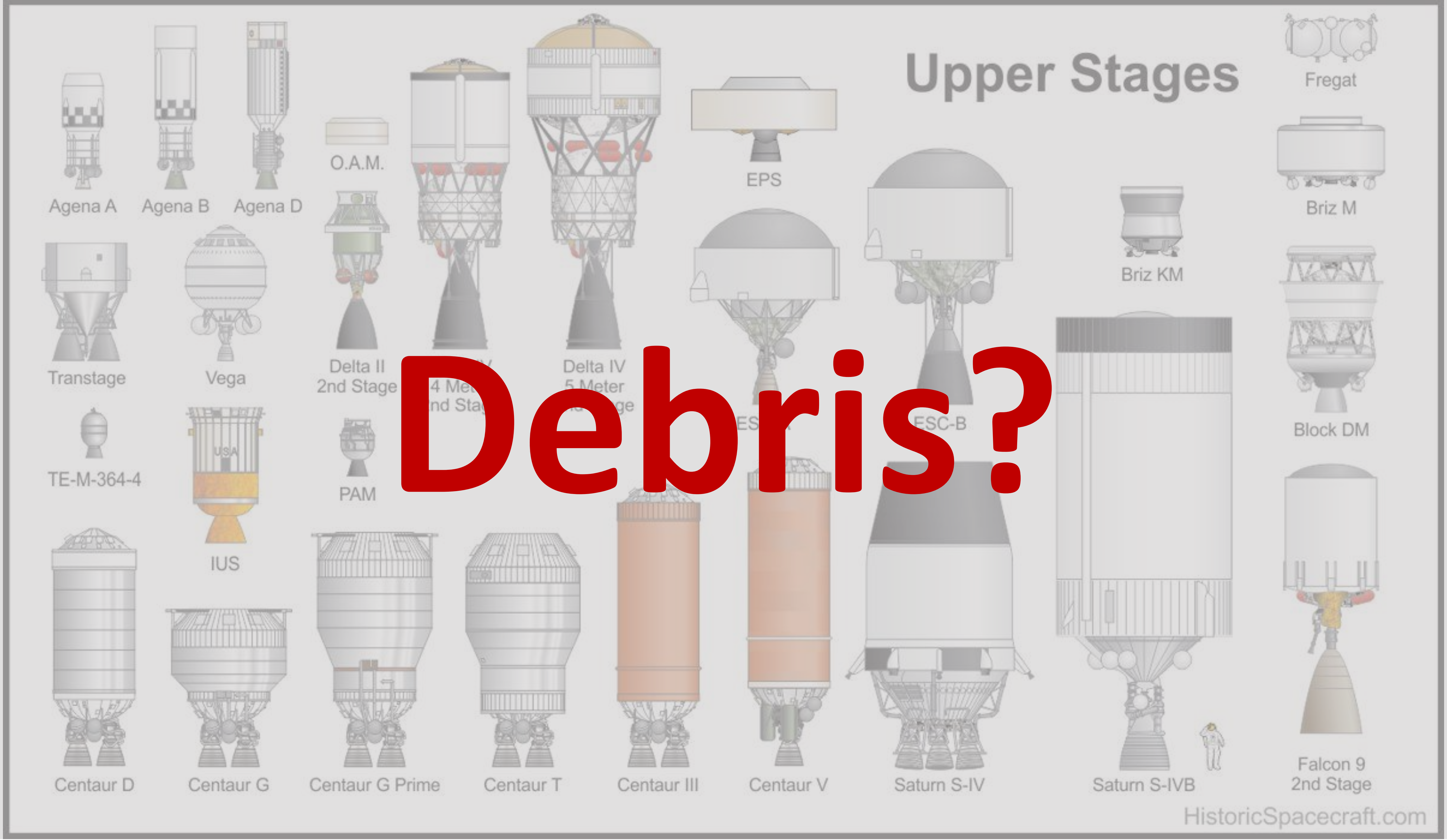
Statistically Most Concerning (SMC) debris in LEO

- Started with list of top 50 SMC objects
  - These were selected using 11 different methods from list of non-operational LEO objects available on Space-track.org [1]
- Important to note:
  - 84% of objects come from CIS (Russia/Soviet)
  - 8% from Japan
  - 6% from ESA
  - 2% from China

SATNAME	APOGEE, km	PERIGEE, km	INCL., deg	MASS, kg	COUNTRY	LAUNCH
SL-16 R/B	848	837	71.0	9000	CIS	3/26/1993
SL-16 R/B	848	827	71.0	9000	CIS	11/17/1992
SL-16 R/B	846	843	71.0	9000	CIS	6/29/2007
SL-16 R/B	854	827	71.0	9000	CIS	March 2, 2000
SL-16 R/B	844	833	71.0	9000	CIS	10/22/1985
SL-16 R/B	853	834	71.0	9000	CIS	5/22/1990
SL-16 R/B	1006	986	99.5	9000	CIS	October 12, 2001
SL-16 R/B	852	831	71.0	9000	CIS	10/31/1995
SL-16 R/B	844	835	71.0	9000	CIS	7/28/1998
SL-16 R/B	845	838	71.0	9000	CIS	11/24/1994
SL-16 R/B	846	823	71.0	9000	CIS	5/13/1987
SL-16 R/B	845	841	71.0	9000	CIS	4/23/1994
SL-16 R/B	844	840	71.0	9000	CIS	12/25/1992
SL-16 R/B	850	823	71.0	9000	CIS	9/16/1993
SL-16 R/B	848	831	71	9000	CIS	11/23/1988
SL-16 R/B	863	839	70.8	9000	CIS	April 9, 1996
SL-16 R/B	848	842	71.0	9000	CIS	October 6, 2004
SL-16 R/B	841	831	71.0	9000	CIS	3/18/1987
SL-16 R/B	842	814	71.0	9000	CIS	5/15/1988
SL-16 R/B	813	801	98.6	9000	CIS	October 7, 1998
ENVISAT	766	764	98.1	7800	ESA	January 3, 2002
METEOR 3 M	1013	994	99.6	2500	CIS	October 12, 2001
ADEOS	794	793	98.9	3560	JPN	8/17/1996
H-2A R/B	836	734	98.2	3000	JPN	12/14/2002
SL-12 R/B(2)	847	838	71.0	2440	CIS	9/28/1984
CZ-2D R/B	846	791	98.7	4000	PRC	11/20/2011
SL-8 R/B	995	966	82.9	1435	CIS	3/15/1978
H-2 R/B	1306	860	98.7	2700	JPN	8/17/1996
COSMOS 2322	854	842	71.0	3250	CIS	10/31/1995
SL-8 R/B	992	961	82.9	1435	CIS	May 2, 1991
COSMOS 2406	863	844	71.0	3250	CIS	October 6, 2004
COSMOS 2278	852	841	71.1	3250	CIS	4/23/1994
COSMOS 1943	851	833	71.0	3250	CIS	5/15/1988
ADEOS 2	801	800	98.5	3680	JPN	12/14/2002
SL-16 R/B	645	622	98.2	9000	CIS	7/17/1999
SL-12 R/B(2)	848	794	71.1	2440	CIS	5/30/1985
SL-8 R/B	989	957	83.0	1435	CIS	2/28/1978
COSMOS 1844	866	824	71.0	3250	CIS	5/13/1987
ARIANE 5 R/B	796	748	98.6	2575	FR	January 3, 2002
SL-8 R/B	981	955	82.9	1435	CIS	12/26/1974
SL-8 R/B	992	950	82.9	1435	CIS	7/14/1994
SL-8 R/B	1001	970	82.9	1435	CIS	August 7, 1977
SL-8 R/B	996	954	82.9	1435	CIS	3/24/1983
SL-3 R/B	896	791	81.3	1100	CIS	12/14/1982
SL-8 R/B	999	969	82.9	1435	CIS	November 1, 1984
COSMOS 2082	856	833	71.0	3250	CIS	5/22/1990
SL-8 R/B	996	953	82.9	1435	CIS	October 12, 1980
SL-8 R/B	988	966	83.0	1435	CIS	7/21/1976
COSMOS 1275	1014	954	83.0	800	CIS	April 6, 1981
SL-8 R/B	996	953	82.9	1435	CIS	11/28/1985

[1] McKnight, D. et al. (2021) “Identifying the 50 statistically-most-concerning derelict objects in LEO,” *Acta Astronautica*, 181, pp. 282–291. doi:10.1016/j.actaastro.2021.01.021.

