Evolved AI for First-Order Conceptual Missile Design Optimization and Threat Assessment

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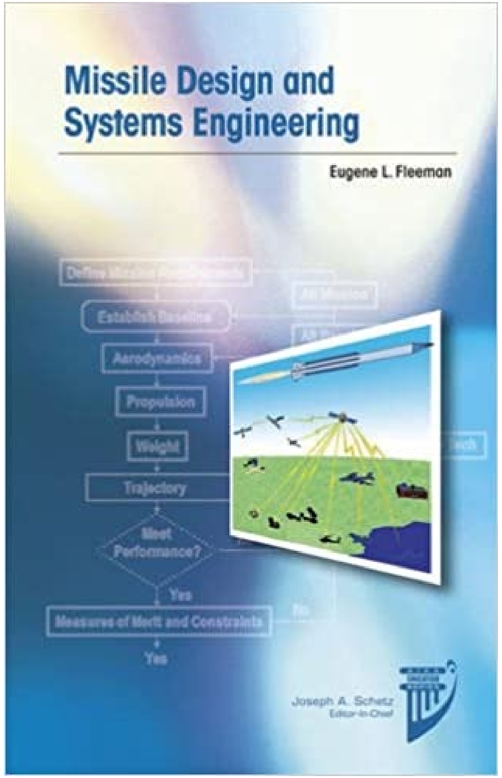
Agenda

- Background
- Conceptual Design → MBSE
- Model implementation
- Propulsion applications
- Threat Assessment
- Conclusion

Background

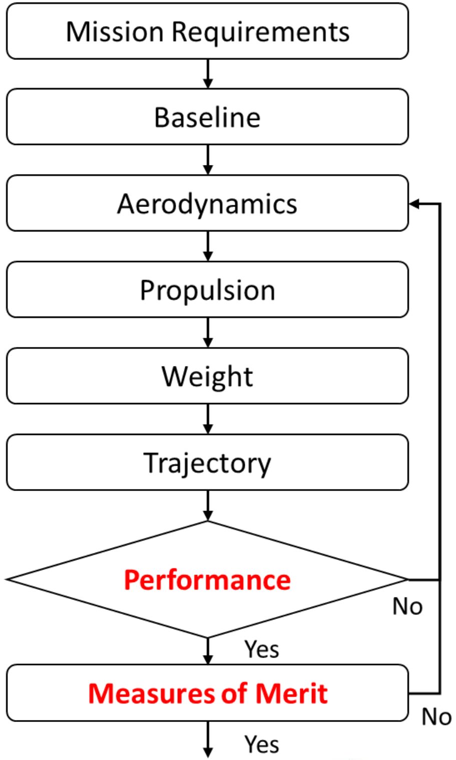
- Fleeman spreadsheet project*
  - Turbojet missiles and guided bombs
- Separate model built for V&V
  - Optimization capability
- All subsystems modeled
  - Propulsion focus this year
  - Aerodynamics was presented in 2022



*Spears, S., Allen, R., and Fleeman, E., “First-Order Conceptual Design for Turbojet Missiles and Guided Bombs,” AIAA SciTech Conference, San Diego, CA, 3-7 January 2022, AIAA-2022-03

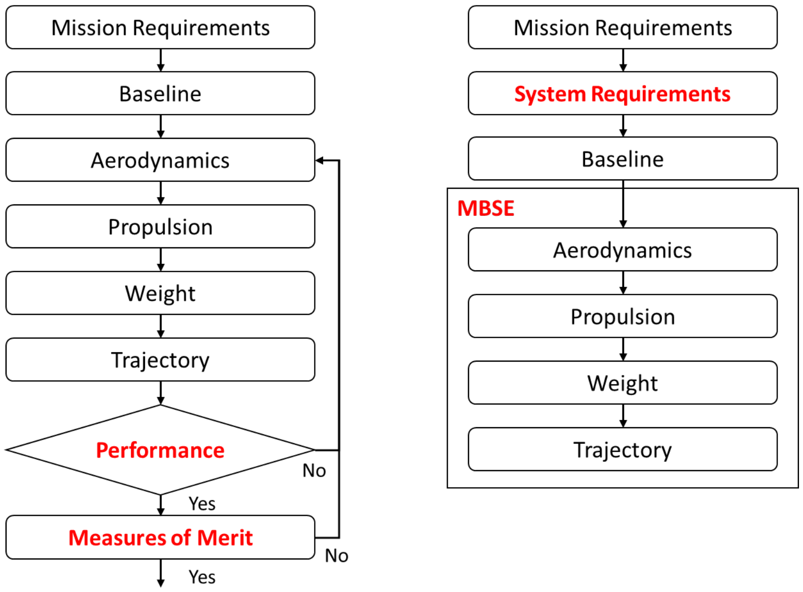
Conceptual Design and System Engineering

