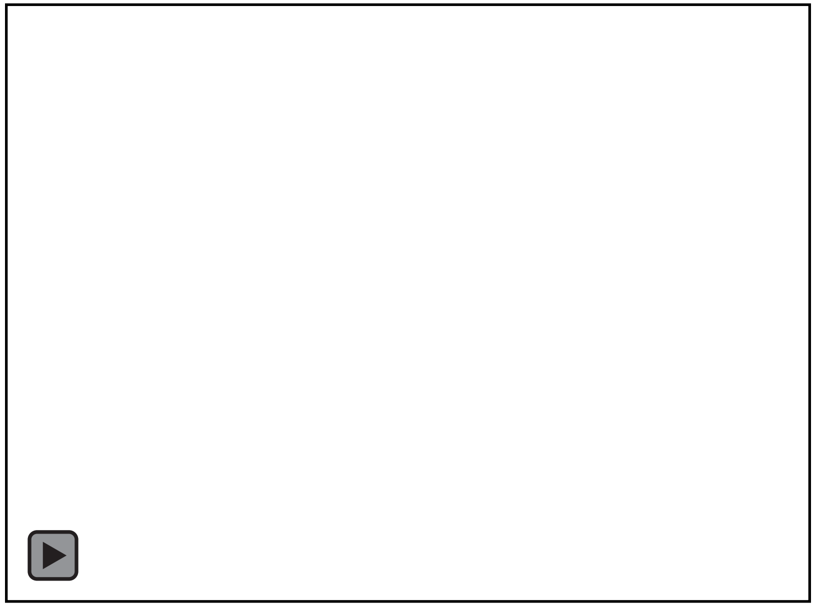
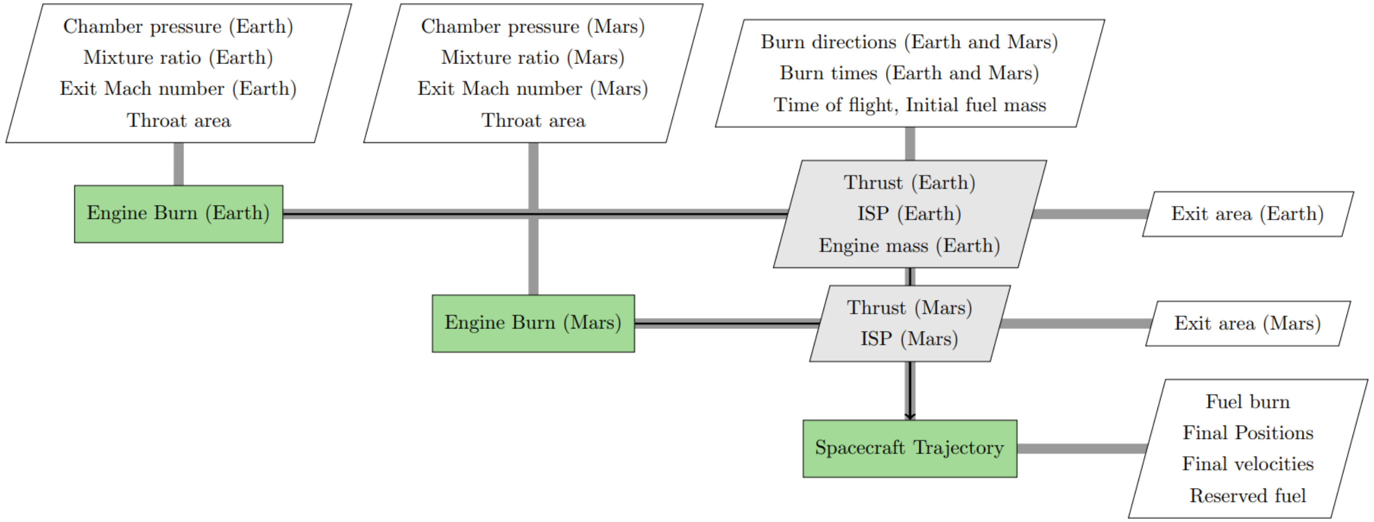
A Coupled Spacecraft System and Trajectory Optimization Framework Using GMAT and OpenMDAO





Gage Harris, Ping He, and Ossama Abdelkhalik

Aerospace Engineering Department, Iowa Stata University

Overview

- Background
- Method
  - GMAT
  - Rocket Engine Model
  - OpenMDAO
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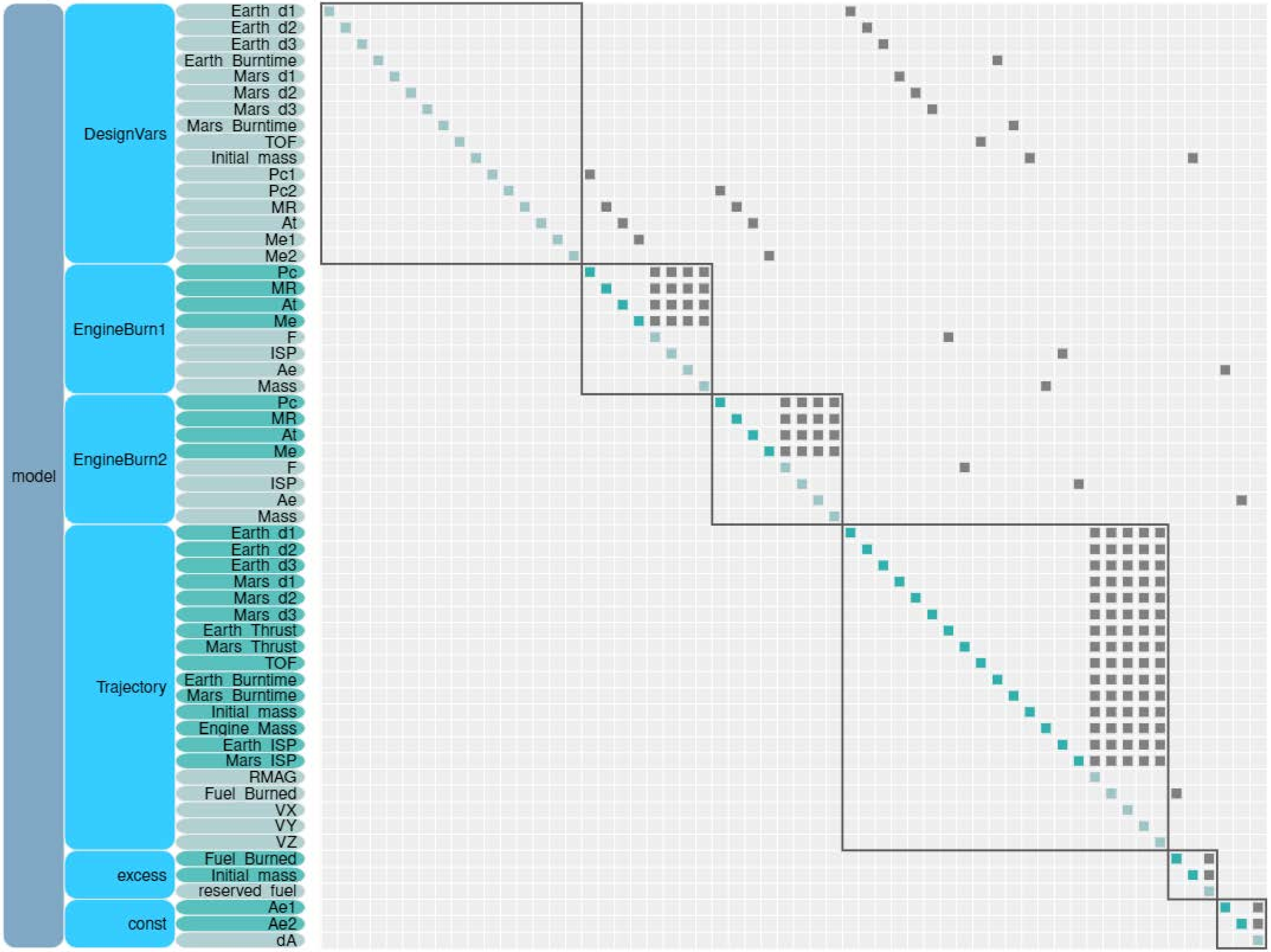
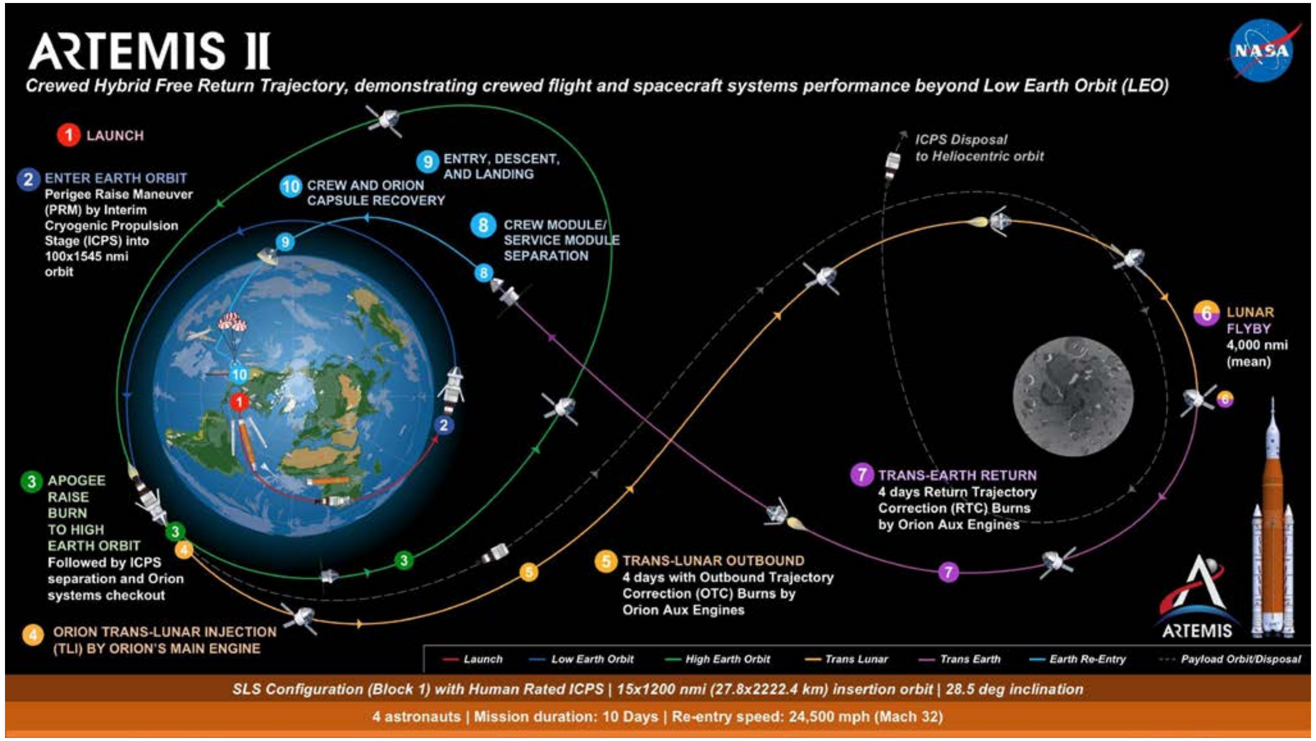
- Results
  - Single Point optimization
  - Multi-point optimization
- Summary
- Future Work