Debris?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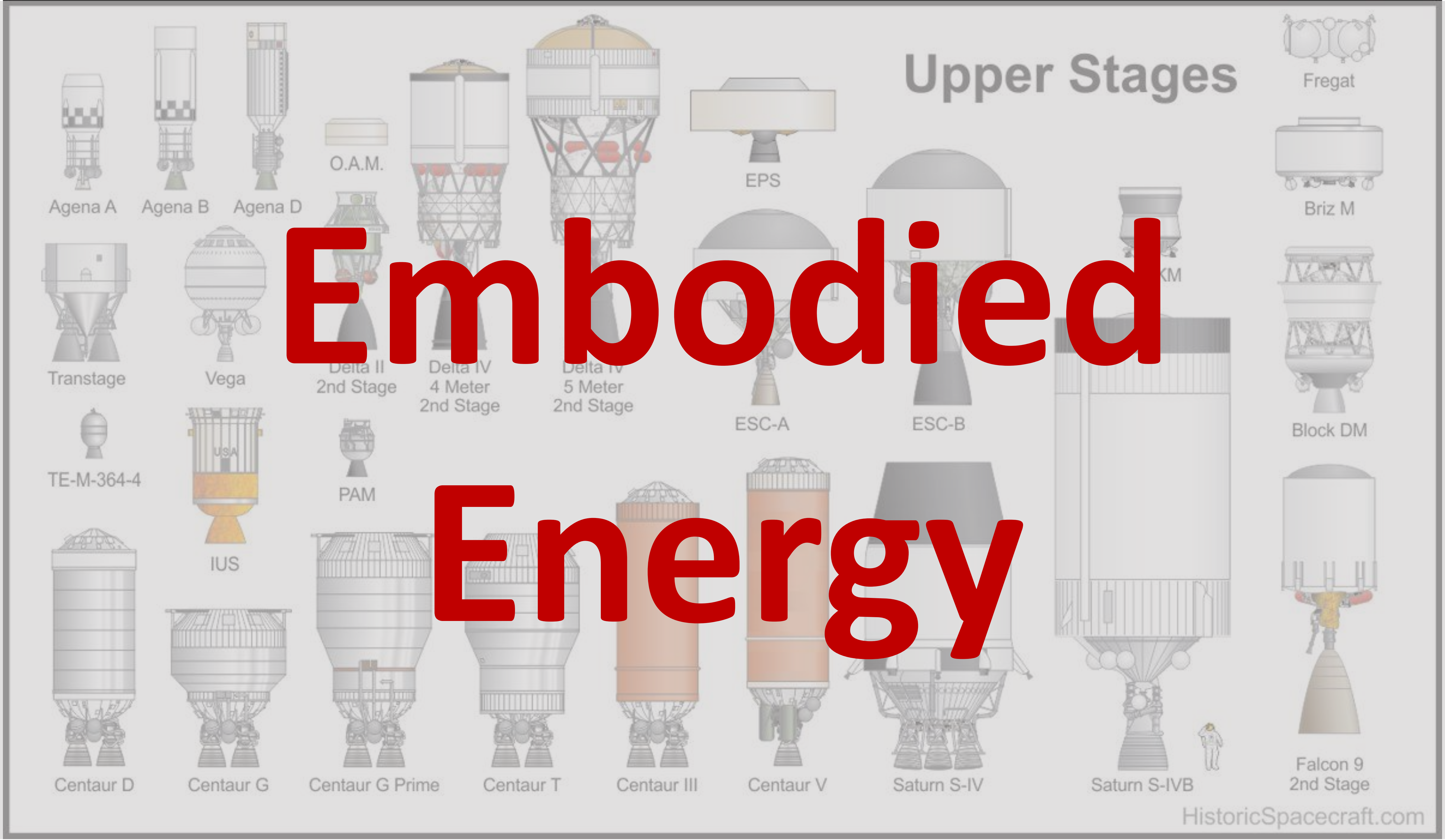
Embodied Energy



Objects in orbit have "embodied" energy due to their kinetic and potential energy

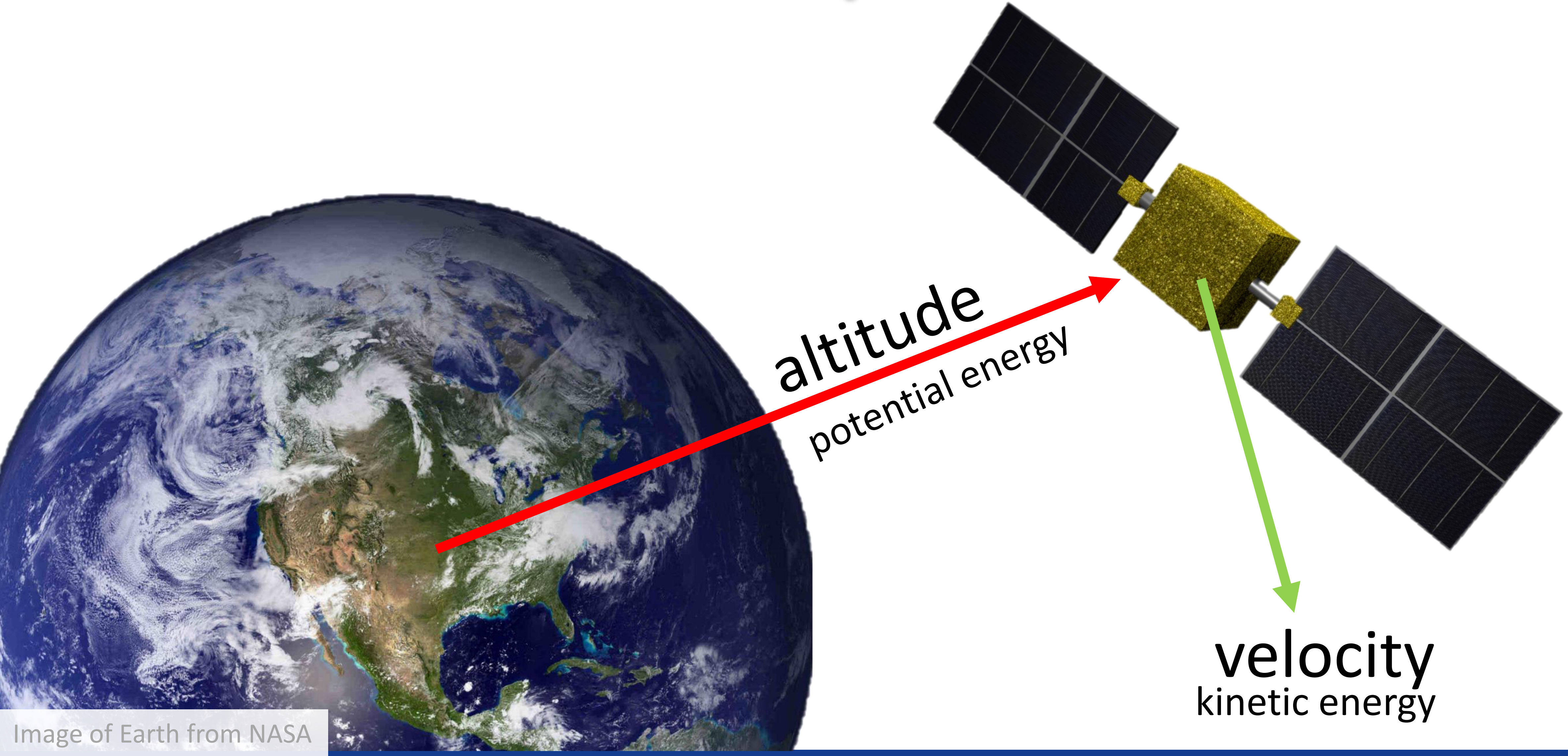


Image of Earth from NASA

The orbital energy can be expressed in terms of the orbiting body's mass and semimajor axis (for elliptical orbits)