- In *Missile Design and System Engineering* (AIAA Education Series), Eugene Fleeman presents a comprehensive approach to first-order conceptual missile design and system engineering
- Two decisions (*Performance* and *Measures of Merit*) within an iterative process



Modified Conceptual Design for MBSE

- Instead of assessing *Performance* and *Measures of Merit* as decisions, move them to *System Requirements*, inserted after the mission requirements, before the baseline is established
- The modified process facilitates Model-Based Systems Engineering (MBSE)



Model Implementation

- Digital twin generic missile model architected from the physics-based engineering equations found in Fleeman's text
- Implemented in *Evolved AI*™ capable of machine learning and optimization
- Model organized by subsystem, i.e., aerodynamics, propulsion, mass properties, flight performance, and other measures of merit
- Equations verified and validated by checking each formula with sample calculations associated with the examples found in Fleeman's text, his course notes, or direct correspondence
- *Evolved AI* and its capability of stochastic optimization allows the System Engineer to optimize the flight performance criteria or measures of merit and observe the influencing parameters

Propulsion Application

- Rocket Motor
  - Given the geometry of the rocket motor, propellant properties, and thrust requirements, determine the required chamber pressure and resulting specific impulse, propellant flow rate, propellant burn area, and propellant burn time
  - Assess range performance assuming optimized aerodynamic properties, which was demonstrated at last year’s Defense Forum
- Ramjet (combustor design considerations)
  - Given a worst-case scenario (high combustion temperature) and working backward from thermal choking conditions, determine the subsonic combustor entrance Mach number and subsequently, the inlet throat area.
  - Assess range performance assuming optimized aerodynamic properties, which was demonstrated at last year’s Defense Forum

Rocket Motor

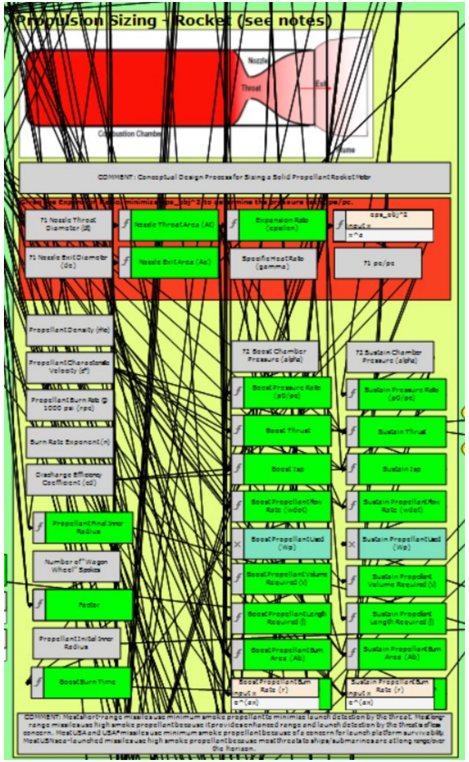
- Geometric properties:
  - Exit diameter (area), e.g., de = 3.78 in
  - Throat diameter (area), e.g., d* = 1.52 in
  - Find the exit-to-chamber pressure ratio (pe/pc) using interpolation