Space missions need to design spacecraft system and trajectory



NASA Artemis II Mission Design

CubeSat Design

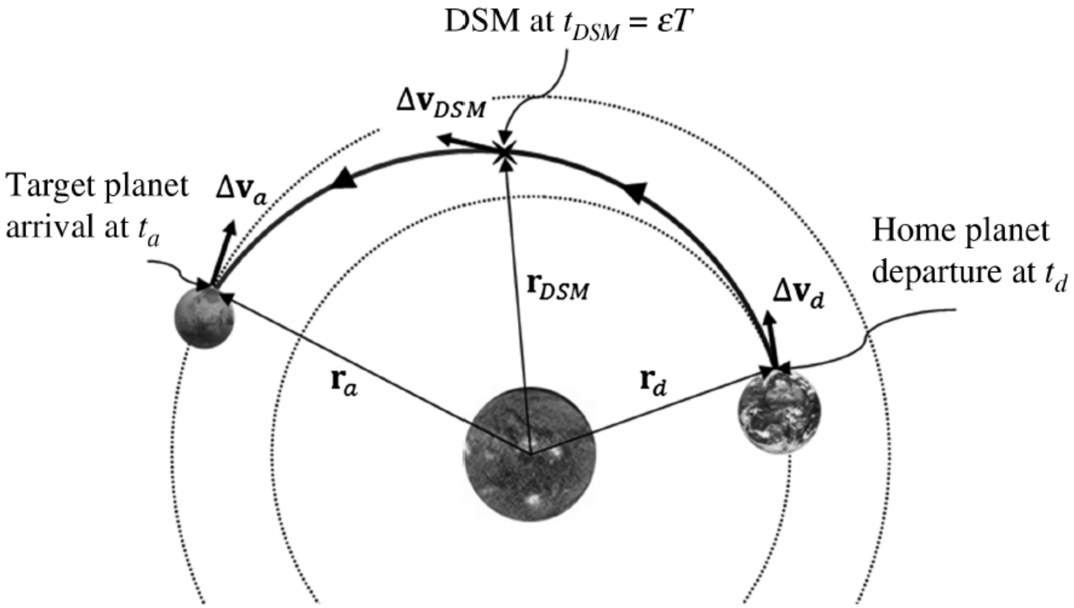
https://techcrunch.com/2019/07/29/tesseract-makes-spacecraft-propulsion-smaller-greener-stronger/

https://exploredeepspace.com/deep-space-mission/artemis-missions/

Existing studies optimized trajectory and systems separately

Recent studies in this field:

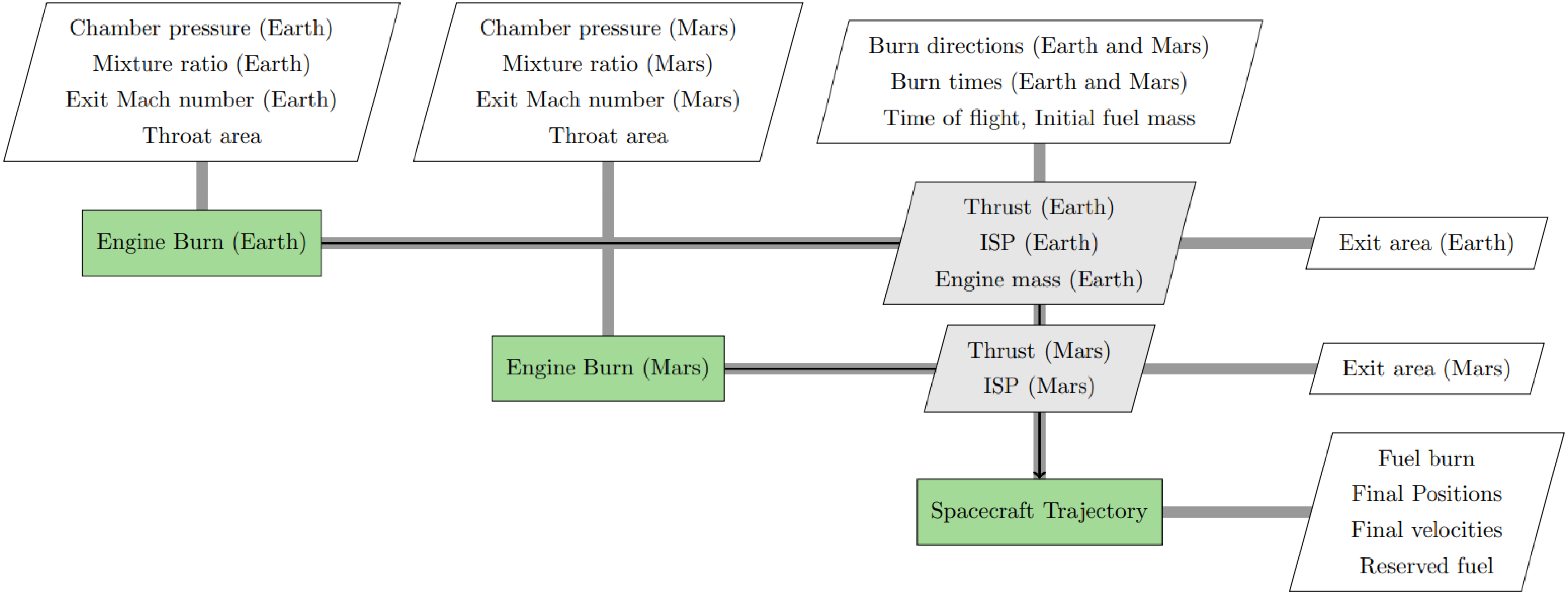
- Frank et al. [2017] optimize rocket engine only.
- Hwang et al. [2014] optimize multiple systems.
- Izzo et al. [2015] optimize trajectory only.
- Darani & Abdelkhalik [2018] optimize trajectory only.
- Lamroqere et al. [2014] optimize trajectory only.



