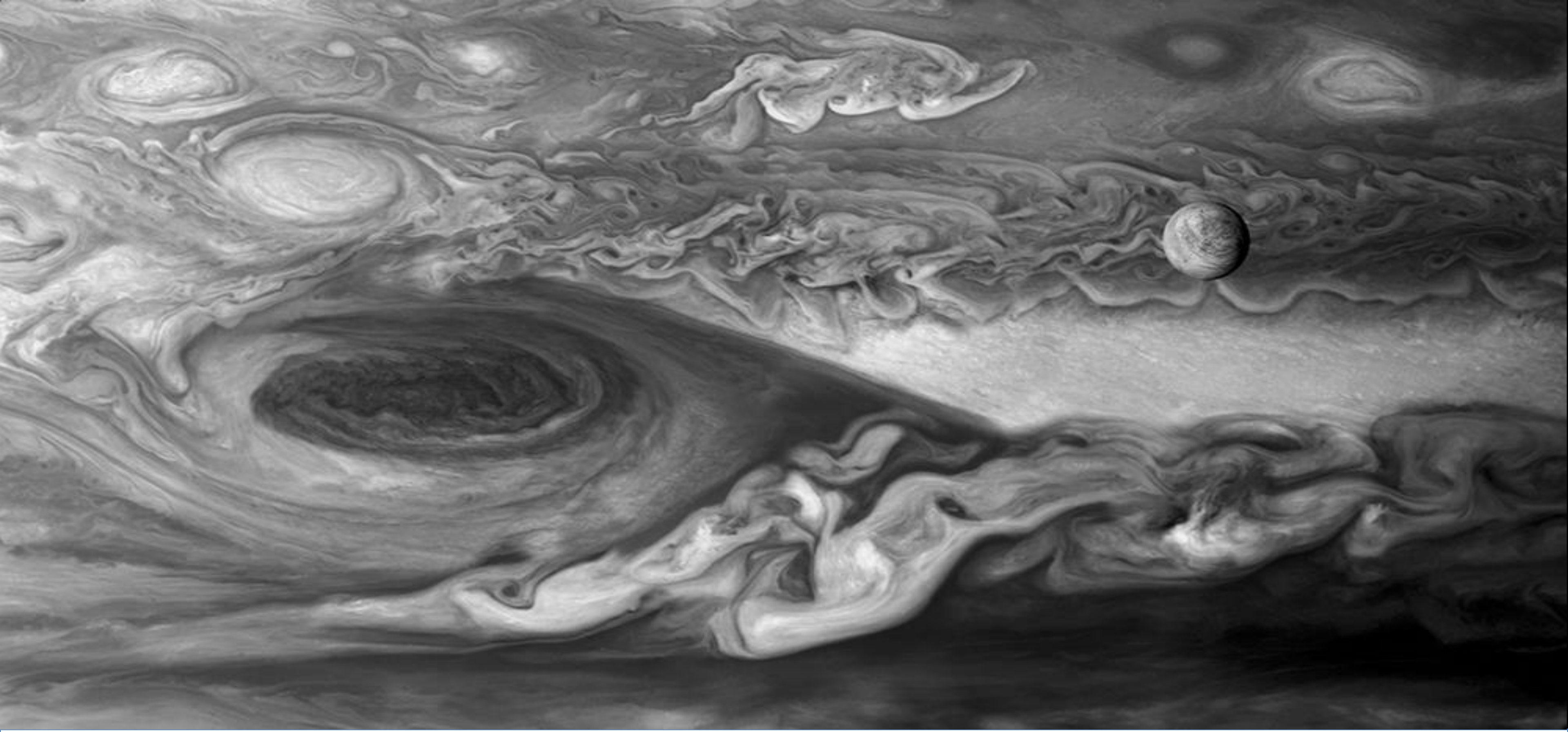

Solid Rocket Motor Incremental Modeling for System Impacts

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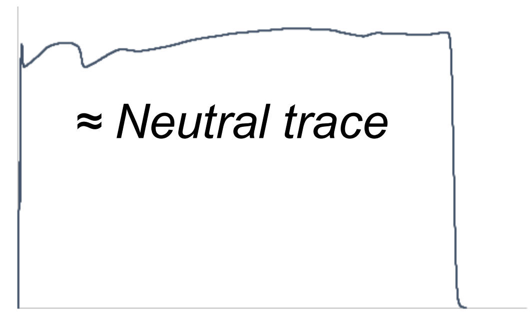
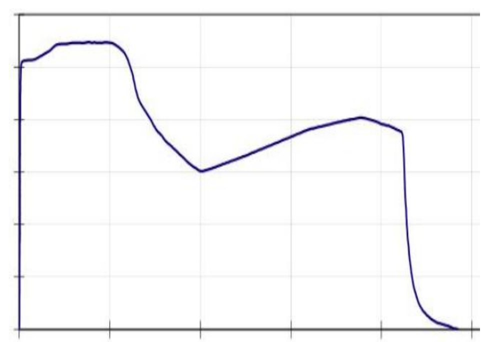
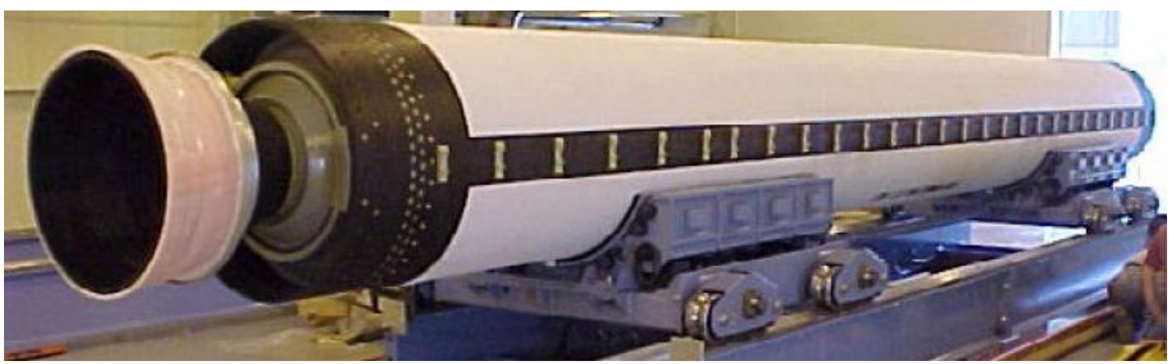
In-space motors are finding new roles

- In-space motors have the same components as launch vehicle boosters or stages
- In-space motors have had different design drivers