$$\frac{pe}{pc} = 0.02491$$

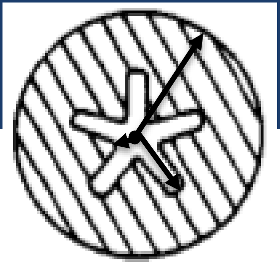
- Propellant properties:
  - Density, e.g., 0.065 lbm/in^3
  - Characteristic velocity, e.g., 5200 ft/sec
  - Burn rate @ 1000 psi, e.g., 0.5 in/sec
  - Burn rate exponent, e.g., 0.3
  - Discharge efficiency, e.g., 96%
  - Given desired thrust, compute the chamber pressure (pc)

$$\text{Desired boost thrust} = 10X\ \text{weight}$$
$$\text{Desired sustain thrust} = \text{drag (at flight conditions)}$$

	Thrust (lb)	*pc (psi)*	*Isp (sec)*	*$\dot{w}$ (lbm/sec)*	*Ab (in2)*
Boost	*5096*	*1764*	*257*	*19.8*	*513*
Sustain	*801*	*301*	*239*	*3.4*	*149*



Rocket Motor



- Finally, determine the burn time
  - Goal: inner and outer radii of propellant to burn to the casing at the same time
  - $r_{CASE} = r_{inner}$ + f * (burn time) * (burn rate), where f depends on the geometry of the propellant, rearranging: **burn time = (r_{CASE} - r_{inner})/(f * burn rate)**
  - Assume a wagon wheel star pattern burning with a factor = 2*sin(2π/N), where N is the number of spokes in the wagon wheel

*For 5 spokes, 2*sin(2π/5)=1.9*

  - Burn rate depends on chamber pressure, given by **burn rate = (burn rate @1000 psi)*(pc/1000)^n**, where pc is the chamber pressure (psi) and n is the burn rate exponent
  - Once burn time is determined, the outer radius of the propellant is computed by subtracting the product of the burn time and burn rate from the casing radius, **r_{outer} = r_{CASE} - (burn time * burn rate)**

Boost chamber pressure = 1764 psi
Burn rate (@1000psi) = 0.5 in/sec
Burn rate exponent = 0.3
Burn rate = 0.5(1764/1000)$^{0.3}$ = 0.59 in/sec*

Assuming a case radius of 4 in and an initial inner radius of 0.5 in
*Burn time = (4-0.5)/(1.9*0.59) = 3.1 sec*
*Outer radius = 4-(3.1*0.5) = 2.4 in*

Rocket Motor Performance

- Boost/Sustain/Coast/Glide @ γ=45°
  - Boost (3.7 sec): Run with Mach=0 at 0 ft
    - End-of-boost velocity (1408 ft/sec)
    - End-of-boost altitude (1836 ft)
    - End-of-boost range (2598 ft)
  - Sustain (10.9 sec): Re-run Mach=1.3 at 1836 ft
    - End-of-sustain velocity (1586 ft/sec)
    - End-of-sustain incremental altitude (12863 ft)
    - End-of-sustain incremental range (18190 ft)
  - Coast (9.9 sec): Re-run Mach=1.5 at 14699 ft
    - End-of-coast velocity (1102 ft/sec)
    - End-of-coast incremental altitude (8680 ft)
    - End-of-coast incremental range (12275 ft)
  - Glide: $(L/D)_{max}$=5 at **23379** ft = 116895 ft

γ=45°	*Incremental Altitude (ft)*	*Incremental Range (ft)*
Boost	*1836*	*1836*
Sustain	*12863*	*12863*
Coast	*8680*	*8680*
Total Altitude	***23379***	
Glide		*116895*
Total Range		*140274*
Total Range		*23(nmi)*