Three-impulse transfer orbit
(Darani & Abelkhalik [2018])

Objective: Use OpenMDAO to optimize the trajectory and system simultaneously with a high number of design variables.

We developed a coupled spacecraft system and trajectory optimization framework using OpenMDAO



We used NASA's General Mission Analysis Tool GMAT to compute the spacecraft trajectory

- Open-source package developed by NASA
- Configure spacecrafts, force models, and other spacecraft hardware
- Mission sequence to simulate spacecraft
- User interaction:
  - Graphical user interface (GUI)
  - **Python API**
- Fixed mission sequence created
- Interplanetary mission to Mars used in this study

	Function/Variable	Description	Quantity
Inputs	F^{Earth}	Thrust magnitude Earth (N)	1
	$I_{\text{sp}}^{\text{Earth}}$	Specific Impulse Earth (s)	1
	d_n^{Earth}	Thrust directions Earth	3
	T_b^{Earth}	Earth burn time (s)	1
	F^{Mars}	Thrust magnitude Mars (N)	1
	$I_{\text{sp}}^{\text{Mars}}$	Specific Impulse Mars (s)	1
	d_n^{Mars}	Thrust directions Mars	3
	T_b^{Mars}	Mars burn time (s)	1
	TOF	Time of flight (day)	1
	m^0	Fuel mass before flight (kg)	1
	m^{engine}	Mass of engine (kg)	1
		Total Inputs	**15**
Outputs	R_{mag}	Center of Mars to spacecraft (km)	1
	m^{excess}	Excess fuel (kg)	1
	$\lvert V_{\text{x,y,z}} \rvert \leq 0.01$	Spacecraft relative velocity (km/s)	3
	m_b	Fuel burn (kg)	1
		Total Outputs	**6**

https://sourceforge.net/projects/gmat/files/GMAT/GMAT-R2016a/

We developed an analytical rocket engine model (1/2)

- Spacecraft system being coupled
- Implemented as a rocket engine model in Python
- Exit Mach number used as input instead of exit area. Allows engine model to be explicit.
- Exit area computed as output and constrained during optimization

	Function/Variable	Description	Quantity
Inputs	P_c	Chamber pressure (MPa)	1
	M_R	Mixture ratio	1
	A_t	Nozzle throat area (m^2)	1
	M_e	Exit Mach number	1
		Total Inputs	**4**
Outputs	F	Thrust magnitude (N)	1
	I_{sp}	Specific Impulse (s)	1
	m^{engine}	Mass of engine (kg)	1
	A_e	Nozzle exit area (m^2)	1
		Total Outputs	**4**

We developed an analytical rocket engine model (2/2)

- Surrogate model created to compute
  - Chamber temperature
  - Specific heat ratio
  - Gas constant
- 500 sample points generated using various chamber pressure and mixture ratios
- Rocket Propulsion Analysis tool (RPA) was run with each sample point
- RBF Surrogate model was trained with RPA outputs
- Assumed constant composition in the nozzle

$$\dot{m} = \frac{A_t P_c}{\sqrt{T_c}} \sqrt{\frac{\gamma}{R}} \left(\frac{\gamma+1}{2}\right)^{\frac{-\gamma-1}{2\gamma-2}}$$

$$\frac{A_e}{A_t} = \left(\frac{\gamma+1}{2}\right)^{\frac{-\gamma-1}{2\gamma-2}} \left(1 + \frac{\gamma-1}{2} M_e^2\right)^{\frac{\gamma+1}{2\gamma-2}} M_e^{-1}$$

$$\frac{T_e}{T_c} = \left(1 + \frac{\gamma-1}{2} M_e^2\right)^{-1}$$

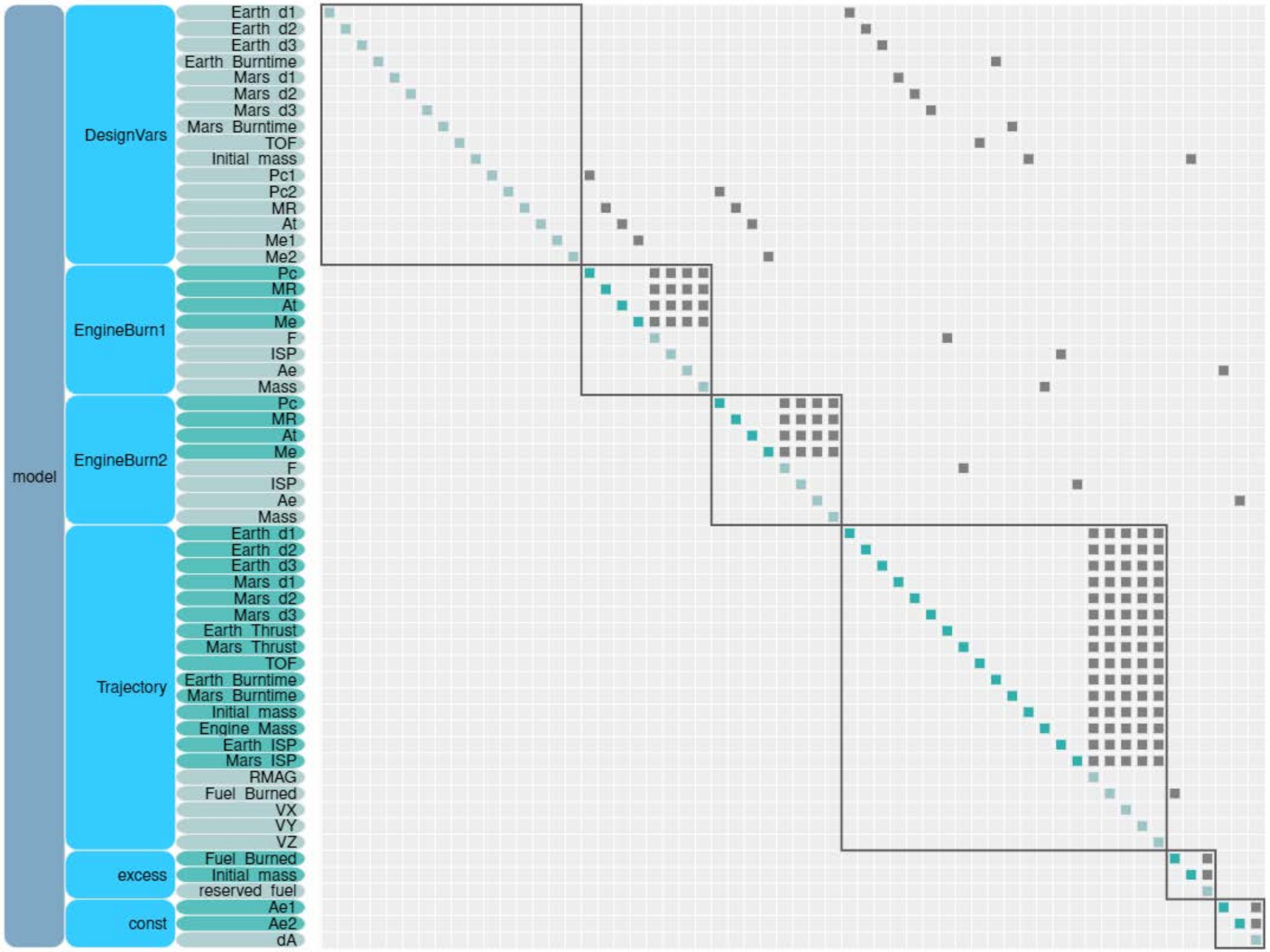
$$\frac{P_e}{P_c} = \left(1 + \frac{\gamma-1}{2} M_e^2\right)^{\frac{-\gamma}{\gamma-1}}$$