Derived from the sum of the kinetic and potential energy

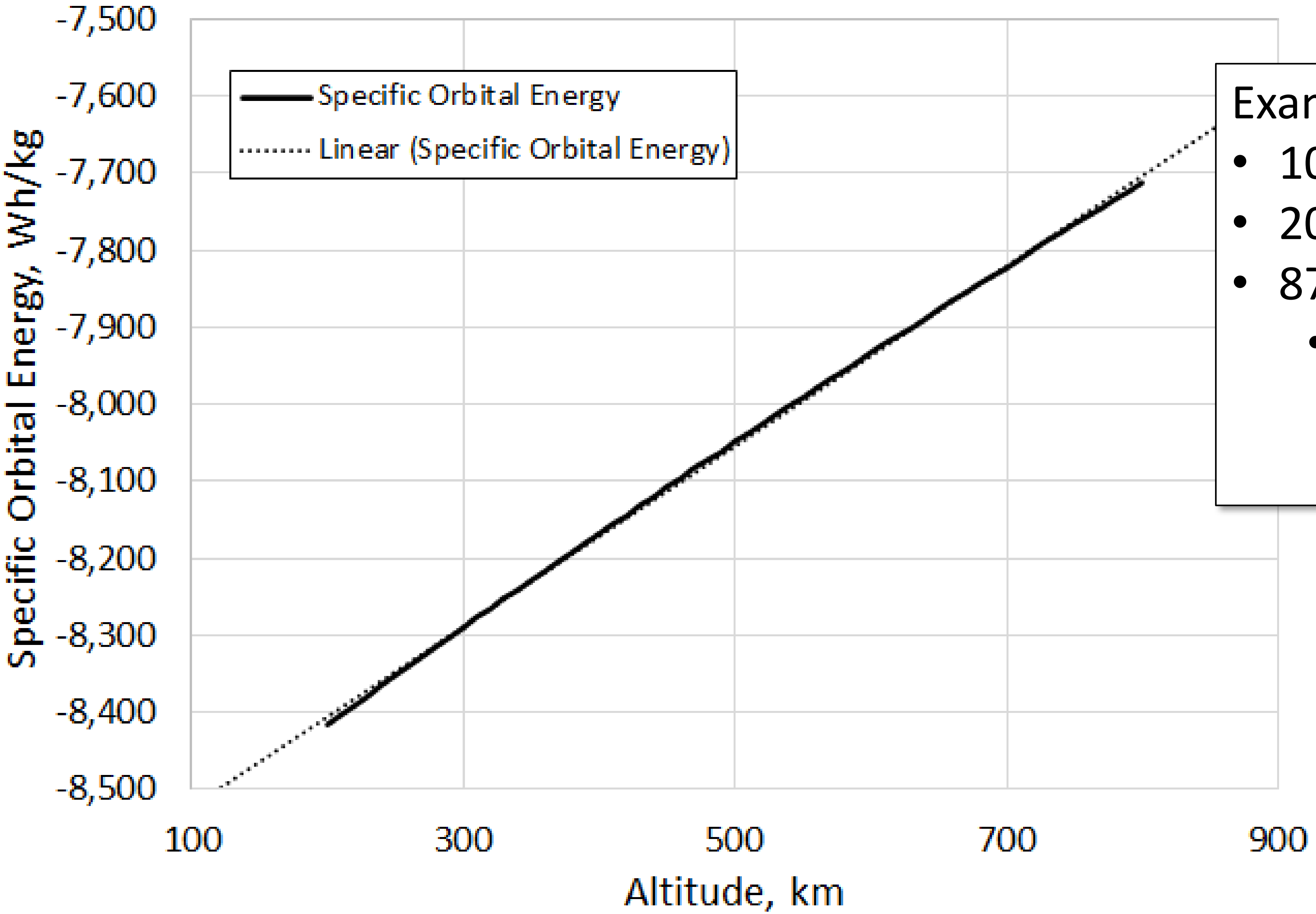
$$E_{\mathrm{orb}} = -\frac{\mu m}{2a}$$

a = semimajor axis of orbit

$\mu = G(m_1 + m_2)$, standard gravitational parameter

m = mass of orbiting body

"Orbital battery" capacity is ~1.08 Wh/(km·kg)



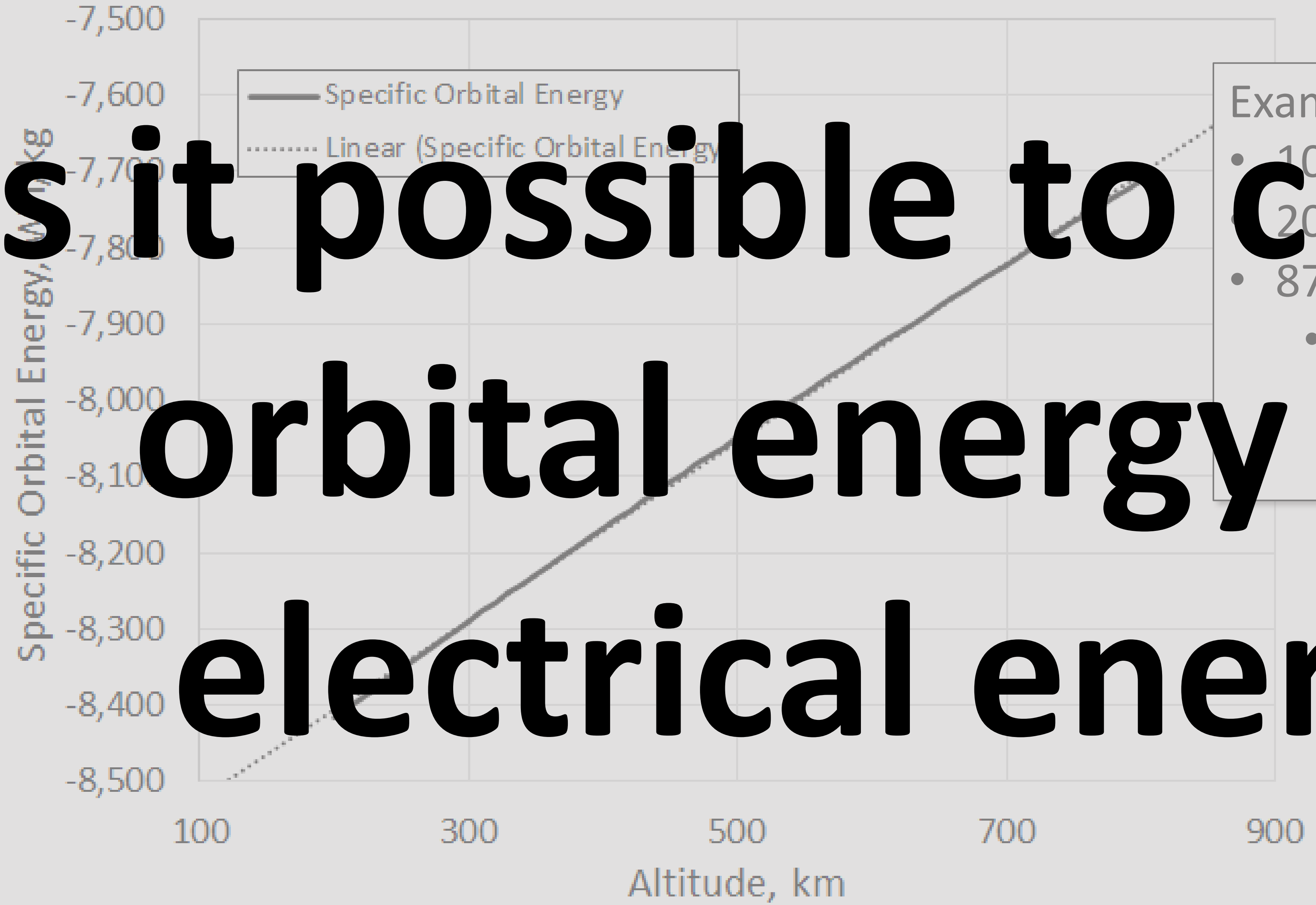
Example:
- 100-kg object in LEO
- 200-km change in altitude
- 87 MJ of energy
  - ~23 kWh
    - 5 W continuous
    - 195 days

McTernan, J.K. and S.G. Bilén, "Embodied energy repurposing via energy-harvesting electrodynamic tethers," *Journal of Spacecraft and Rockets*, Vol. 54, No. 4, 2017, DOI: 10.2514/1.A33783.

1 Wh = 3600 J

“Orbital battery” capacity is ~1.08 Wh/(km·kg)

Is it possible to convert orbital energy into electrical energy?