Ramjet

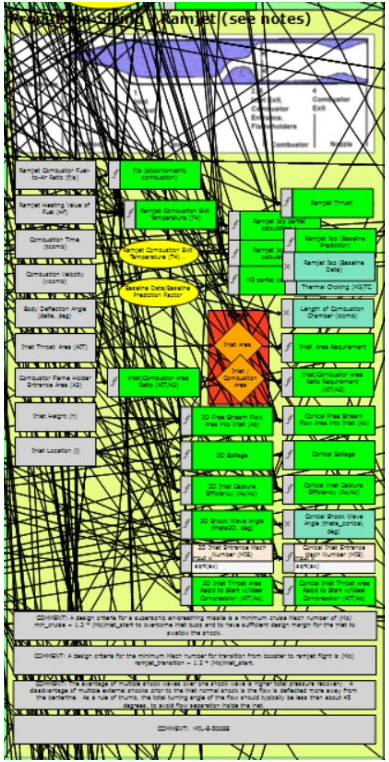
- Max thrust occurs at high combustion temp and high fuel/air
- But insulation technology limits combustor temp and fuel/air
- Assume high combustor temp $T_4 = 4000\ R$
  - From this thermally choked $M_4 = 1$ condition, determine the subsonic combustor entrance Mach number and the inlet throat area

Combustor Entrance	$M_3=0.193$
Inlet Throat Area	$93.7\ in^2$
Pressure Loss	$p_4/p_3=0.79$

  - ← Upper limit on the throat area
  - Reducing the combustor entrance Mach number by 10%...

Combustor Entrance	$M_3=0.174$
Inlet Throat Area	$84.8\ in^2$
Pressure Loss	$p_4/p_3=0.79$

  - ← No additional pressure loss
  - …reduces the inlet throat area to provide good specific impulse while satisfying the thrust requirement.