Thrust vs. time





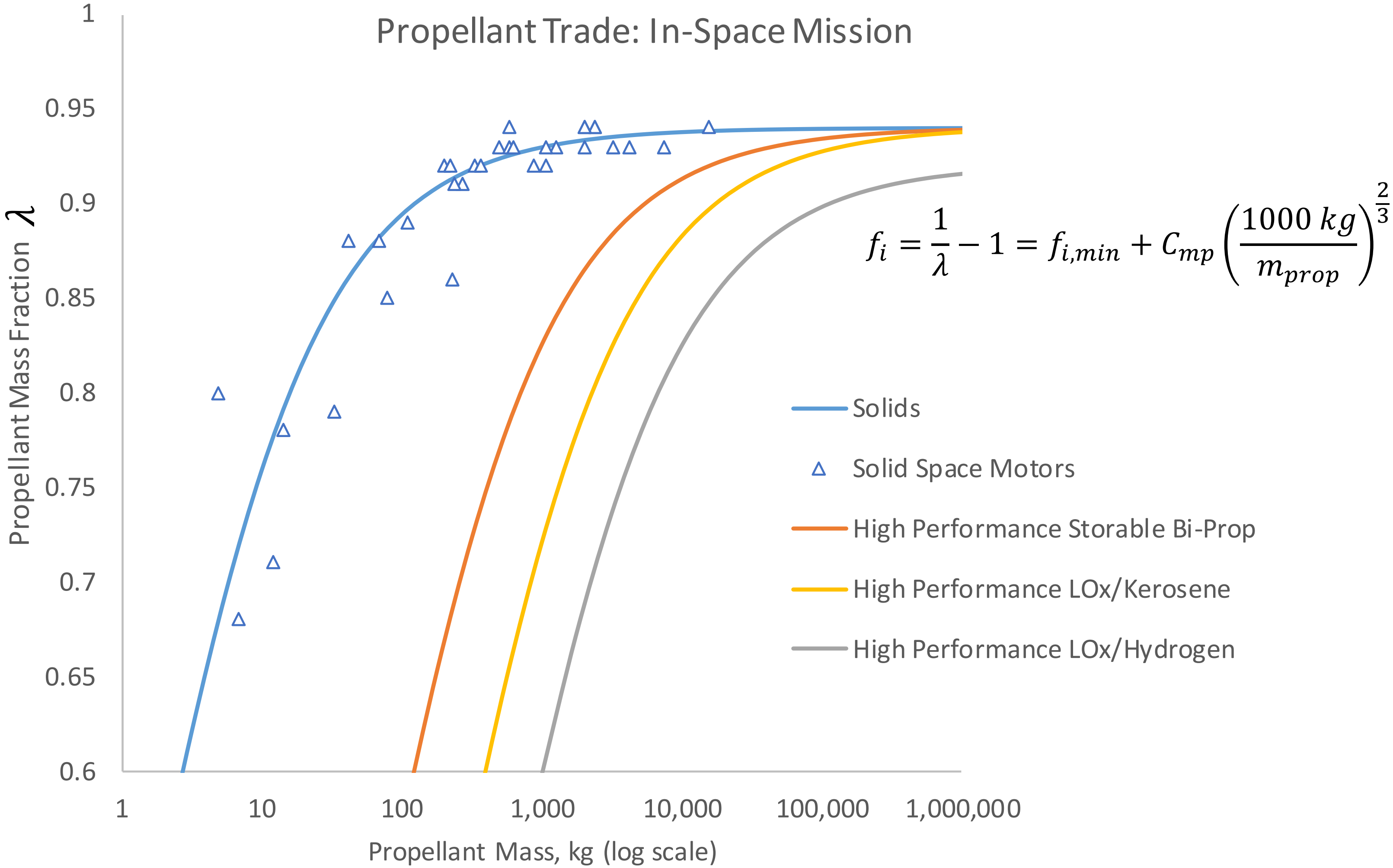
Space Solids experience highlights

- **Inertial Upper Stage (IUS)[1] –** 2 solid stages flown out of Shuttle, Titan

- **Leonidas Stg 2 & 3[2] –** Super Strypi upper stages
- **Delta II –** 60+ STAR 48B launches

- **Magellan –** Longest in-space aging before firing, 15 mos[3]
- **Ulysses –** highest ΔV (4 km/s), max acceleration (11 g's)
- **New Horizons –** highest final velocity leaving earth
- **LADEE Minotaur V –** recent 5-solids lunar launch vehicle
  - 3 Peacekeeper stages, STAR 48BV, STAR 37FMV
- **SFDT[4] –** Demonstrated offload repeatability and lots of data
- **Parker Solar Probe** –
  STAR 48BV atop a Delta IV-Heavy – high ΔV mission

When to use Solids? When the mass fraction *pro* outweighs the Specific Impulse *con*



What's new now?
Missions where time and shape is of the essence

- **0 impulse**
  - Specific impulse
- **1 burn time and dimensions**
- **2 shape-limiting**
  - Limit Q or g: approximate regressive trace
  - Scaling a catalog design
- **3 variation sensitive**

ΔV, impulse estimate

?

Detail design & analysis

0: Estimate Propellant Mass & Specific Impulse

$$\Delta V = Isp \ln\left(1 + \frac{m_{prop}}{m_{final}}\right) = Isp \ln\left(1 + \frac{1}{\left(\frac{m_{payload}}{m_{prop}}\right) + f_i}\right)$$

- **A: General target motor**
  - Assume typical high-performance propellant (c* contribution to Isp)
    - For first iteration, f_i = 0.9 and Isp = 290 s
  - Expansion ratio (ε) is primary driver for Isp
    - propellant mass can change it, but less drastically than for f_i
  - Correlations:

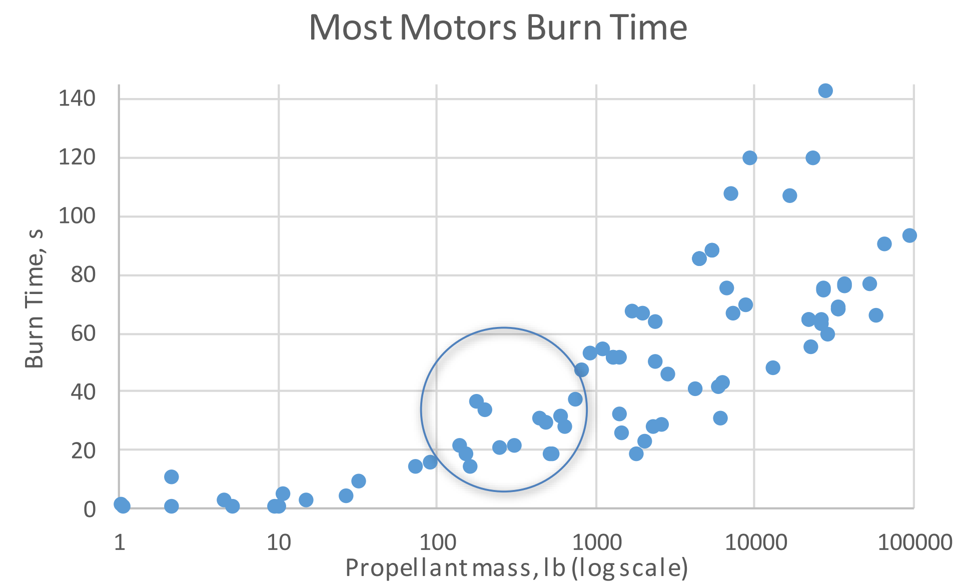
$$Isp = a_{Isp} \ln\left(\frac{\epsilon}{\epsilon_0}\right) + b_{Isp} \qquad f_i = f_{i,min} + C_{mp}\left(\frac{1000\ kg}{m_{prop}}\right)^{\frac{2}{3}}$$

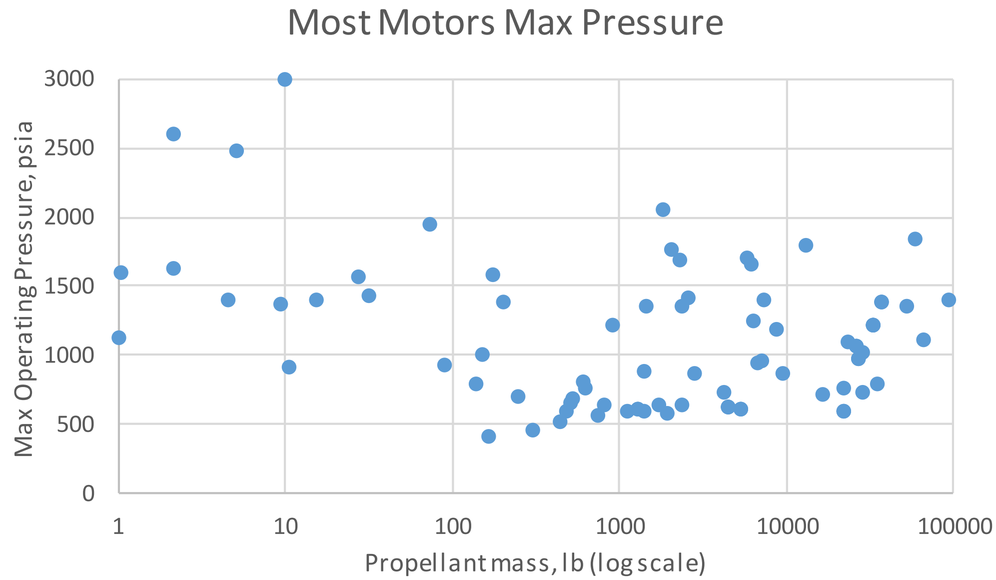
- **B: Departing from a reference motor**

$$Isp_2 = Isp_1 + a_{Isp} \ln\left(\frac{\epsilon_2}{\epsilon_1}\right) \qquad f_{i2} = f_{i1} + C_{mp}\left[\left(\frac{1000\ kg}{m_{prop2}}\right)^{\frac{2}{3}} - \left(\frac{1000\ kg}{m_{prop1}}\right)^{\frac{2}{3}}\right]$$

State of the Art

- **Longer burn times drive innovation for smaller motors**
  - End burners possible, but increase fi
- **Smaller motors → higher pressure easier**
  - "Min gage"





1: Burn time and dimensions

- **If burn rate is limited:**
  4 equations, 7 unknowns =
  2 independent variables
  - Often, case diameter and pressure drive
    - Burn Time
    - Thrust
    - Case Length
    - Throat Area

$$P_{avg}A_t t_{b,max} = m_{prop}c^*$$

$$t_{b,max} \approx \frac{0.8D_{case}/2}{\dot{r}_{ref}\left(\frac{P_{avg}}{P_{ref}}\right)^n}$$

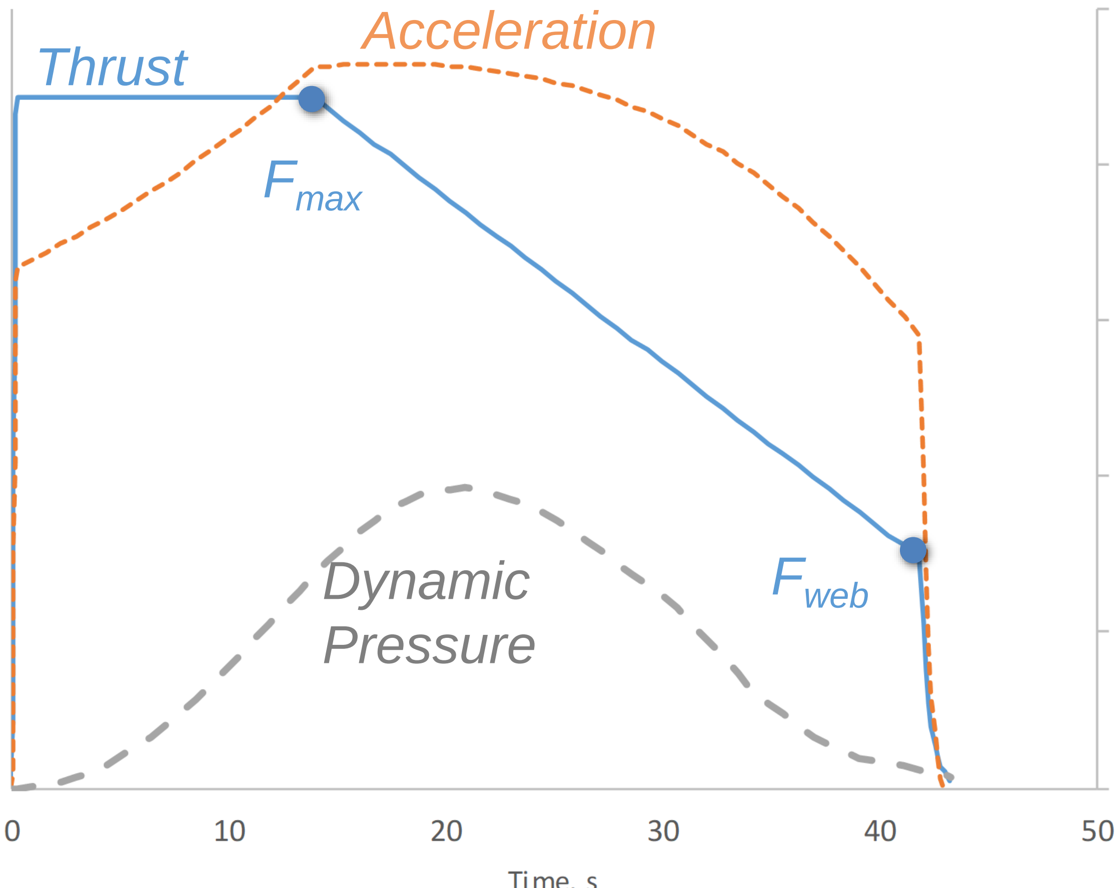
$$F_{avg} = \frac{m_{prop}Isp}{t_{b,max}}$$