$$v_e = M_e \sqrt{\gamma R T_e}$$

$$F = \dot{m} \lambda_b v_e + (P_e - P_o) A_e$$

$$m^{\text{engine}} = 1.866 \times 10^{-10} F^2 + 0.00130 F + 77.4$$

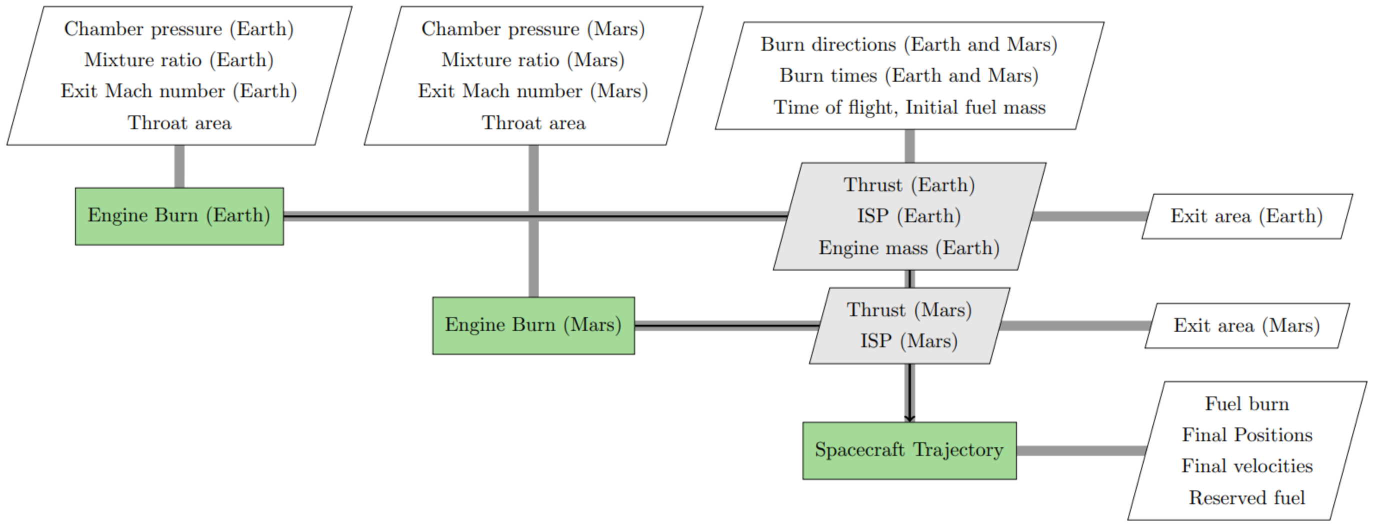
The N2 diagram of the coupled optimization



https://openmdao.org/

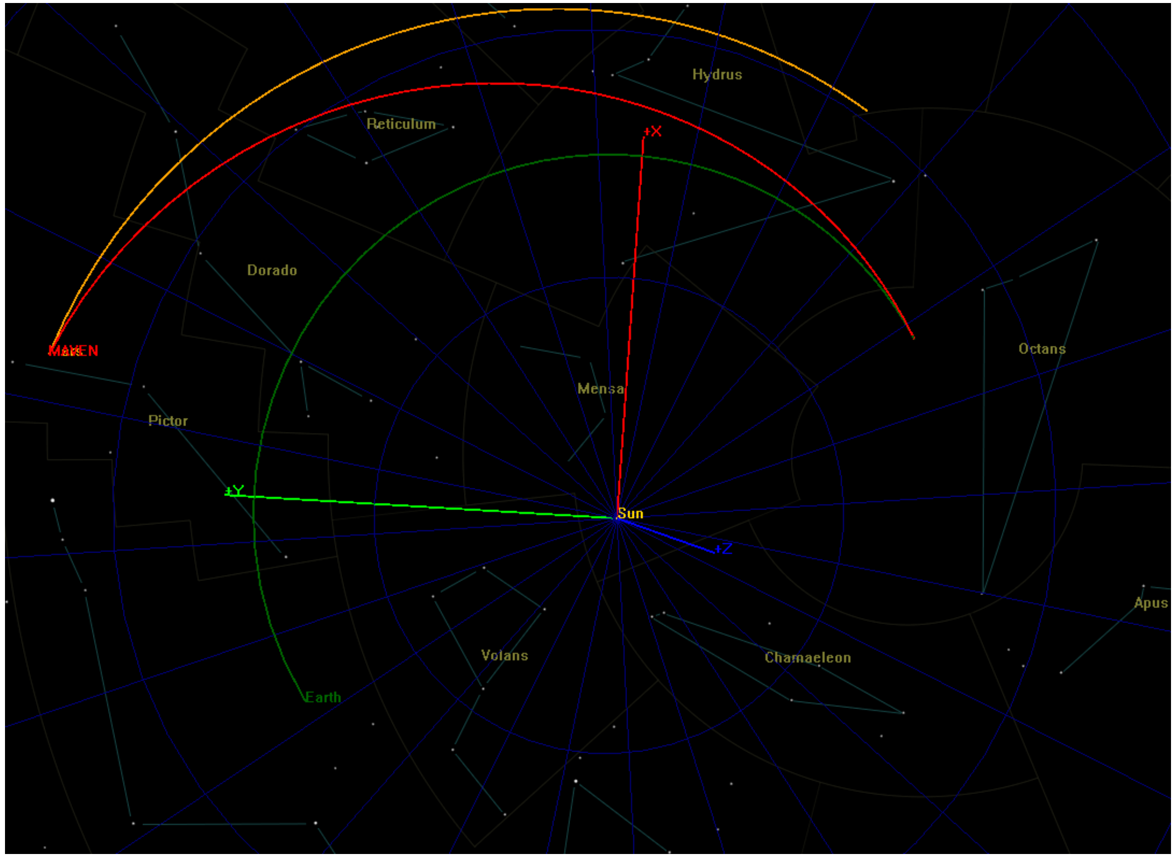
We developed a Spacecraft Mission Optimization Tool called SMOT

- We developed a python interface combing GMAT and OpenMDAO
- GMAT acts as trajectory analysis tool
- OpenMDAO acts as the optimizer
  - All components defined
  - Optimizer: SNOPT
- Python runscript is how to interact with SMOT
  - Tells OpenMDAO which trajectory component or mission to optimize
  - Set up initial conditions
  - Modify constraints
- Currently only one trajectory component in SMOT
  - Interplanetary mission to Mars



Results

- Mission set up:
  - Two impulsive burns
  - No fly-bys considered
  - Sun only gravitational body
- Three different departure days studied
  - May 27th , 2020
  - July 27th , 2020
  - September 8th , 2020
- Two optimization types
  - Single point
  - Multi-point
- Objective to reach Mars with smallest amount of fuel burned
- Constraints
  - 3000 km from Mars
  - 0.01 km/s relative velocity
  - Exit areas equal