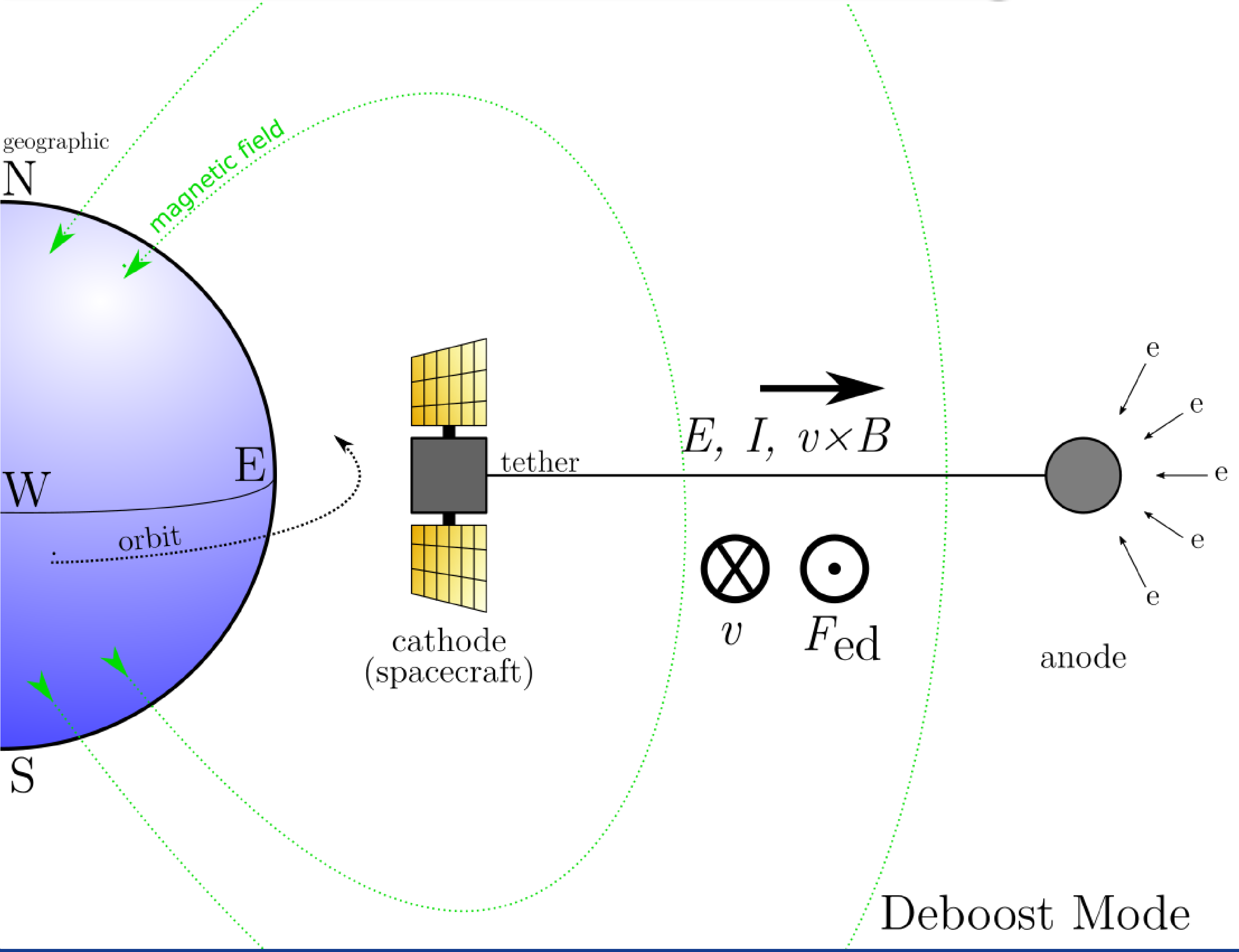
Example:

- 100 kg object in LEO
- 200-km change in altitude
- 87 MJ of energy
    - ~23 kWh
        - 5 W continuous
        - 1[illegible]5 days

McTernan, J.K. and S.G. Bilén, “Embodied energy repurposing via energy-harvesting electrodynamic tethers,” *Journal of Spacecraft and Rockets*, Vol. 54, No. 4, 2017, DOI: 10.2514/1.A33783.

1 Wh = 3600 J

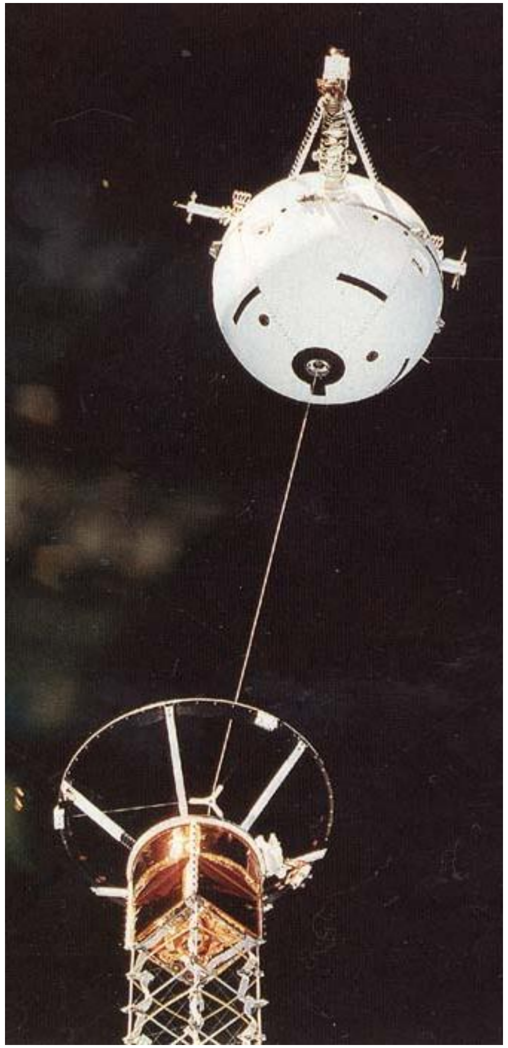
An electrodynamic tether can convert orbital energy to electrical energy

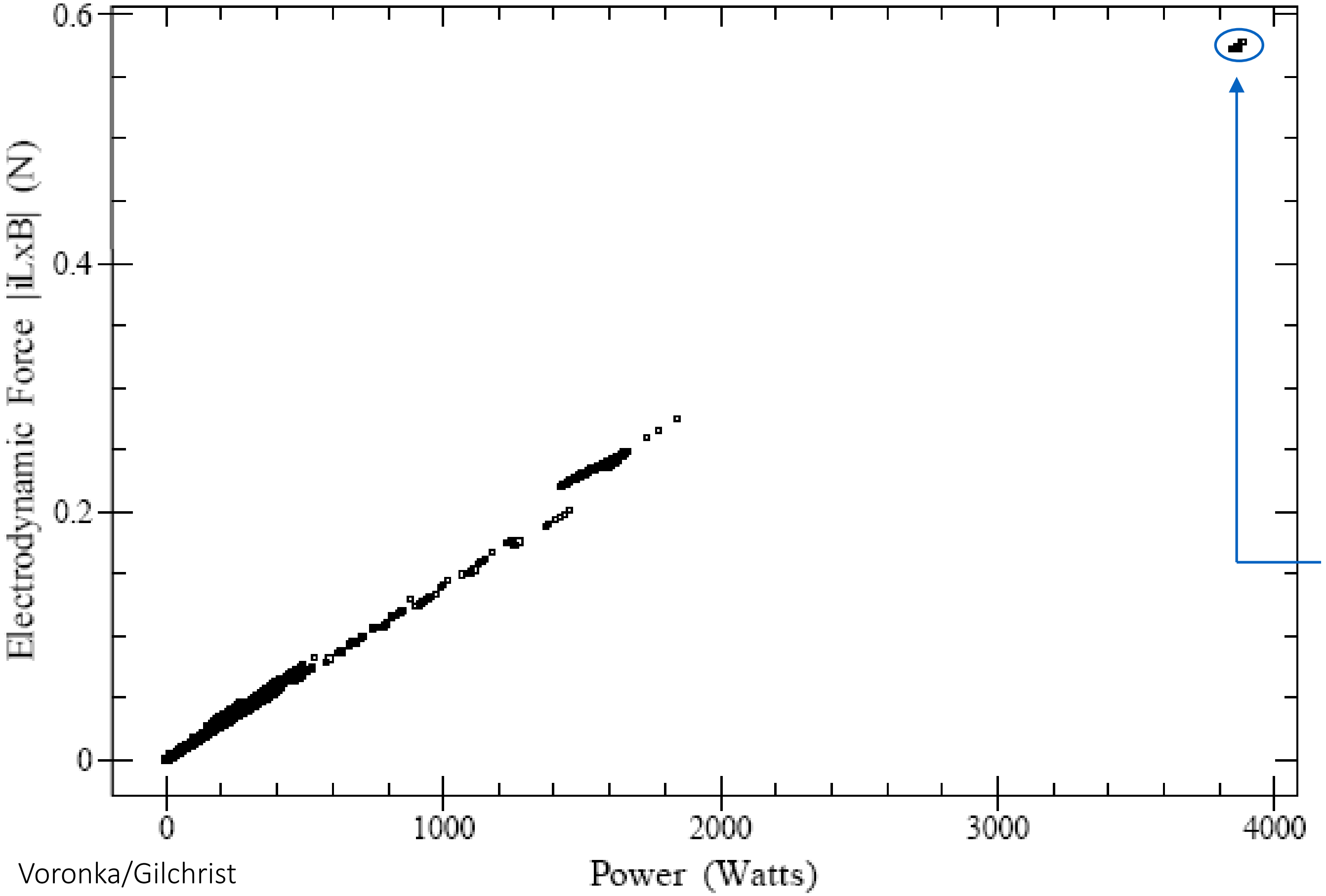


electrons from the surrounding plasma

Power generation with EDTs was demonstrated on TSS-1R

1992 (TSS-1) & 1996 (TSS-1R)





Voronka/Gilchrist
UM

- Space Shuttle Orbiter-based demonstration of EDTs (1996)
- TSS-1R system generated ~3.8 kW power
- Max. electrodynamic drag calculated to be ~0.55 N at tether break

Energy Harvesting

- Total energy defined in ECI (Earth-centered Inertial) frame of spacecraft system:

$$E_{\mathrm{tot}} = E_{\mathrm{orb}} + E_{\mathrm{elec}} + E_{\mathrm{earth}} + E_{\mathrm{drag}}$$

E_{drag} → Energy obtained from atmospheric drag force on the satellite

E_{earth} → Rotational kinetic energy of Earth

E_{elec} → Electrical energy dissipated or inject into electrical system

E_{orb} → Total orbital energy (kinetic + potential energy)

- Change in total energy for each timestep:

$$\Delta E_{\mathrm{orb}} = \Delta E_{\mathrm{elec}} + \Delta E_{\mathrm{earth}} + \Delta E_{\mathrm{drag}}$$

- A spacecraft equipped with an EDT can rendezvous with space debris to strategically repurpose the debris' embodied orbital energy into electrical energy while deorbiting it

Energy Harvesting (cont.)