$$m_{prop} \approx f_v \rho_{prop} \pi D_{case}^2 \left[\frac{1}{6}D_{case} + \frac{1}{4}(L_{case} - D_{case})\right]$$

- **With selectable burn rate, one more free variable**

2: Design-to regressive trace

- **Goal: Notional but realistic traces limiting dynamic pressure at separation**
  - 0: Set
    - m_{prop} & Isp
  - 1: Set
    - Propellant
    - Case Diameter
    - Throat Area
  - 2: Set F_{max}/F_{web}
  - Results in
    - Pressure (max & avg)
    - Thrust (max & avg)
    - Burn Time
    - Case Length



  - Burn time and dynamic pressure or acceleration not acceptable? iterate

2: “Can I get this thrust trace in a size XXL?”

- **Assume same propellant**
- **2 steps:**
  - Scale to new diameter and propellant mass, at constant pressure

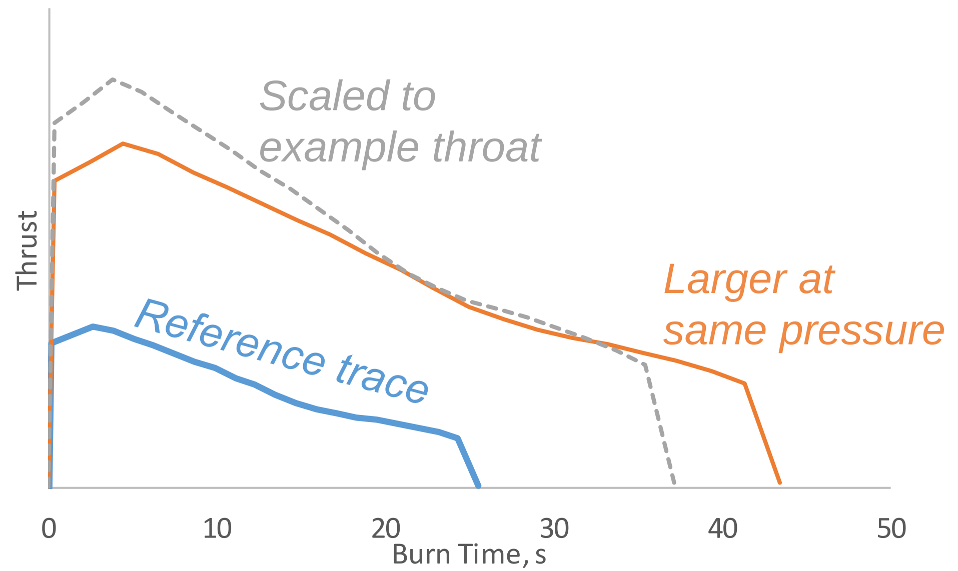
$$\frac{t_2}{t_1} \approx \frac{D_2}{D_1} \qquad \frac{\dot{m}_2}{\dot{m}_1} = \frac{m_{p2}}{m_{p1}}\left(\frac{t_2}{t_1}\right)^{-1}$$

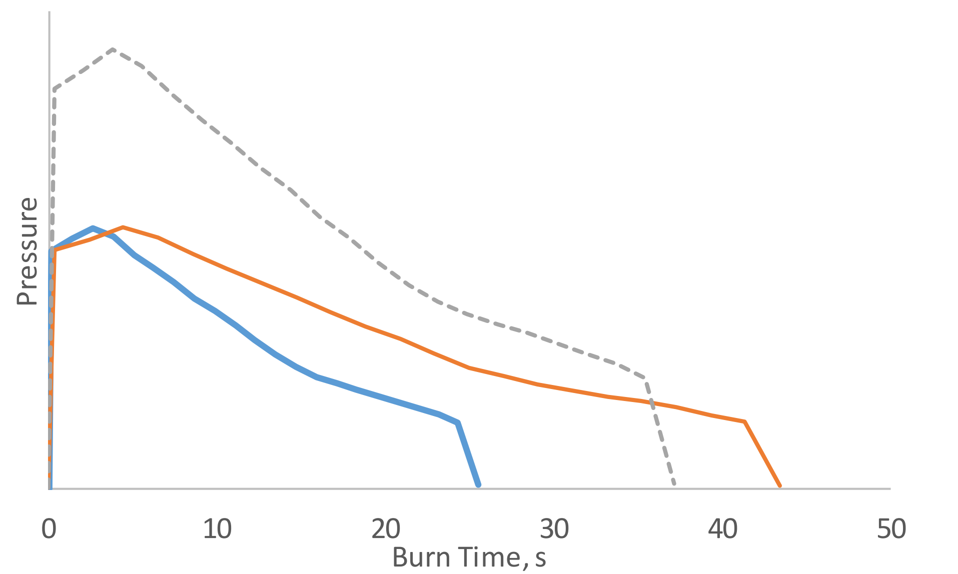
$$\frac{A_{t2}}{A_{t1}} = \frac{\dot{m}_2}{\dot{m}_1} \qquad \frac{F_2}{F_1} = \frac{\dot{m}_2}{\dot{m}_1}\frac{Isp_2(\epsilon_2)}{Isp_1(\epsilon_1)}$$

  - Scale to new condition by setting throat

$$\frac{t_3}{t_2} = \left(\frac{A_{t3}}{A_{t2}}\right)^{\frac{n}{1-n}} \qquad \frac{\dot{m}_2}{\dot{m}_1} = \left(\frac{A_{t3}}{A_{t2}}\right)^{-\frac{n}{1-n}}$$