GMAT GUI

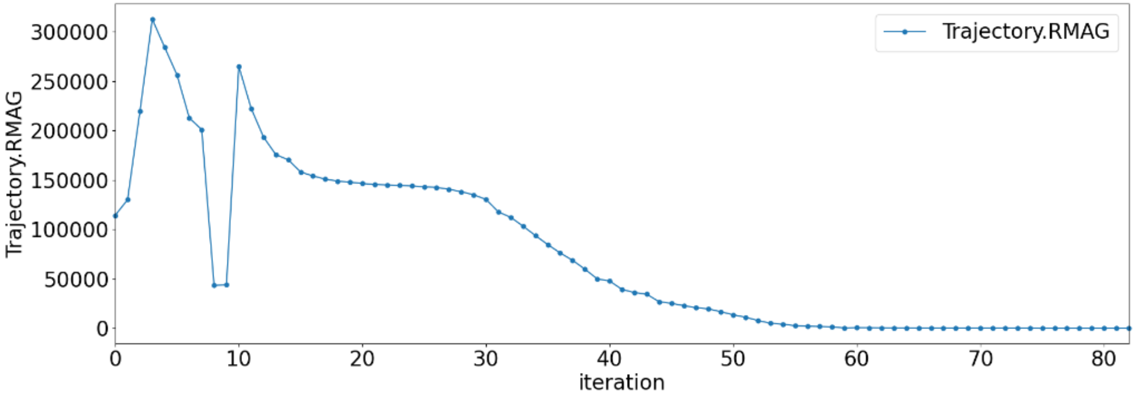
Single Point Optimizations

- Five different optimization configurations
- [1] Trajectory
  - ISP fixed; all other engine parameters neglected
- [2] Fixed Engine Geometry (FEG)
  - Engine geometry fixed; chamber pressure allowed to change
- [3] Fixed Engine Geometry MR
  - Engine geometry fixed; chamber pressure and mixture ratio allowed to change
- [4] Coupled
  - Coupled trajectory and full engine optimization, but mixture ratio must be equal
- [5] Coupled MR
  - Coupled trajectory and full engine optimization
- Optimization formulation for Coupled MR can be seen

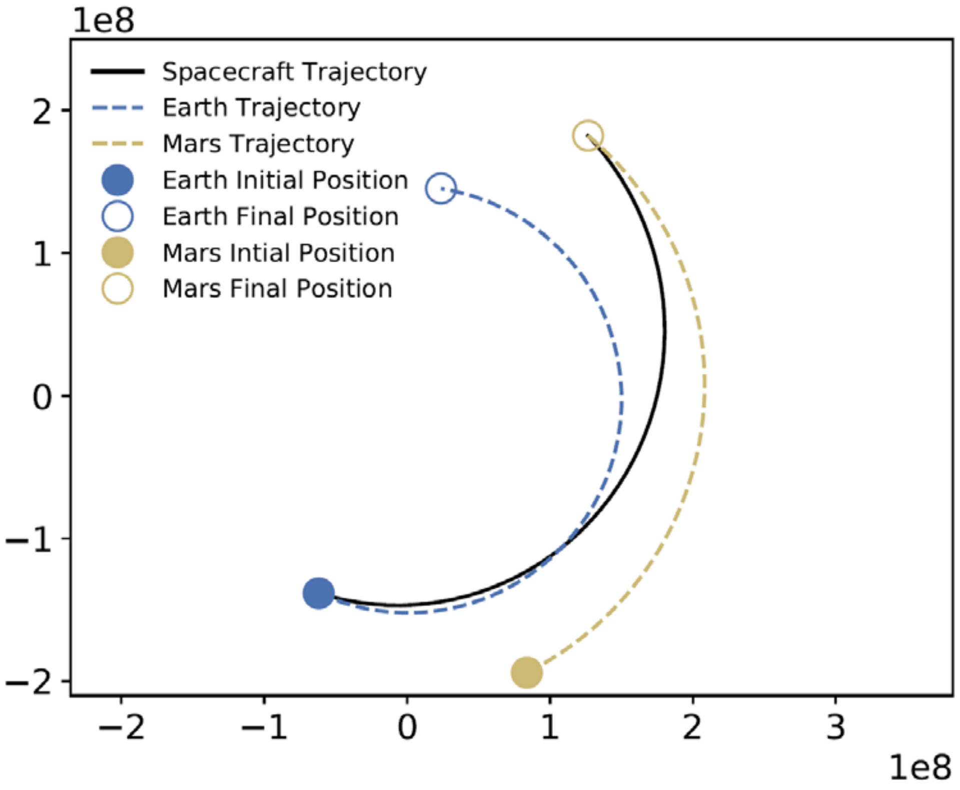
	Function/Variable	Description	Quantity
Minimize	m_b	Total fuel burned (kg)	1
w.r.t	P_c^{Earth}	Chamber pressure (MPa) Earth	1
	P_c^{Mars}	Chamber pressure (MPa) Mars	1
	M_R^{Earth}	Mixture ratio Earth	1
	M_R^{Mars}	Mixture ratio Mars	1
	A_t	Throat area (m^2)	1
	M_e^{Earth}	Exit Mach number Earth	1
	M_e^{Mars}	Exit Mach number Mars	1
	d_n^{Earth}	Thrust directions Earth	3
	T_b^{Earth}	Earth burn time (s)	1
	d_n^{Mars}	Thrust directions Mars	3
	T_b^{Mars}	Mars burn time (s)	1
	TOF	Time of flight (day)	1
	m^0	Fuel mass before flight (kg)	1
		Total Design Variables	**17**
Subject to	$R_{\text{mag}} \leq 3000$	Center of Mars to spacecraft (km)	1
	$m^{\text{excess}} \geq 200$	Forced excess fuel (kg)	1
	$\text{V}\lvert V_{\text{x,y,z}} \rvert \leq 0.01$	Spacecraft relative velocity (km/s)	3
	$A_e^{\text{Earth}} = A_e^{\text{Mars}}$	Earth-Mars exit areas equal	1
		Total Constraints	**6**

Coupled optimization used 25% less fuel than the trajectory-only case: May 27, 2020

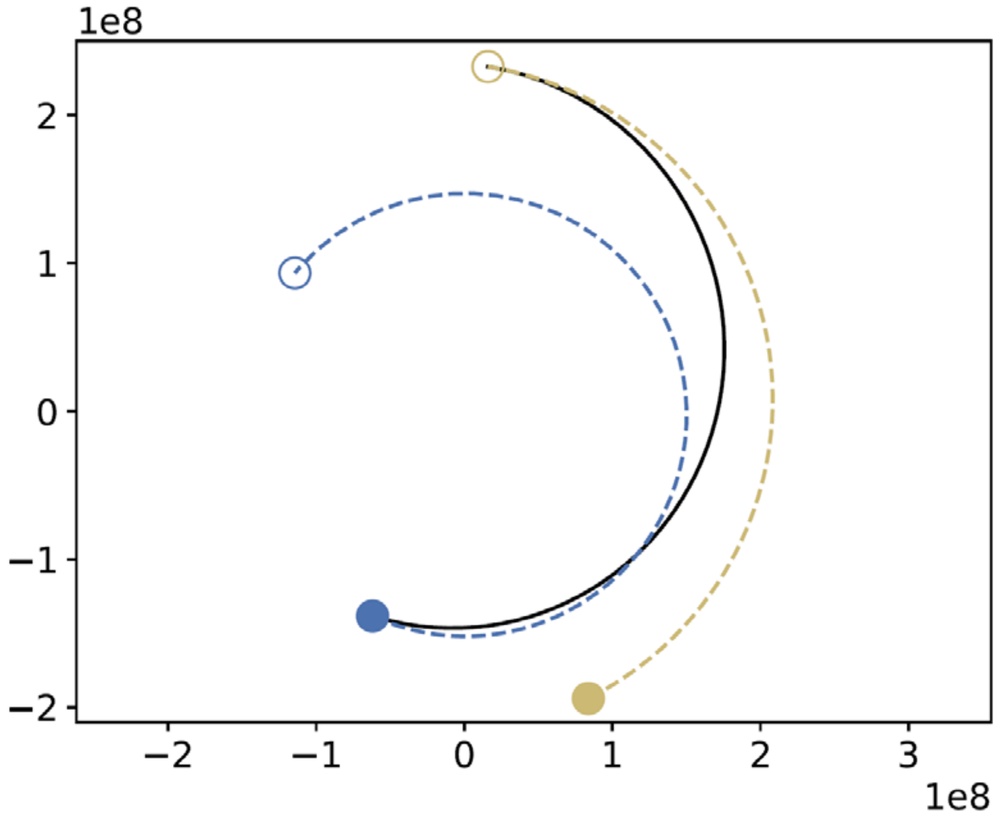
Function/Variable	Trajectory	FEG	FEG MR	Coupled	Coupled MR
m_b (kg)	**5150**	**5250**	**4819**	**4071**	**4014**
R_{mag} (km)	3004	3025	3000	2807	2856
m^{excess} (kg)	200	200	200	200	200
$\lvert V_{\mathrm{x}}\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert V_{\mathrm{y}}\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert V_{\mathrm{z}}\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert A_e^{\mathrm{Earth}} - A_e^{\mathrm{Mars}}\rvert$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$
F_{Earth} (N)	51200	38580	36540	31800	31520
F_{Mars} (N)	15730	38580	6974	7644	6399
I_{sp} (s)	445.7	441.8	460.0	477.5	480.6
A^e/A_t	61	61	61	126.9	153.4
m^{engine} (kg)	206	206	206	118.9	118.6