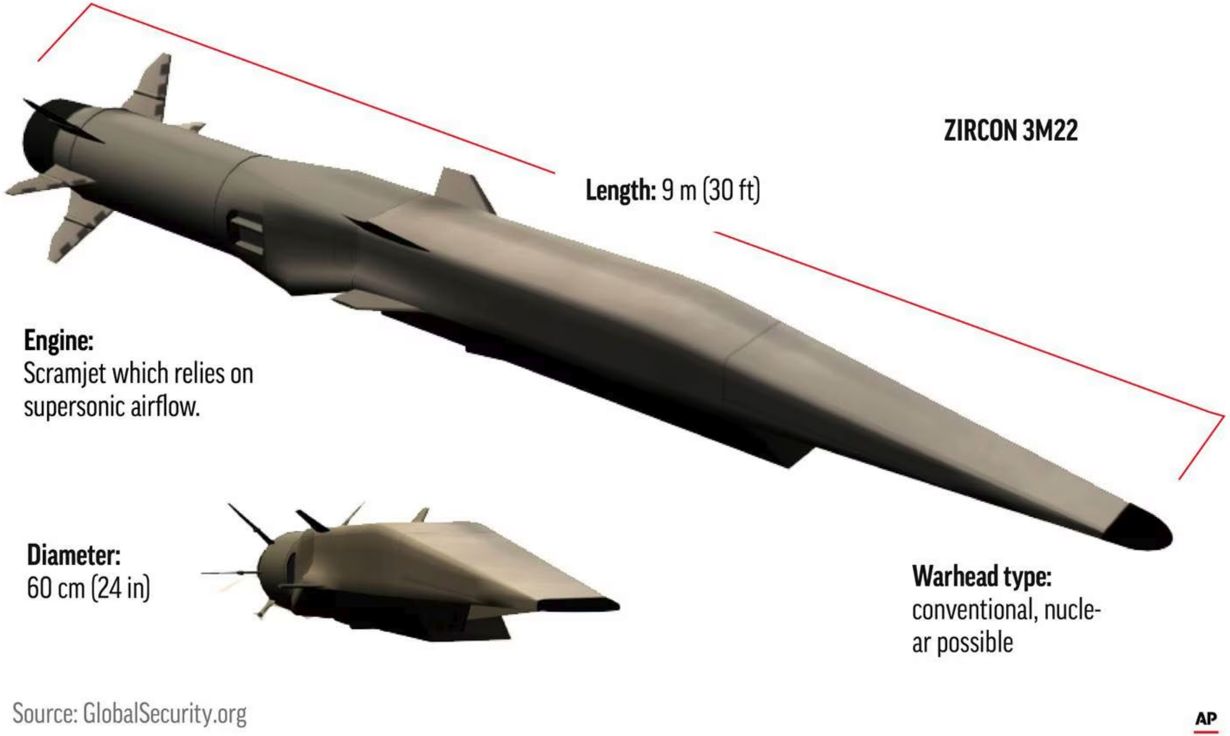


Ramjet Performance

- Cruise – Breguet Range Equation
  - Given flight conditions: M=3 at 60 kft
    - $(L/D)_{MAX}$ = 2.05 at 15.2° angle-of-attack
  - Maximum range is **568 nmi** (3.5E+06 ft)
  - Note: $W_{propellant}/W_{BeforeCruise}$ = 476 lb / 1739 lb = 0.27

Threat Assessment – 3M22 Zircon

- https://www.defensenews.com/naval/2023/01/05/russias-hypersonic-missile-armed-ship-to-patrol-global-seas/



Threat Assessment – 3M22 Zircon

- https://en.wikipedia.org/wiki/3M22_Zircon
- Working from a ramjet (ASALM) baseline (flight conditions imply supersonic combustion, scramjet)
- Geometry
  - Length = 360 in (given), Diameter = 24 in (given), Nose length = 108 in (estimate)
- Surface sizing estimates
  - Canards: none
  - Wings: 2 sets, chord = 9 in, span = 9 in, sweep = 45 deg, planform area = 121.5 in^2, mac at 234 in from nose
  - Tails: 3 sets, chord = 4.5 in, span = 9 in, sweep = 10 deg, planform area = 243 in^2, mac at 333 in from nose
- Flight Conditions
  - Altitude = 92 kft (given), Mach number = 9 (given)
- TruSolve™ determines $(L/D)_{MAX}$ occurs at 13° angle-of-attack
- Maximum range = **500 nmi** = 926 km

Threat Assessment – 3M22 Zircon

- https://en.wikipedia.org/wiki/3M22_Zircon (**540 nmi**)
- The article mentions development of new fuels
- The article mentions Some internet sources even claim the range of missile can reach 1,000 - 2,000 km

Fuel	h_{PR} (BTU/lbm)	Range (nmi)	Range (km)
RJ-5	17,900	500	926
Hydrocarbon	19,000	528	978
	25,000	681	1261
	30,000	804	1489
	35,000	925	1713
	40,000	1041	1928
	45,000	1155	2139
Hydrogen	51,600	1301	2409

Conclusions / Future Work

- With two modifications to the conceptual design and system engineering process, the System Engineer can apply MBSE to optimize the design of a missile based on mission and system requirements
- While the examples focused on aerodynamics (last year) and rocket motor / ramjet propulsion (this year), the digital twin has multidisciplinary design optimization (MDO) capability
- By adding a scramjet model (Heiser & Pratt), FOCDO may be used for hypersonics
- Furthermore, since hypersonics involve uncertainties and nonlinearities, Evolved AI's stochastic optimization is fully equipped for MDO

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TruSolve™ Stochastic Optimization

- COCO – COmparing Continuous Optimizers
  - Benchmarking of global optimizers
  - Black-Box Optimization Benchmarking (BBOB) at the Genetic/Evolutionary Computation Conference (GECCO)
  - Compiled C code with wrappers in other languages (Python, Java, MATLAB, Rust)
  - Provides test function suite, logging, and postprocessing for graphing and tables
  - Data archive for large (>200) collection of optimization algorithms

	TruSolve	Particle Swarm	Genetic Algorithm	Simulated Annealing	Gradient Descent
Nonconvex Accuracy (complex, high dimensionality…)	✓	–	✓	×	×
Noncontinuous Accuracy (Boolean, PSK,…)	✓	✓	✓	–	×
Stochastic Accuracy (noisy functions)	–	✓	✓	×	×
Stochastic Compatibility (supports non-discrete processes)	✓	✓	✓	✓	×
Practical Scaling above 1,000 variables (or neurons), 100-time intervals	–	–	×	–	×
Processing Load (speed, memory…)	✓	✓	×	×	✓