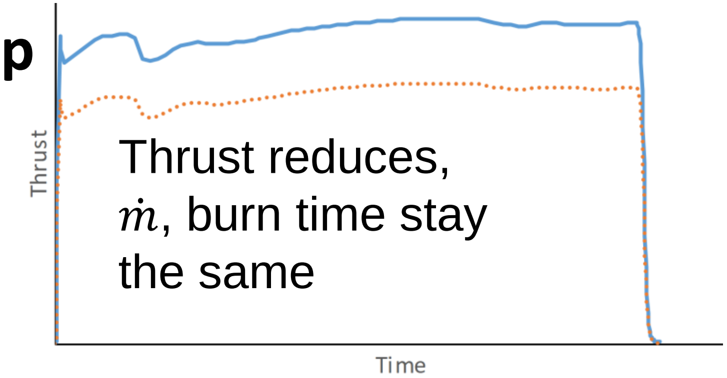
$$\frac{P_3}{P_2} = \left(\frac{A_{t3}}{A_{t2}}\right)^{-\frac{1}{1-n}} \qquad \frac{F_3}{F_2} = \frac{\dot{m}_3}{\dot{m}_2}\frac{Isp_3(\epsilon_3)}{Isp_2(\epsilon_2)}$$

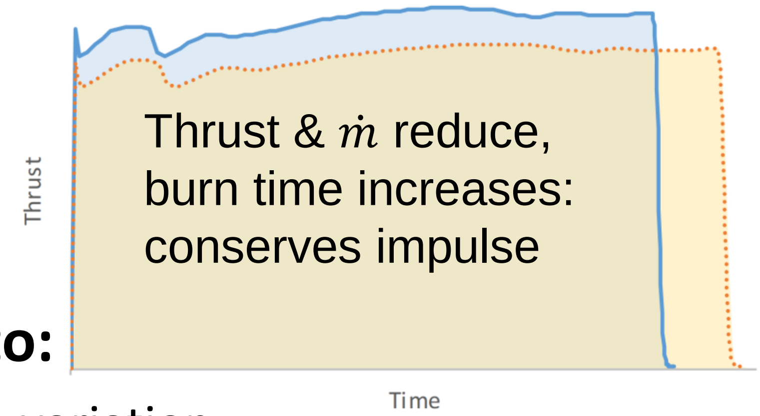




3: Lander mission driven by variation

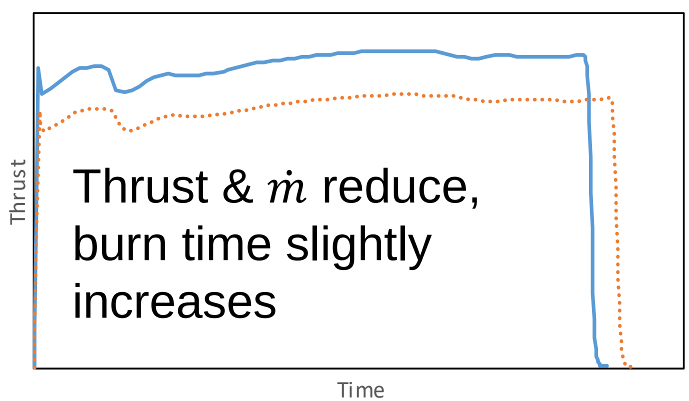
- **Isp**



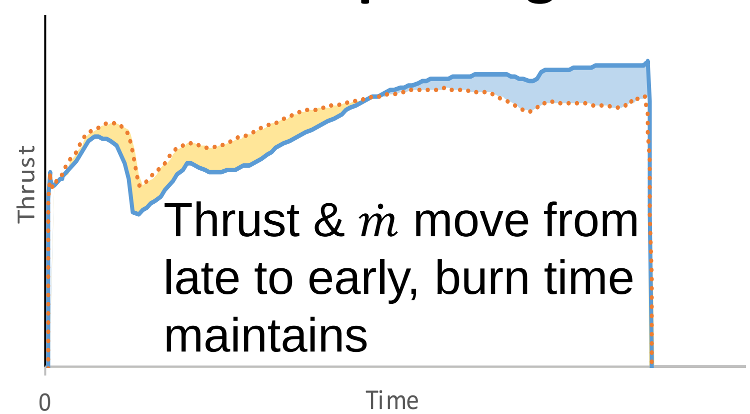


- **Burn time, due to:**
  - intrinsic burn rate variation
  - Propellant Mean Bulk Temperature (PMBT)