Comparison between the baseline and optimized trajectories: May 27, 2020



Initial conditions



Coupled MR trajectory

Earth Thrust: **56360** (N)
Mars Thrust: **20590** (N)
TOF: **200** (d)
Fuel burn: **9000** (kg)

Earth Thrust: **31520** (N)
Mars Thrust: **6399** (N)
TOF: **258.6** (d)
Fuel burn: **4014** (kg)

Coupled optimizations outperformed the trajectory-only case for each departure date

Coupled optimization uses 28% less fuel than the trajectory only case: July

Function/Variable	Trajectory	FEG	FEG MR	Coupled	Coupled MR
m_b (kg)	**3258**	**3348**	**3076**	**2474**	**2462**
R_{mag} (km)	3200	3234	3120	3000	2940
m^{excess} (kg)	200	200	200	200	200
$\lvert V_x\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert V_y\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert V_z\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert A_e^{\text{Earth}} - A_e^{\text{Mars}}\rvert$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$
F_{Earth} (N)	29860	23040	21750	17500	17450
F_{Mars} (N)	15180	23040	38390	7014	7320
I_{sp} (s)	445.7	439.4	459.6	487.3	488.6
A^e/A_t	61	61	61	212.9	237.1
m^{engine} (kg)	206	206	206	100.2	100.1

Coupled optimization uses 22% less fuel than the trajectory only case: Sep

Function/Variable	Trajectory	FEG	FEG MR	Coupled	Coupled MR
m_b (kg)	**7103**	**7192**	**6626**	**5739**	**5705**
R_{mag} (km)	3000	3024	3003	2439	3000
m^{excess} (kg)	200	200	200	200	200
$\lvert V_x\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert V_y\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert V_z\rvert$ (km/s)	<0.01	<0.01	<0.01	<0.01	<0.01
$\lvert A_e^{\text{Earth}} - A_e^{\text{Mars}}\rvert$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$
F_{Earth} (N)	55240	55980	53880	45630	45450
F_{Mars} (N)	13200	55980	56210	9083	7670
I_{sp} (s)	445.7	443.4	459.0	473.9	475.3
A^e/A_t	61	61	61	112.1	112.5
m^{engine} (kg)	206	206	206	137.1	136.9

Trajectory and Engine separate optimizations

- Final single point optimization to be compared
- How traditional trajectory and engine optimizations are done
- Trajectory optimization run first
- Engine optimization constrained given thrust magnitudes
- Fuel burned calculated using mass flow rate and burn time from trajectory optimization

	Function/Variable	Description	Quantity
Maximize	ISP_{av}	Average Specific Impulse (s)	1
w.r.t	P_c^{Earth}	Chamber pressure (MPa) Earth	1
	P_c^{Mars}	Chamber pressure (MPa) Mars	1
	M_R^{Earth}	Mixture ratio Earth	1
	M_R^{Mars}	Mixture ratio Mars	1
	A_t	Throat area (m^2)	1
	M_e^{Earth}	Exit Mach number Earth	1
	M_e^{Mars}	Exit Mach number Mars	1
		Total Design Variables	**7**
Subject to	F_{Earth} = Trajectory	Center of Mars to spacecraft (km)	1
	F_{Mars} = Trajectory	Earth-Mars exit areas equal	1
	$A_e^{\text{Earth}} = A_e^{\text{Mars}}$	Earth-Mars exit areas equal	1
		Total Constraints	**3**

$$m_b(kg) = \dot{m}_{\text{Earth}} T_b^{\text{Earth}} + \dot{m}_{\text{Mars}} T_b^{\text{Mars}}$$

Coupled optimizations outperformed the trajectory and engine separate optimizations

- Results are between the FEG and FEG MR case for each date
- Both coupled optimizations are far superior
- Coupled MR shown for comparison
- Clearly shows the benefit of coupling both disciplines
  - DV's and constraints are the same for both shown optimizations
  - Only difference is the coupling

Date	Separate (kg)	Coupled MR (kg)	Percent difference
May 27^{th}	4855	4014	19.0
July 27^{th}	3020	2462	20.4
September 8^{th}	6715	5705	16.3

Multi-point optimization

- Single point optimization worked well for specific departure date
- Multi-point combines all three departure dates
- Objective function is average fuel burned from all three dates
- Formulation similar to Coupled MR of single point optimization
- Each date has it own design variables and constraints, except throat area

	Function/Variable	Description	Quantity
Minimize	$(m_b^{\text{May}} + m_b^{\text{Jul}} + m_b^{\text{Sep}})/3$	Averaged fuel burned (kg) for the May, Jul, & Sep simulations	1
w.r.t.	P_c^{Earth}	Chamber pressure (MPa) Earth	3
	P_c^{Mars}	Chamber pressure (MPa) Mars	3
	M_R^{Earth}	Mixture ratio Earth	3
	M_R^{Mars}	Mixture ratio Mars	3
	A_t	Throat area (m^2)	1
	M_e^{Earth}	Exit Mach number Earth	3
	M_e^{Mars}	Exit Mach number Mars	3
	d_n^{Earth}	Thrust directions Earth	9
	T_b^{Earth}	Earth burn time (s)	3
	d_n^{Mars}	Thrust directions Mars	9
	T_b^{Mars}	Mars burn time (s)	3
	TOF	Time of flight (day)	3
	m^0	Fuel mass before flight (kg)	3
		Total Design Variables	**49**
Subject to	$R_{\text{mag}} \leq 3000$	Center of Mars to spacecraft (km)	3
	$m^{\text{excess}} \geq 200$	Forced excess fuel (kg)	3
	$\lvert V_{\text{x,y,z}} \rvert \leq 0.01$	Spacecraft relative velocity (km/s)	9
	$A_e^{\text{Earth}} = A_e^{\text{Mars}}$	Earth-Mars exit areas equal	3
	$A_e^{\text{Earth}}(\text{May}) = A_e^{\text{Earth}}(\text{Jul})$	May-Jul exit areas equal	1
	$A_e^{\text{Earth}}(\text{May}) = A_e^{\text{Earth}}(\text{Sep})$	May-Sep exit areas equal	1
		Total Constraints	**20**

Multi-point optimization compromised the engine size between the three departure dates

Function/Variable	May	July	September
m_b (kg)	4190	2692	5707
R_{mag} (km)	3009	2873	3047
m^{excess} (kg)	200.0	200.0	200.0
$\lvert V_{\text{x}}\rvert$ (km/s)	<0.01	<0.01	<0.01
$\lvert V_{\text{y}}\rvert$ (km/s)	<0.01	<0.01	<0.01
$\lvert V_{\text{z}}\rvert$ (km/s)	<0.01	<0.01	<0.01
$\lvert A_e^{\text{Earth}} - A_e^{\text{Mars}}\rvert$	$< 10^{-8}$	$< 10^{-8}$	$< 10^{-8}$
F_{Earth} (N)	35610	26602	45450
F_{Mars} (N)	12071	9518	9563
I_{sp} (s)	475.2	475.4	475.1
A^e/A_t	113.8	113.8	113.8
m^{engine} (kg)	136.9	136.9	136.9

Optimization Type	Throat Area (m^2)	Expansion ratio
May (single point)	0.003222	153.4
July (single point)	0.001752	237.1
September (single point)	0.004701	112.5
Multi-point	0.004698	113.8

Multi-point optimization verification (1/2)

- Engine design for one date might work poorly for another
- This final optimization answers this question
- Uses the engine geometry generated by one date and uses it in coupled optimization of another date
- Appendix shows optimized A_t and M_e for any Coupled MR optimization
- These three values fixed and used in a coupled optimization for other dates