- Total electrical power:

$$P_{\mathrm{elec}} = P_{\mathrm{anode}} + P_{\mathrm{cathode}} + P_{\mathrm{storage}} + P_{\mathrm{load}} + P_{\mathrm{tether}}$$

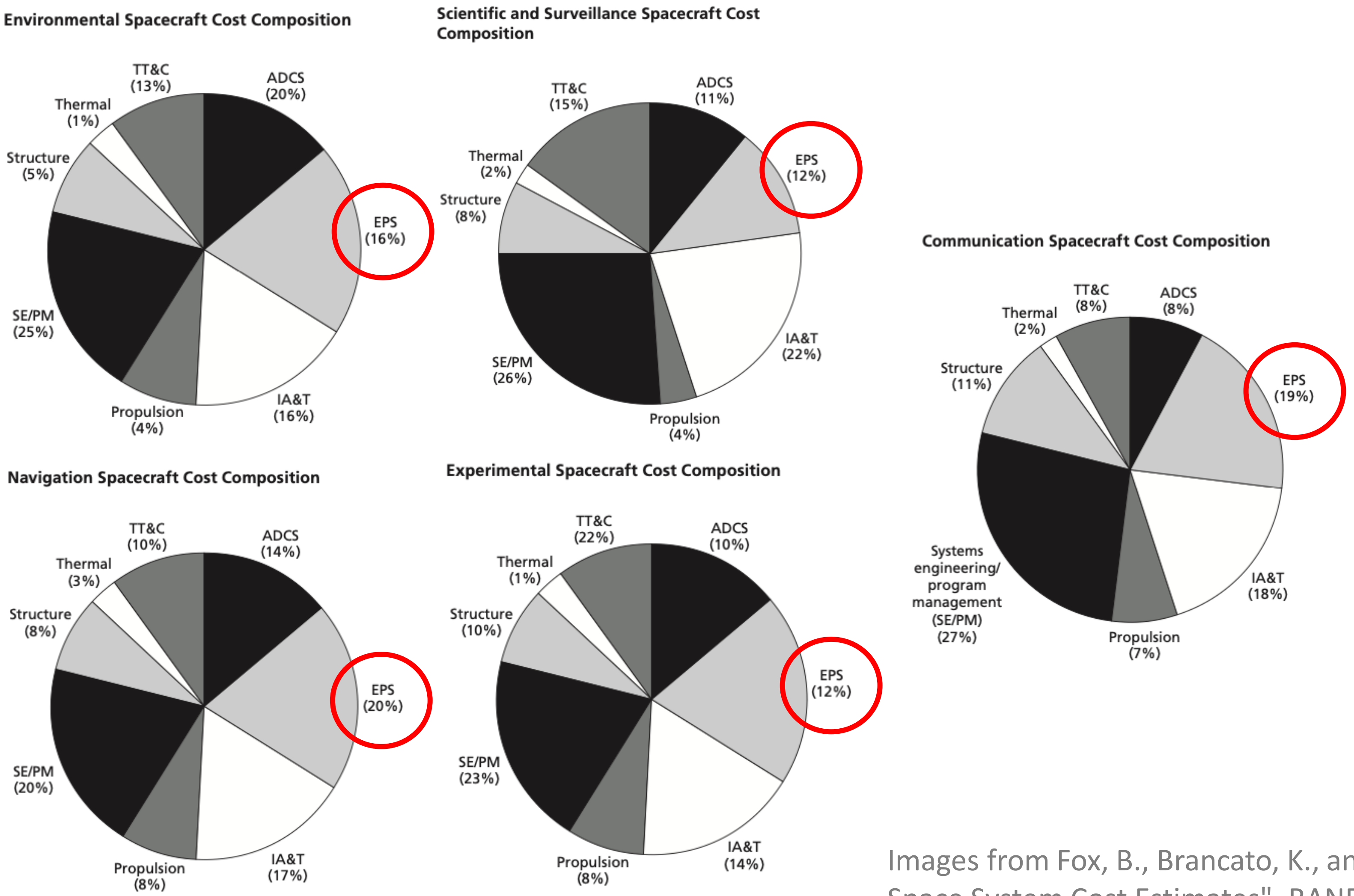
$$P_{\mathrm{elec}} = P_{EMF \times I} = \varepsilon I$$

- Through conservation of energy, a relation between orbital and electrical energy can be given by:

$$\Delta E_{\mathrm{orbital}} = E_{\mathrm{electrical}} = \varepsilon I \Delta t$$

$$\frac{\mu m}{2}\left(\frac{1}{a_2} - \frac{1}{a_1}\right) = \varepsilon I \Delta t$$

Using an EDT system to remove debris could reduce costs of high-power systems missions



Traditional EPS can make up 20% of the spacecraft cost

An important factor of EPS design is high specific power, produced by low mass, HE arrays

Many of these missions require < 5 kW of power and could be replaced by EDTs

Images from Fox, B., Brancato, K., and Alkire, B., "Guidelines and Metrics for Assessing Space System Cost Estimates", RAND Corp., TR-418-AF, Santa Monica, Calif., 2008

EDTs could provide economic value to a deorbit mission

The power generated by deorbiting debris could save **millions** per mission* [2]

→

Orbital debris has a value worth **millions**

incentive

Larger debris → Greater energy → Greater power → Greater savings

*Excluding EDT system cost

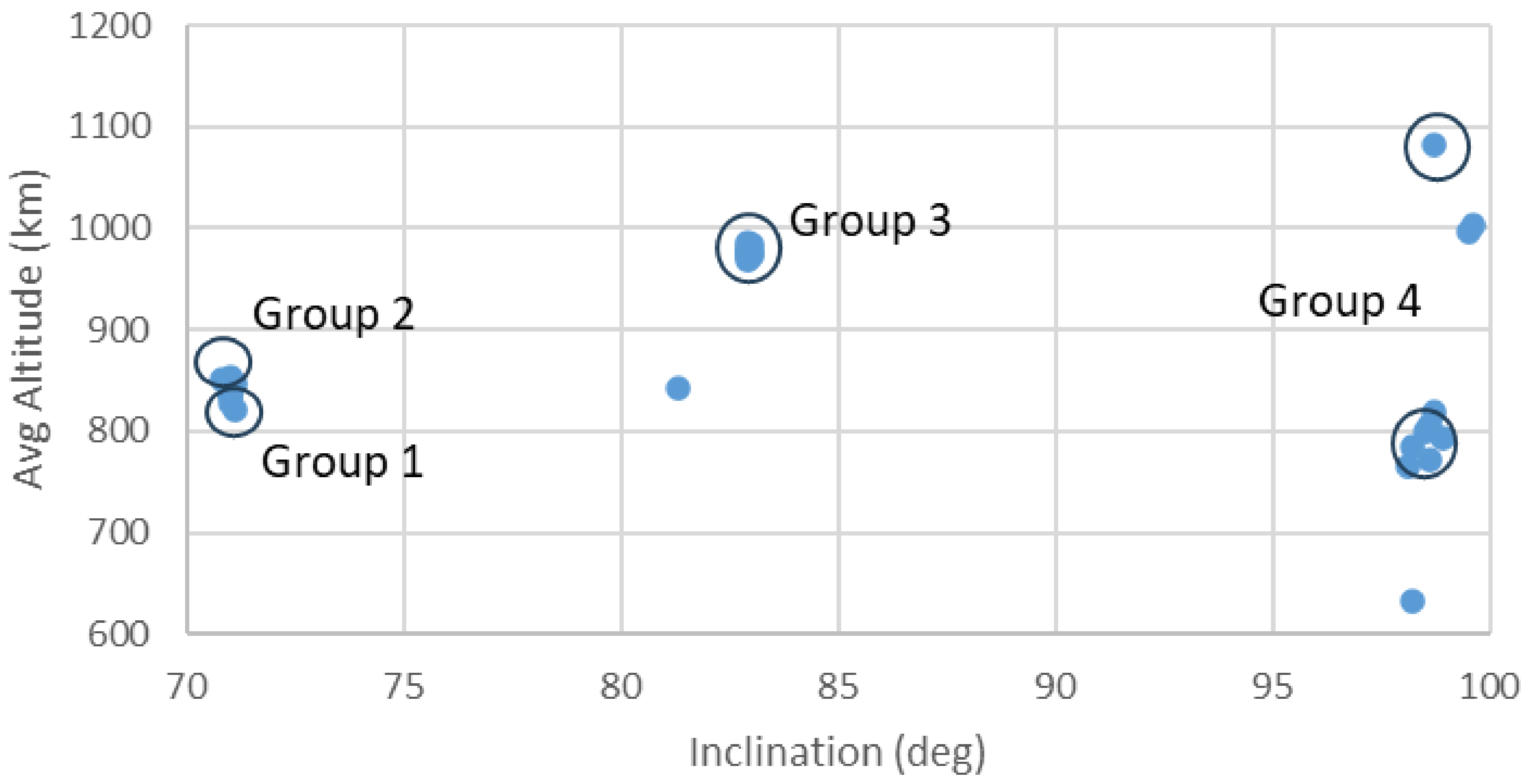
Therefore, EDT systems could incentivize the **dual purpose** of