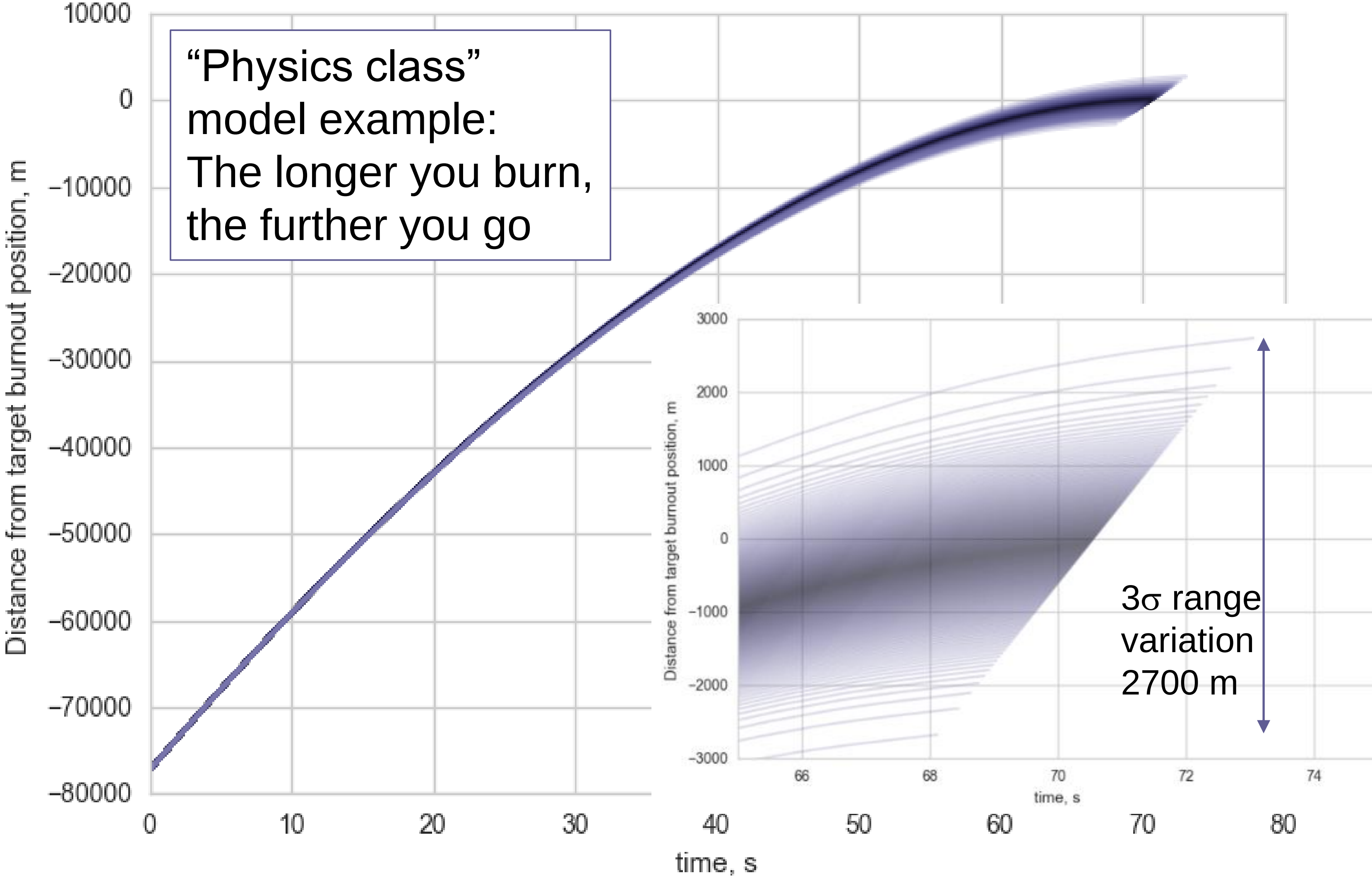
- **Propellant mass**



- **Thrust Shape Magnitude**



How does burn time affect mission performance?



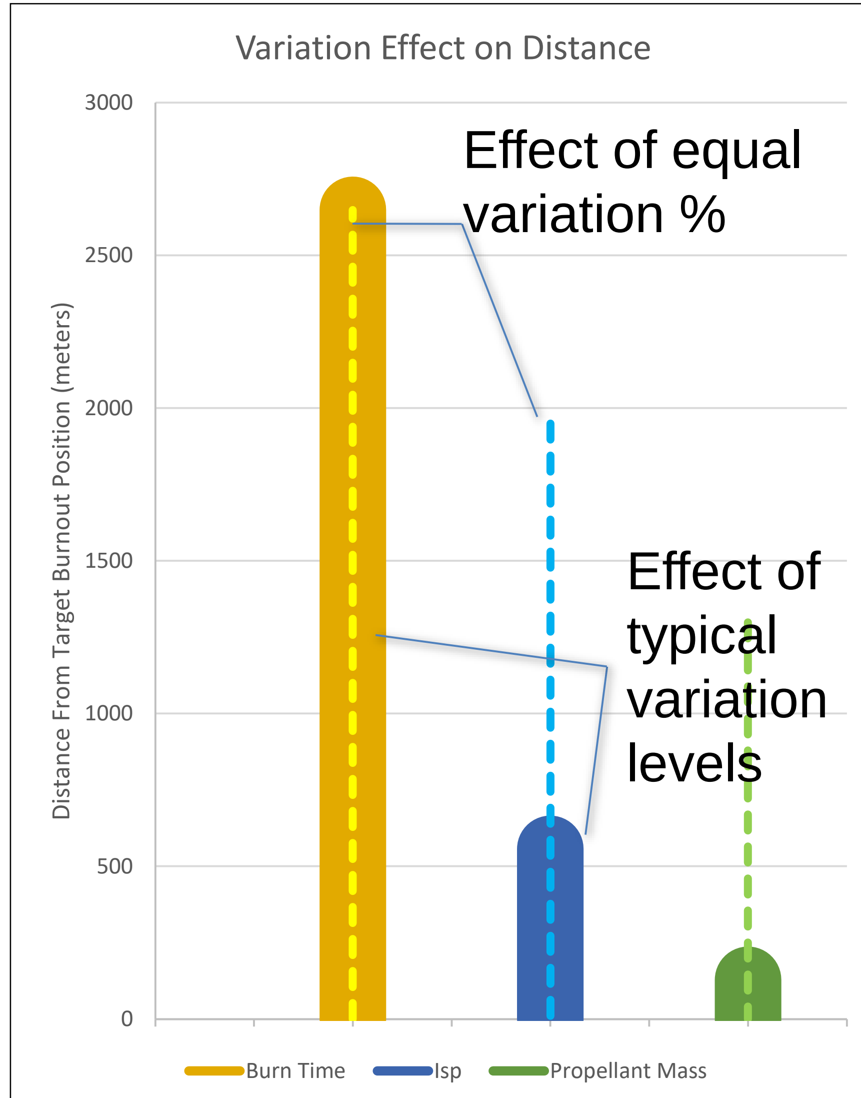
Variation Effects on Distance

- **Isp proportional to ΔV**

$$\Delta V = \boldsymbol{Isp}\ \ln\left(\frac{1}{\frac{m_{prop}}{m_{final}}+1}\right)$$