	Function/Variable	Description	Quantity
Minimize	m_b	Total fuel burned (kg)	1
w.r.t	P_c^{Earth}	Chamber pressure (MPa) Earth	1
	P_c^{Mars}	Chamber pressure (MPa) Mars	1
	M_R^{Earth}	Mixture ratio Earth	1
	M_R^{Mars}	Mixture ratio Mars	1
	d_n^{Earth}	Thrust directions Earth	3
	T_b^{Earth}	Earth burn time (s)	1
	d_n^{Mars}	Thrust directions Mars	3
	T_b^{Mars}	Mars burn time (s)	1
	TOF	Time of flight (day)	1
	m^0	Fuel mass before flight (kg)	1
		Total Design Variables	**14**
Subject to	$R_{\text{mag}} \leq 3000$	Center of Mars to spacecraft (km)	1
	$m^{\text{excess}} \geq 200$	Forced excess fuel (kg)	1
	$\text{V}\lvert V_{\text{x,y,z}}\rvert \leq 0.01$	Spacecraft relative velocity (km/s)	3
	$A_e^{\text{Earth}} = A_e^{\text{Mars}}$	Earth-Mars exit areas equal	1
		Total Constraints	**6**

Multi-point optimization verification (2/2)

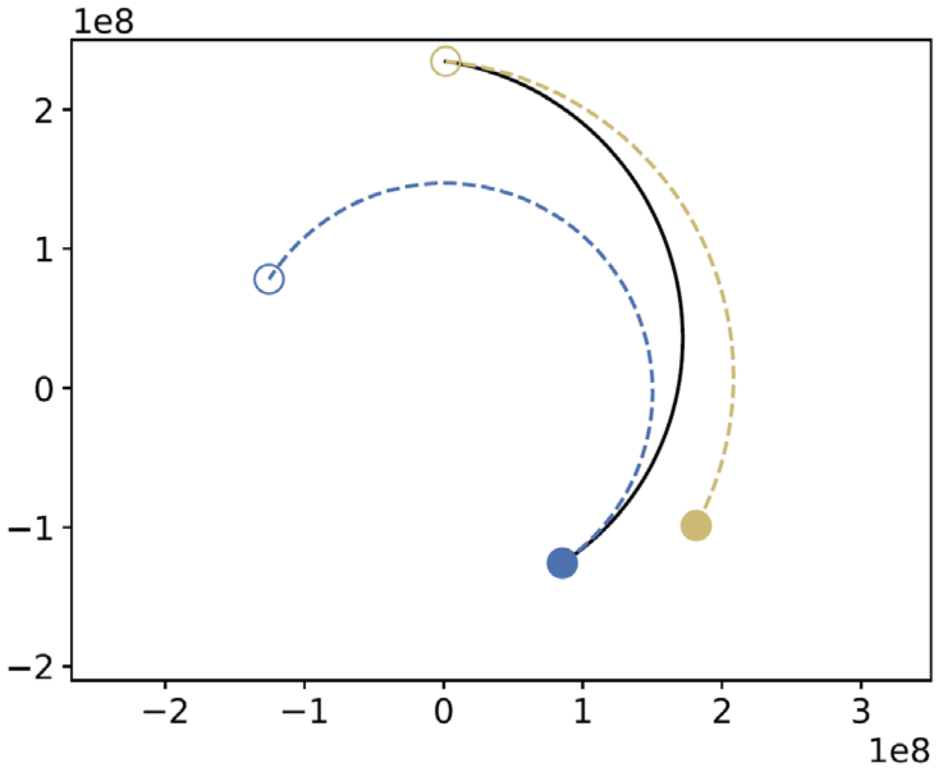
- Diagonal entries are Coupled MR cases for each respective date
- Every other entry was a fixed engine geometry from another date
- NA terms come from infeasible solutions due to DV limit violations
- Namely the Earth burn chamber pressure due to engine sizing
  - Larger thrust = Larger engine size
- Single point optimizations made engines big enough for specific date

Engine Design	Used in May 27^{th}	Used in July 27^{th}	Used in September 8^{th}
May 27^{th}	4014	2588	NA
July 27^{th}	NA	2462	NA
September 8^{th}	4208	2702	5734

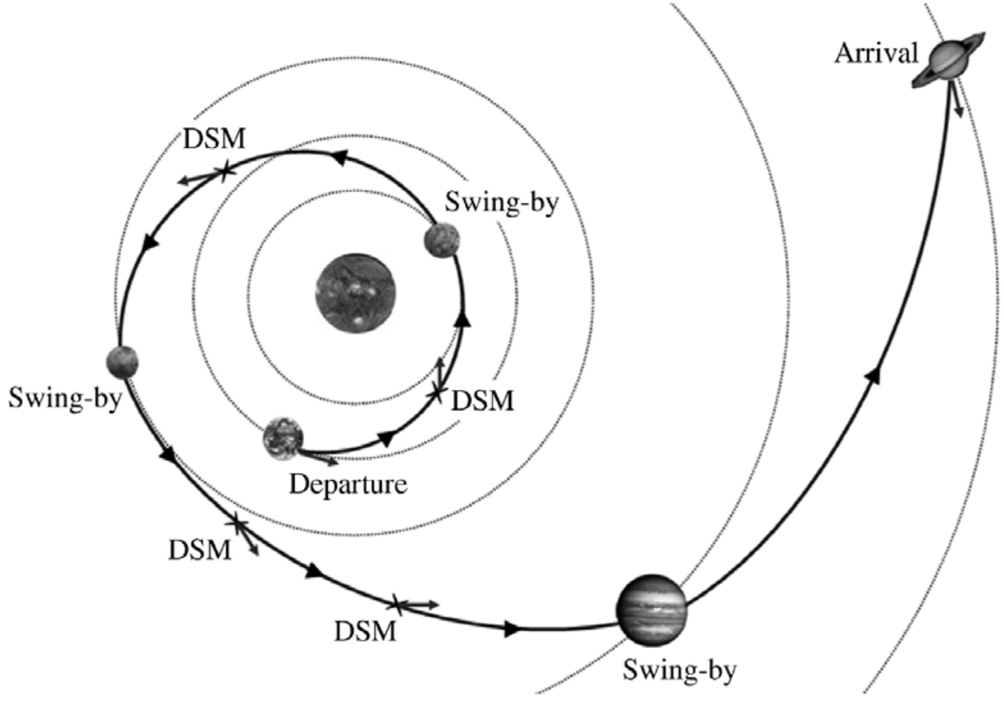
Summary

- Developed an efficient method to simultaneously optimize spacecraft trajectory and systems by combining GMAT and OpenMDAO.
- The more design freedom given to the engine design the more fuel reduction was achieved. The coupled engine and trajectory optimizations obtained 16-20% more fuel burn reduction than the separate optimizations.
- This study can be extended to more spacecraft onboard systems and has the potential to enable larger design freedom for more efficient spacecraft missions.