debris mitigation

and

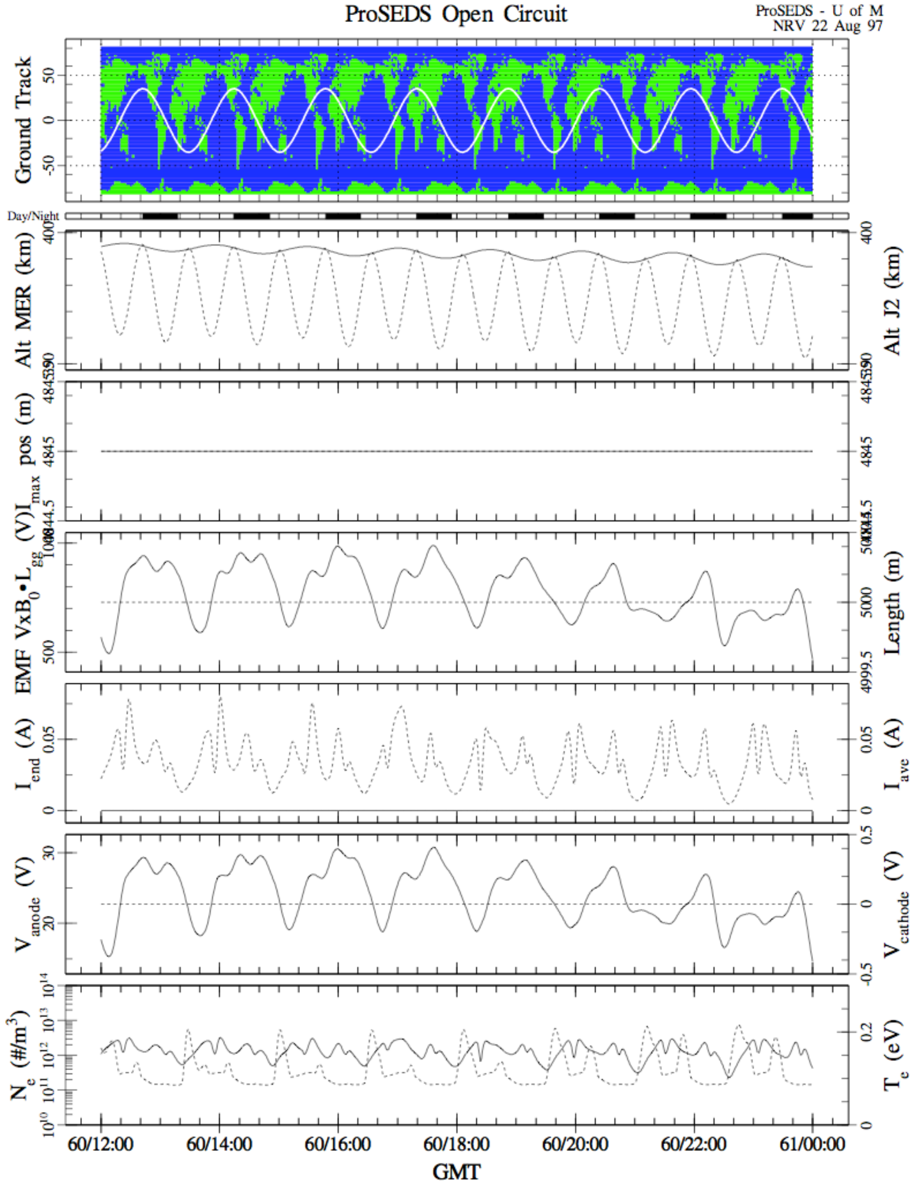
supplying high power

39 SMC objects put into 4 groups based their similarity for use in simulations



TeMPEST is a powerful tool for analysis of EDT systems

- Ability to perform analyses at different scale size
  - Global scale
  - At fixed location during the course of a mission as function of any parameter (e.g., altitude)
- Capable of handling time scales from sub-second to years (in orbital mode)



Collaborator Brian Gilchrist
University of Michigan

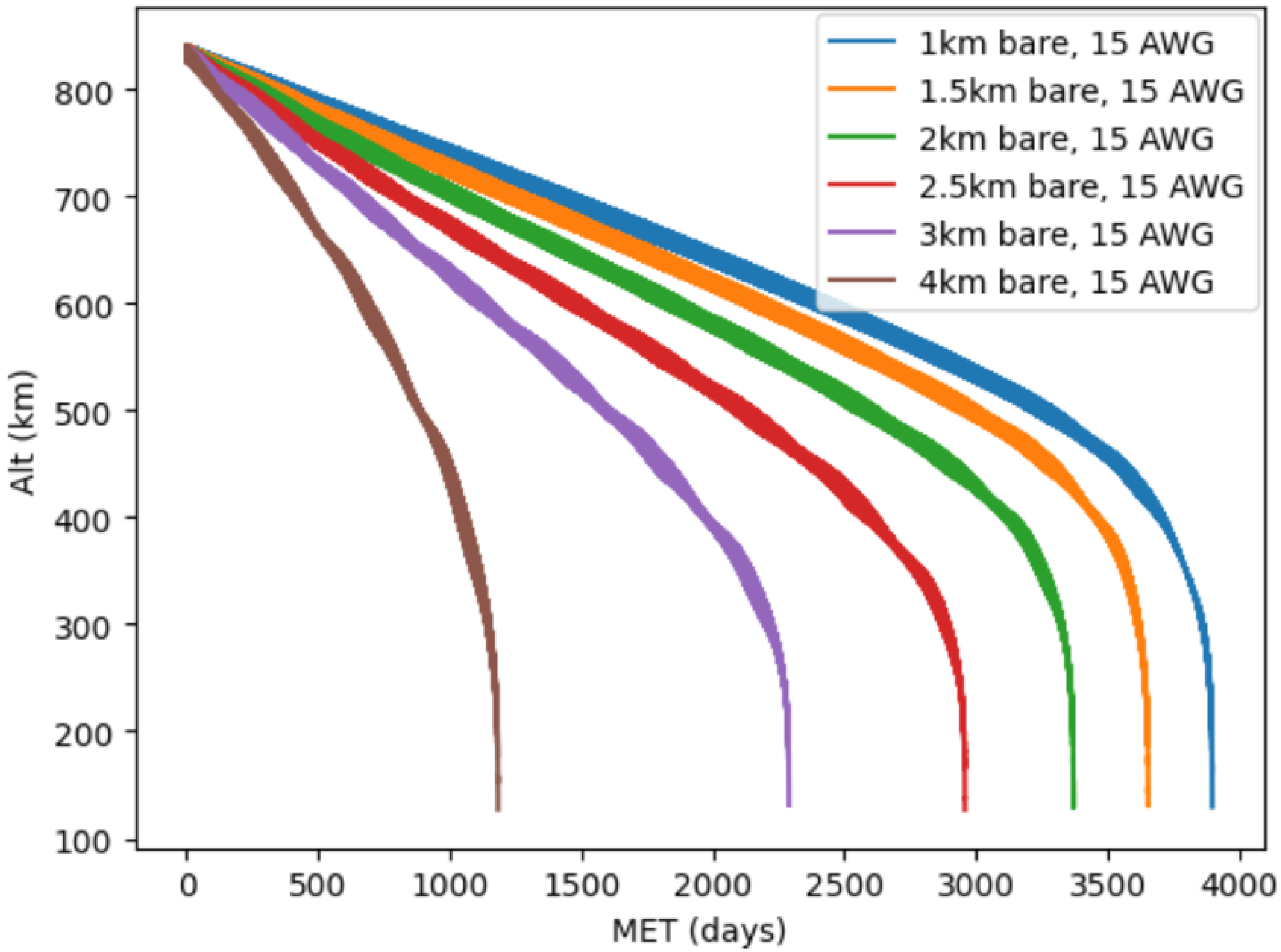
We use TeMPEST to perform EDT system trades and simulations

