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INDUSTRIAL APPLICATION

A multi-objective, multidisciplinary design optimization methodology for the conceptual design of a spacecraft bi-propellant propulsion system

H. R. Fazeley[1] · H. Taei[2] · H. Naseh[3] · M. Mirshams[4]

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Abstract Space propulsion systems play an increasingly important role in planning of space missions. The traditional method for design of space propulsion systems includes numerous design loops, which does not guarantee to reach the best optimal solution. Multidisciplinary Design Optimization (MDO) is an approach for the design of complex systems that considers a design environment with multiple disciplines. The aims of this study are to implement and compare Multidisciplinary Feasible and Collaborative Optimization architectures for the multi-Objective optimization of a bi-propellant space propulsion system design. Several disciplines such as thrust chamber, cooling, and structure were exploited in a proper combination. The main optimization objectives in the MDO frameworks were to minimize the total wet mass and maximize the total impulse by considering several constraints. Furthermore, Genetic Algorithm and Sequential Quadratic Programming are employed as the system-level and local-level optimizers. The presented design methodology provides an interesting decision making approach to select the best system parameters of space propulsion systems under conflicting goals.

Keywords Space propulsion systems · Multidisciplinary Design Optimization (MDO) · Multi-objective optimization · Multidisciplinary feasible · Collaborative optimization

✉ H. R. Fazeley
h.fazeley@gmail.com

H. Taei
taei@mut.ac.ir

H. Naseh
hnaseh@ari.ac.ir

M. Mirshams
mirshams@kntu.ac.ir

1 Space Research Lab, K.N. Toosi University of Technology, P.O. Box 16765-3381, Tehran, Islamic Republic of Iran

2 Faculty of Aerospace Engineering, Malek Ashtar University of Technology, P.O. Box 15875-1774, Tehran, Islamic Republic of Iran

3 Ministry of Science, Research and Technology, Aerospace Research Institute, P.O. Box 14665-834, Tehran, Islamic Republic of Iran

4 Faculty of Aerospace Engineering, K.N. Toosi University of Technology, P.O. Box 16765-3381, Tehran, Islamic Republic of Iran

1 Introduction

Space propulsion systems play an important role in the planning of space missions and have received great attention from scientists in many countries. The ever growing weight and complexity of space systems have changed optimum space propulsion cost and performance parameters (Johnson et al. 2013). As a result, bipropellant propulsion systems are more competitive than current other systems. Bi-propellant thrusters, due to their relatively simple design, low development costs and high reliability, have been widely used in the field of space propulsion.

The conceptual design phase of a bi-propellant thruster contains interaction between specialized disciplines such as thrust chamber, propellant tanks, structure analysis and pressurization system, to mention a few, often with conflicting objectives and constraints. A considerable amount of literature has been published on design and development of bi-propellant thrusters. For example, (Hearn 1988) reviewed thruster requirements and concerns for bipropellant blow-down systems in the design process. More information in this field can be found in previous studies, which are major sources in space propulsion systems design. (Humble et al. 1995; Huzel et al. 1992; Sutton and Biblarz 2010; Turner 2009)

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In general, design of a bi-propellant thruster is a complex and multidisciplinary process. For example, the designer of these systems is faced with many design issues including how oxidizer and fuel are synthesized, how the thrust chamber should be cooled, how many injectors should be used, and how to evaluate different thruster configurations. Recently, emphasis has been on the advances that can be achieved with the interaction between two or more disciplines. It is fundamentally a multidisciplinary and multi-objective process. The traditional method for design of space propulsion systems includes numerous design loops, which do not guarantee to reach the best optimal solution. The principled application of formal optimization techniques to complex system design has led to the rapid development of an optimization field named Multidisciplinary Design Optimization (MDO).

MDO is an approach for the design of coupled engineering systems that coherently exploits the synergism of mutually interacting phenomena (Alexandrov and Hussaini 1997). In recent years, there has been an increasing amount of literature on the MDO domain, centered at the beginning on aerospace industries. However, nowadays they are used in various kinds of enterprises (e.g., automotive industries and marine industries) to improve the quality of products (Grujicic et al. 2009; Hart and Vlahopoulos 2010; Mirshams et al. 2014). As a rule, MDO techniques bridge the gap between subsystem analysis and optimal system design by providing a different optimization framework for design groups. The framework supports design improvement by methodically considering the system level penalties of various disciplinary components and configuration options.