Future work: develop the capability to consider discrete design variables in OpenMDAO

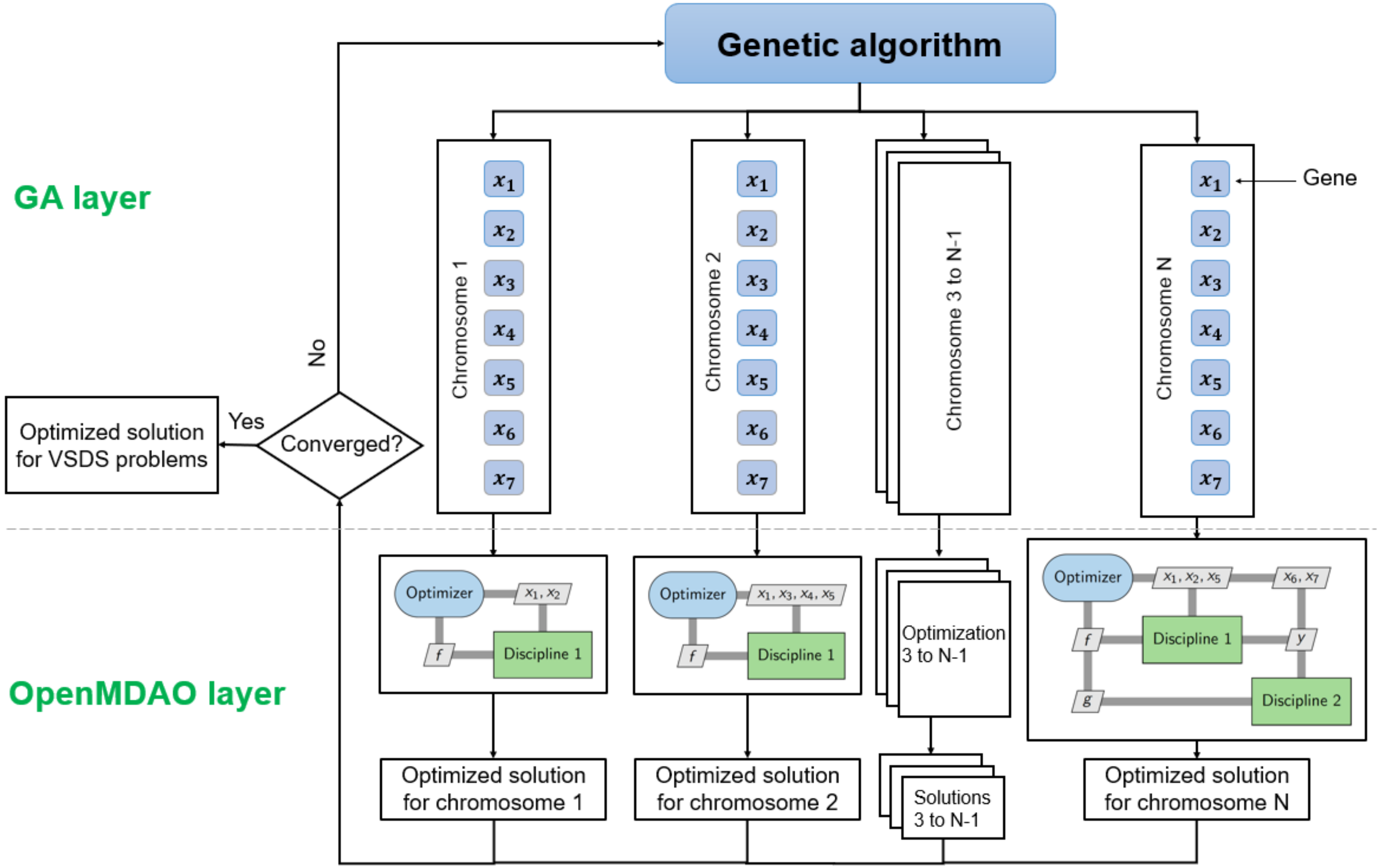


Current mission framework
(No DSMs or gravity assists)

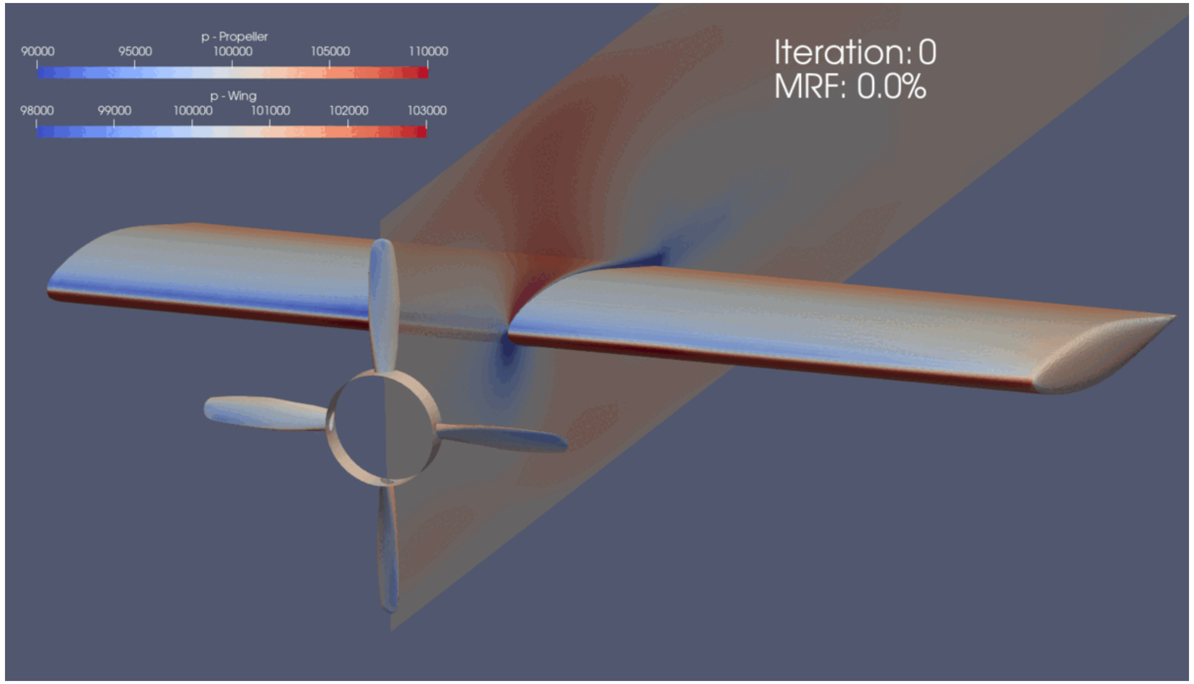


MGA trajectory with free number of DSMs
(Darani & Abelkhalik [2018])

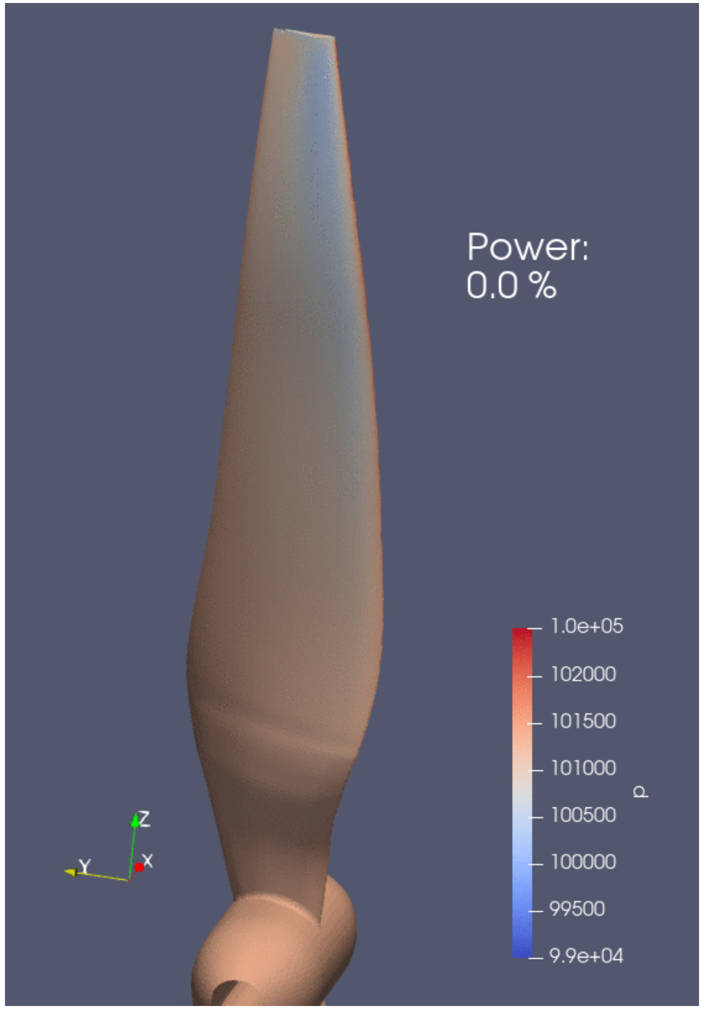
Coupled genetic algorithm and gradient-based optimization framework in OpenMDAO



More MDO results will be presented at Mphys workshop



Wing-propeller multi-component aerodynamic optimization



UAV propeller aerostructural optimization

Thank You