- TeMPEST – Tethered Mission Planning and Evaluation Software Tool computation capabilities
  - Orbital position/velocity
  - Local magnetic field strength (IGRF)
  - Local ionospheric and atmospheric conditions (IRI, MSIS)
  - Motional induced *emf*
  - Drag/thrust forces produced by the tether current
- TEMPEST was extended at Penn State to model energy harvesting and storage

Define EDT System Composition

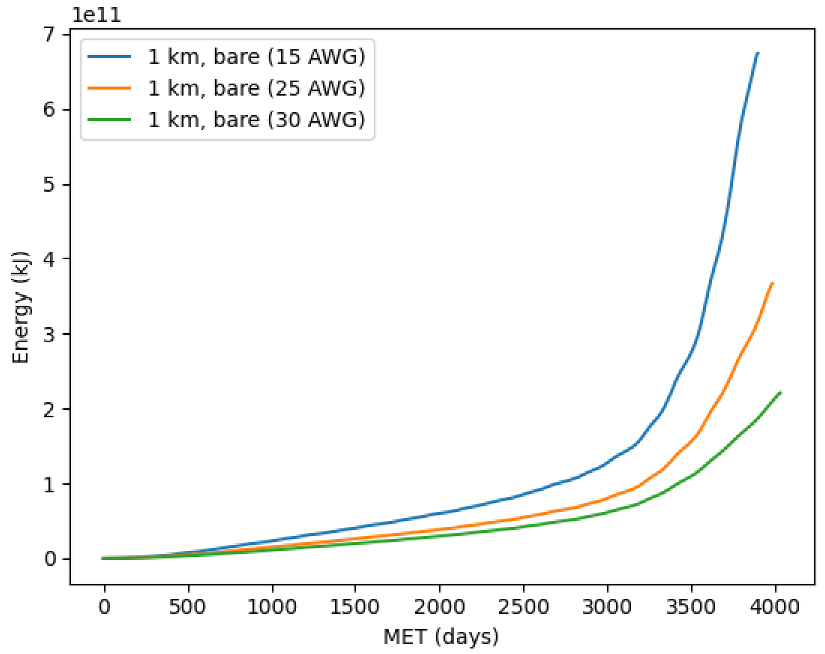
- Small sats system (representing a satellite under 180 kg)
- Total mass = 200 kg
- Tether mode = passive (deorbiting)
- Apogee, perigee, inclination = as described in orbital parameters of debris
- RAAN = 0.0 deg
- True Anomaly = 0.0 deg
- Tether length = 1–4 km
- Tether diameter = 15 AWG, 25 AWG, 30 AWG
- Tether material = Aluminum

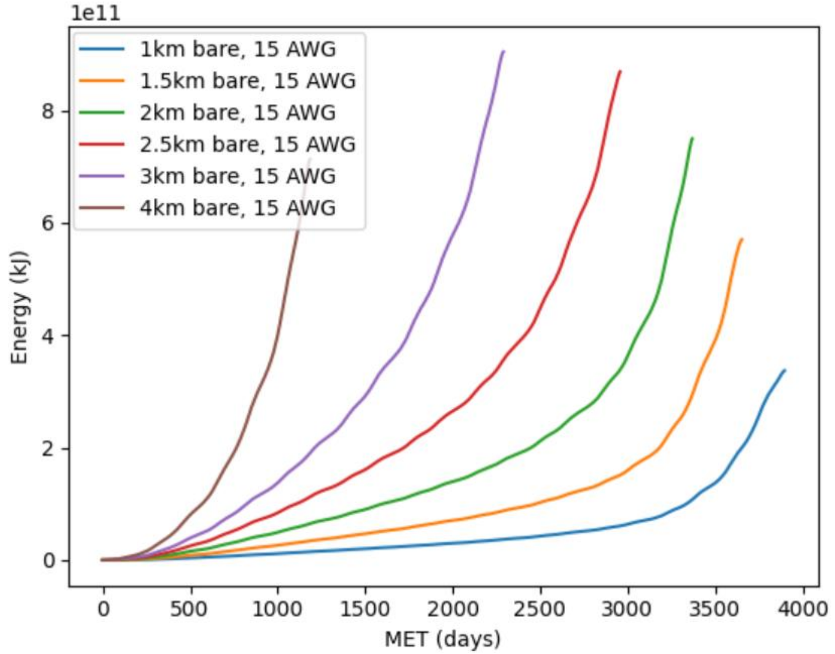
Altitude vs. MET (Mission Elapsed Time)



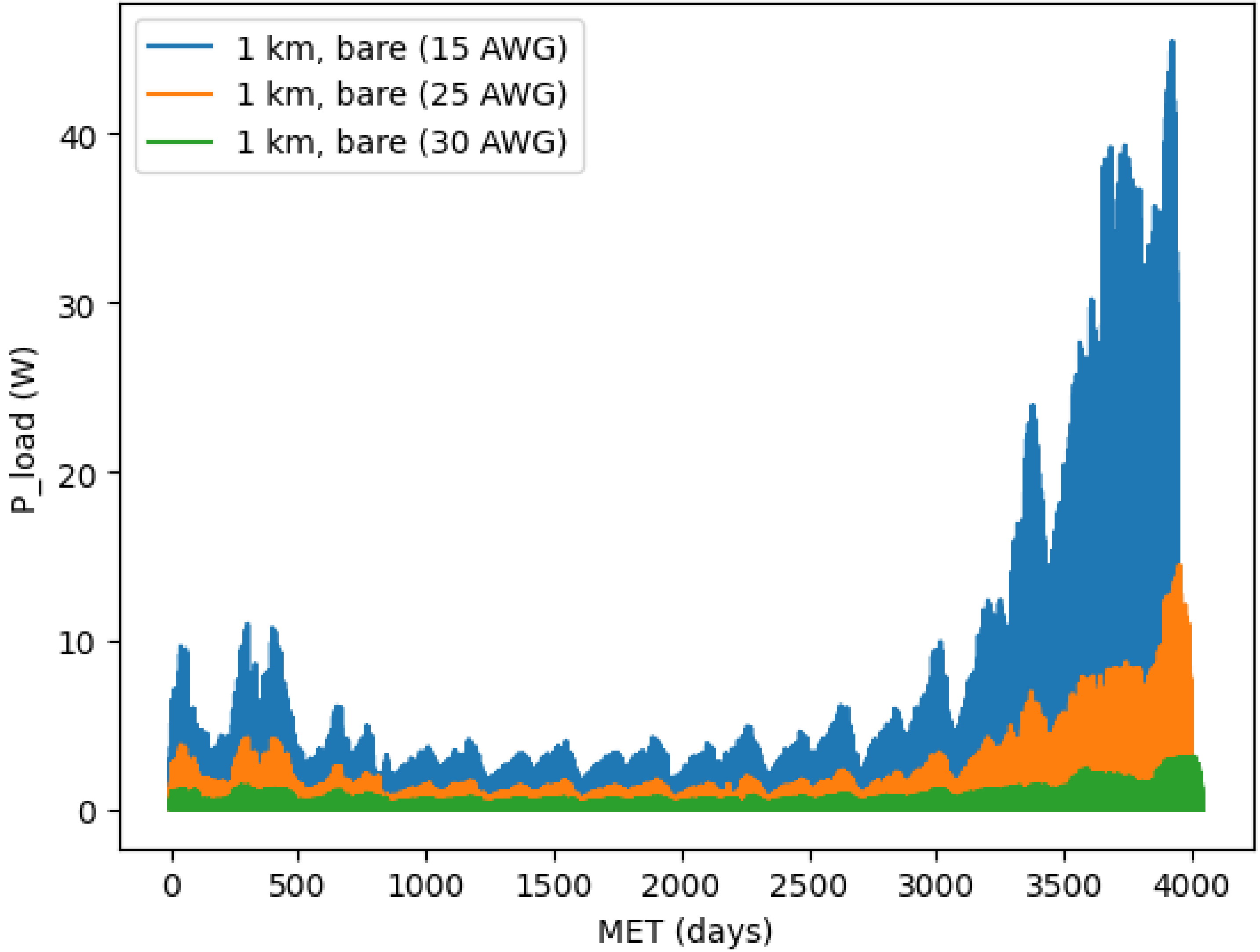
Total Energy vs MET

- **Top:** Total energy vs MET for constant tether length (1 km) different tether diameter
- **Bottom:** Total energy vs Met for constant tether diameter (15 AWG) different tether length
- Increased tether diameter results in higher current flow, leading to greater energy harvesting
- As the length of the tether increases, the generation of energy (electrical power) increases rapidly