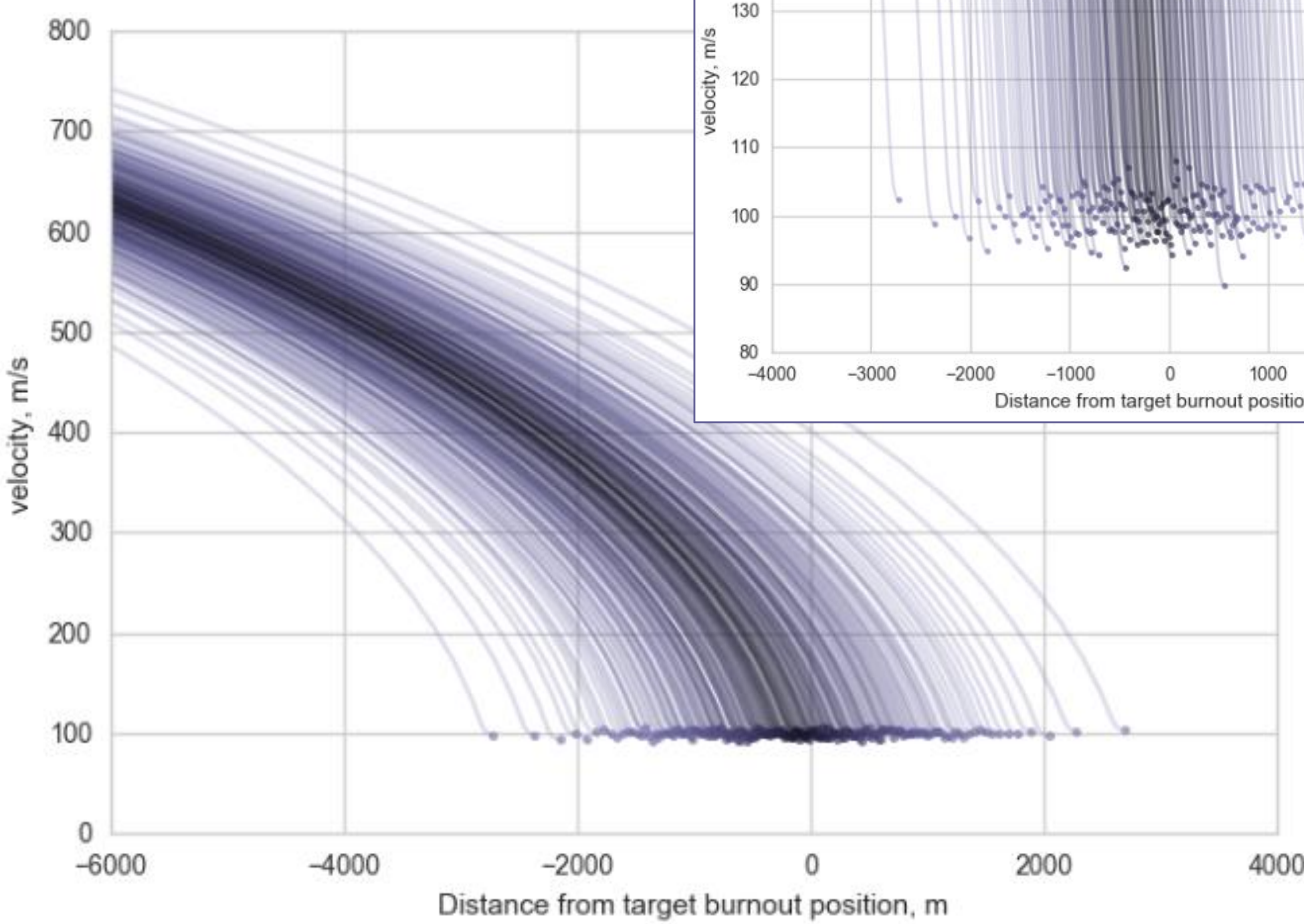
- **Burn time proportional to range traveled; Isp affects less**

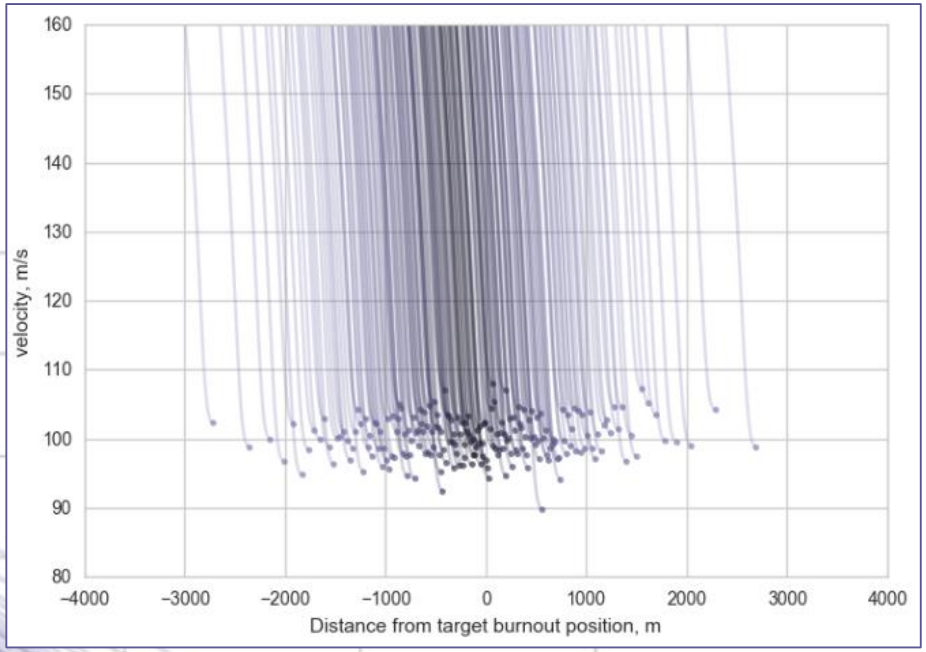
$$\Delta x = \boldsymbol{t_b}\left(V_0 - \boldsymbol{Isp}\left[\frac{m_{final}}{m_{prop}}\ \ln\left(\frac{1}{\frac{m_{prop}}{m_{final}}+1}\right)+1\right]\right)$$