Power vs MET

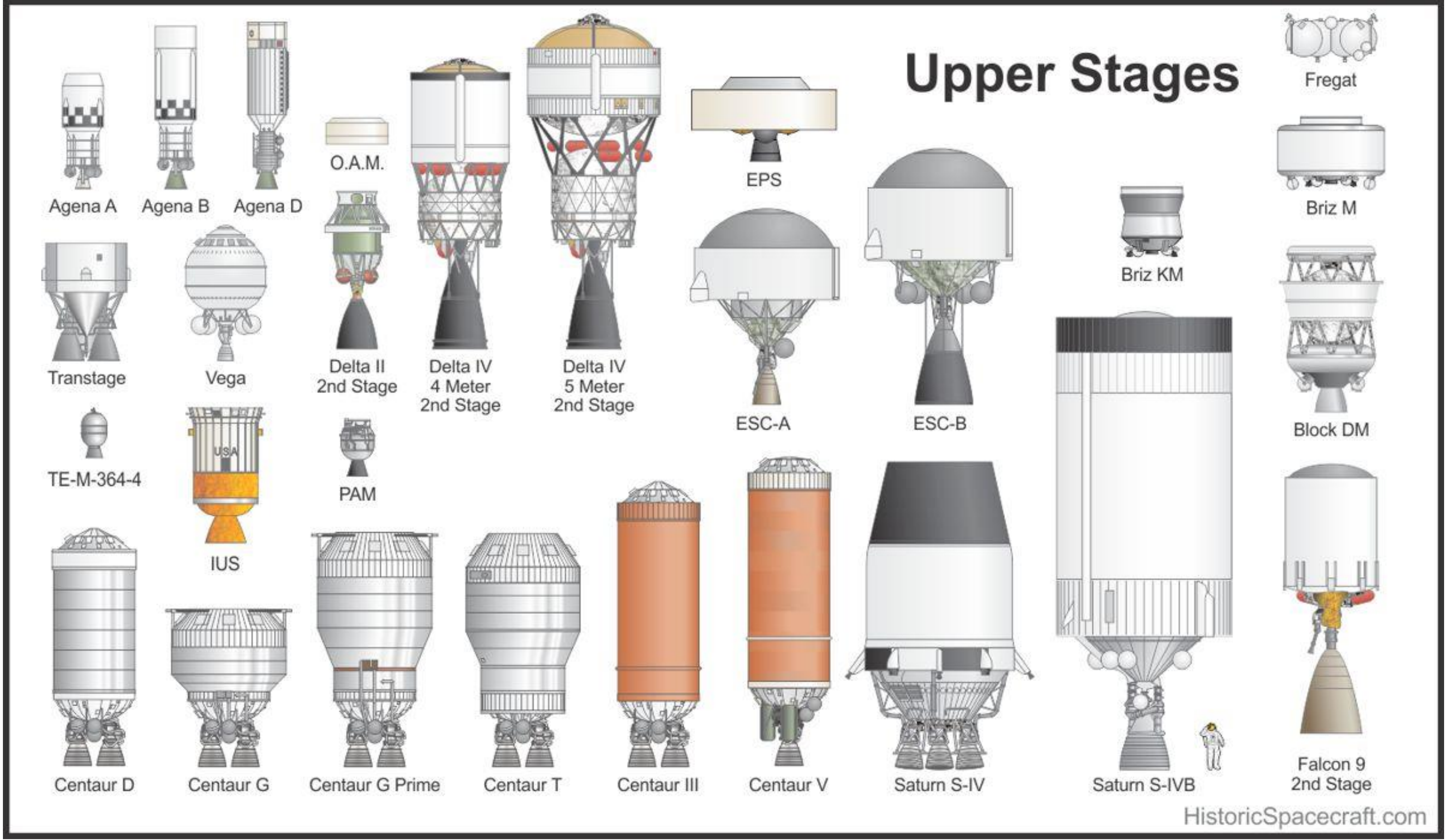


1 km, bare (15 AWG)
1 km, bare (25 AWG)
1 km, bare (30 AWG)
P_load (W)
0
10
20
30
40
0
500
1000
1500
2000
2500
3000
3500
4000
MET (days)

Summary: There is a lot of energy available

Approximate energy per object that could be harvested assuming 1.08 Wh/(kg km) orbital energy density and assuming a final altitude 300 km

Group	Objects in group	Average apogee (km)	Eccentricity	Inclination	Mass (kg)	Energy available (kWh)[1]
1	18	848	0.00105	70.99	9000	5251
2	6	857	0.00144	71.00	3250	1919
3	11	993	0.00229	82.92	1435	1048
4	4	934	0.00949	98.58	3235	1976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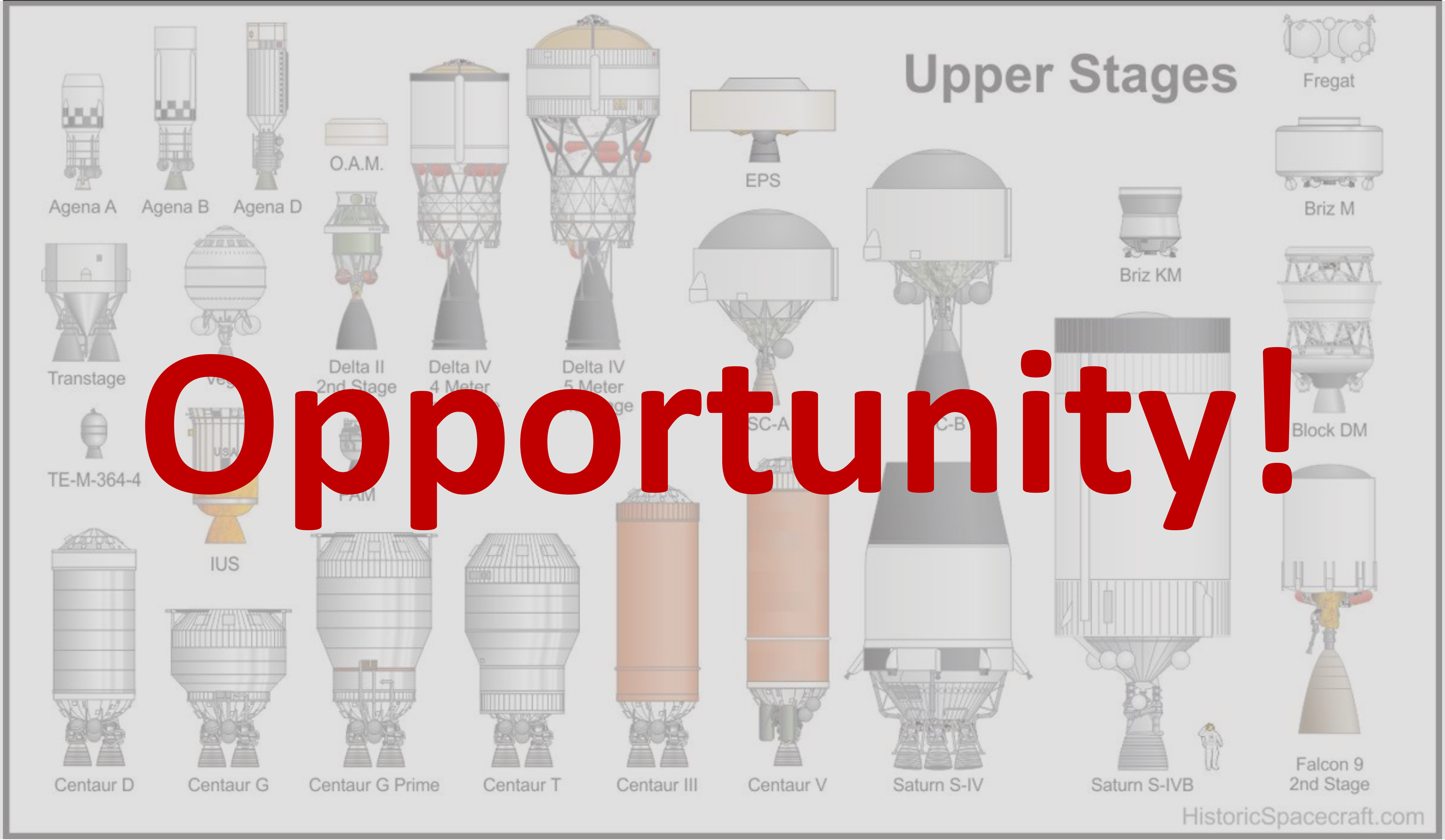

Opportunity!