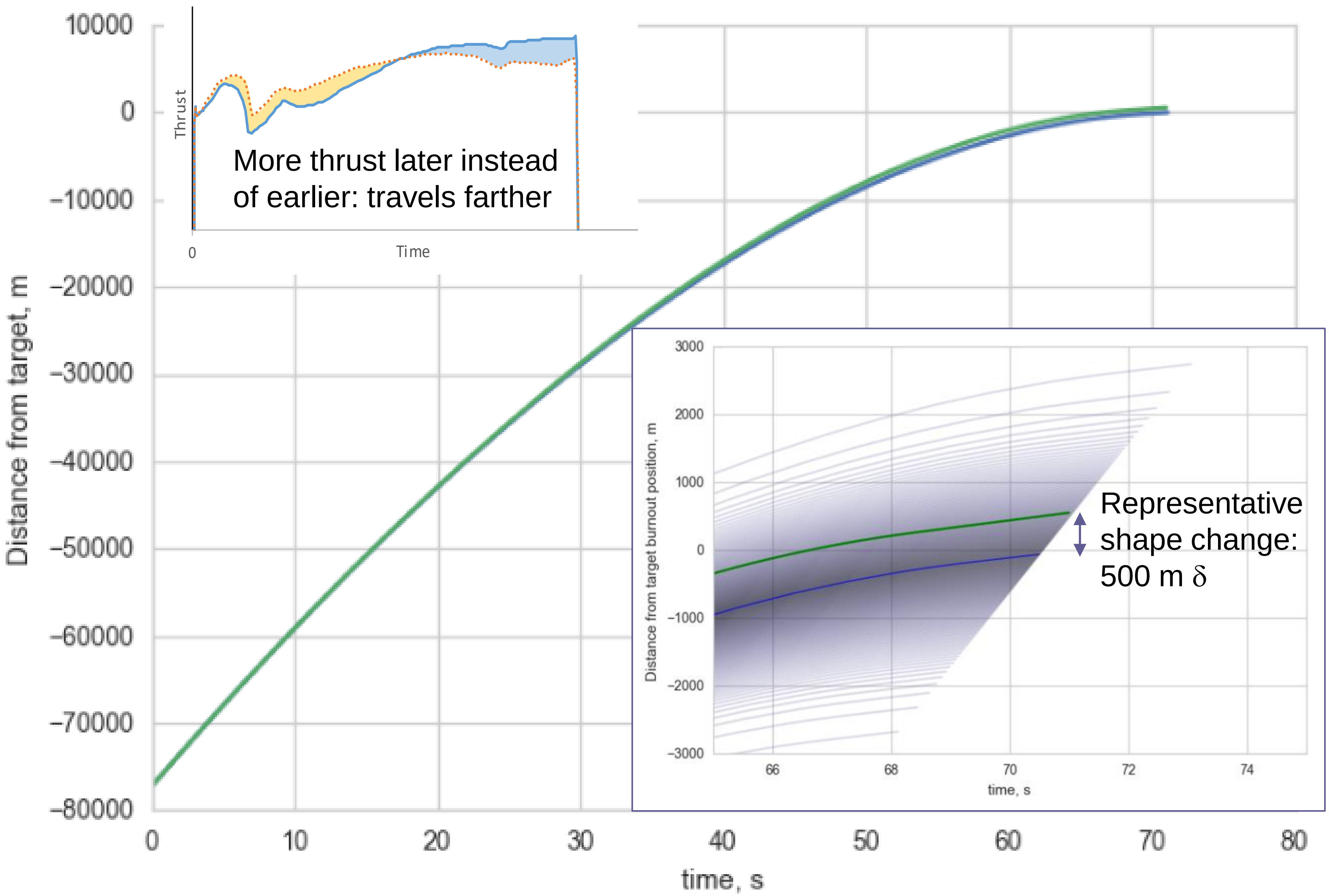
Example: Combined effect of burn rate and Isp

- **Range footprint dominated by burn time variation (2700 m)**





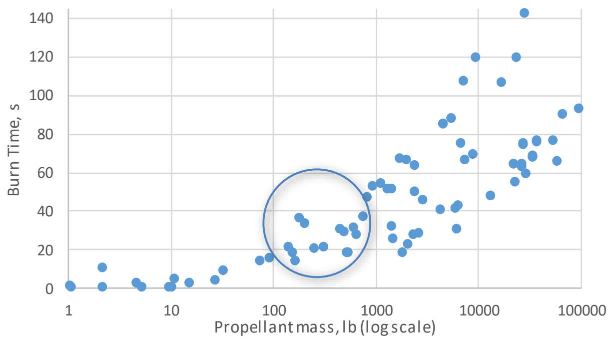
Thrust shape effect on range estimation



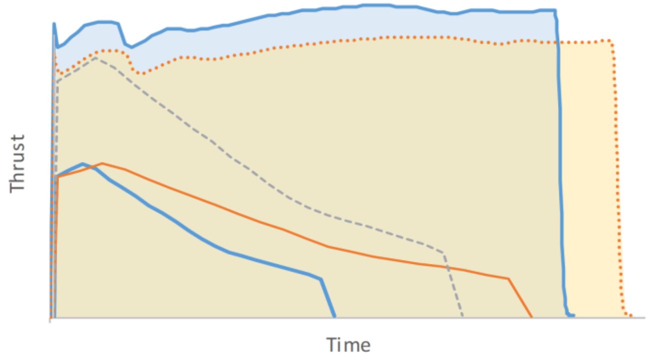
Conclusions

- **For lander de-orbit missions**
  - Variations matter
  - For planning estimates, a bit of margin on burn time variation should cover range variation from all sources
- **For loads-driven missions like Mars ascent**
  - Use dimensions, pressure, and thrust to extend burn times
  - Scale a desired regressive shape for other sizes

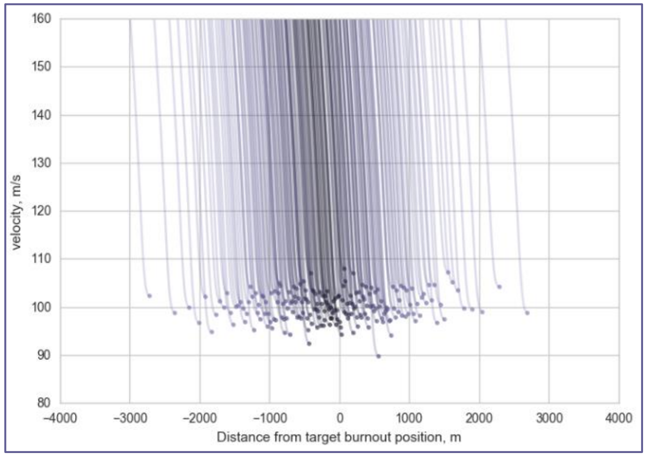
Questions?



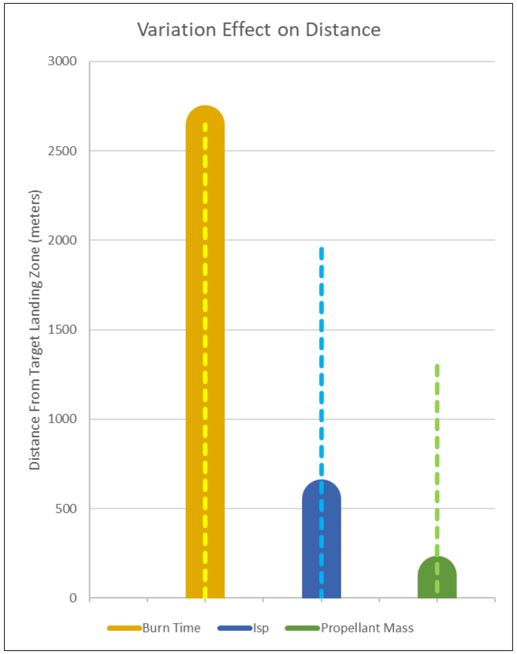
Burn Time, s
140
120
100
80
60
40
20
0
1
10
100
1000
10000
100000
Propellant mass, lb (log scale)



Thrust
Time



velocity, m/s
160
150
140
130
120
110
100
90
80
−4000
−3000
−2000
−1000
0
1000
2000
3000
4000
Distance from target burnout position, m



Variation Effect on Distance
Distance From Target Landing Zone (meters)
3000
2500
2000
1500
1000
500
0
Burn Time
Isp
Propellant Mass