Most studies in the field of bipropellant space propulsion systems have only focused on conventional design methods (Humble et al. 1995; Huzel et al. 1992; Sutton and Biblarz 2010). In addition, very few studies have been found that applied advanced conceptual design techniques in the field of spacecraft propulsion systems. Based on the aforementioned notes, the major objective of this study is to present new application of MDO to the design of bi-propellant low thrust space propulsion systems. Furthermore, investigating the results of applying Multidisciplinary Feasible and Collaborative Optimization frameworks for the multi-objective optimization of the mentioned system is another objective of this work. In the field of bi-propellant space propulsion systems, the coupling of objective functions due to the design variables (such as chamber pressure, oxidizer to fuel ratio, propellant type, etc.) in an engineering design process will results in difficulties for evaluating and comparing various thruster options. In the present paper, by solving the design problem in the MDO frameworks, a set of Pareto solutions is obtained. This methodology can help the designers to easily evaluate and compare various thruster system designs. The outline of this paper is organized as follows. Section 2 introduces the design problem and in Section 3 the design problem is implemented in the MDO architectures. Section 4 presents the optimization results and finally, the conclusions are drawn.

2 Problem definition

Improved spacecraft mass and extended on-orbit life requirements need increased performance using integrated space propulsion systems. In recent years, there has been an increasing demand for higher performance bi-propellant propulsion systems for future space missions. In order to improve the performance of these type of systems, a design process integration approach is presented to optimize the design process. A bi-propellant thruster with thrust range of 3000 to 5000 N was selected as the test problem for this study. This propulsion system could be integrated into satellites for orbital transfer maneuvers. The schematic of the integrated bi-propellant propulsion subsystem is shown in Fig. 1. This propulsion system provides both the apogee and on-orbit maneuvers. Because of weight and complexity considerations, a gas pressurized feed system is selected for the engine cycle.

Generally, the selection of number of propellant and pressurant tanks depends on many factors (e.g., propellant type, mission requirements, vehicle configuration, reliability, modularity, etc.). In many cases, vehicle configuration dictates this decision. Because of geometrical constraints, three

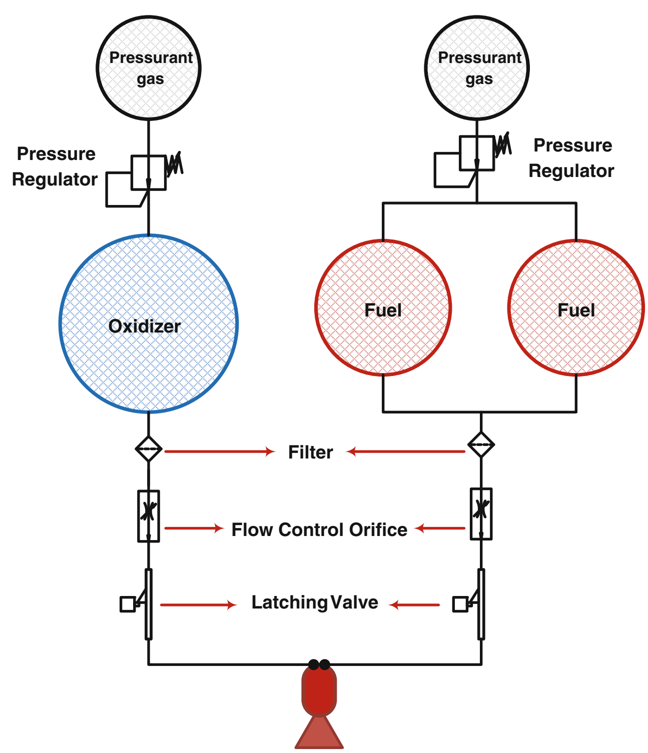


Fig. 1 Schematic of the bi-propellant propulsion system components

spherical propellant tanks and two pressurant tanks were considered in the test problem. In the design procedure, based on the customer and mission requirements, many objectives (e.g., maintenance, reliability, performance, life cycle cost, and mass) may be considered. The basic overall goals will normally be maximizing performance as well as minimizing weight within reasonable technology limits. Total impulse is an important figure of merit of a vehicle, and is derived from integrating the thrust produced over the operating time. The total impulse is proportional to Specific Impulse (Isp) and also total energy released by the propellant of a system. The higher the Isp, the less propellant is needed to produce a given thrust during a given time. Therefore, by increasing the Isp, propellant mass and thus propellant tanks volume can be reduced, while the thrust-chamber mass and dimension may be increased. Based on the aforementioned notes, in the present paper, maximizing the total impulse and minimizing the total wet mass were considered as design goals in the multi-objective optimization approach, and were implemented in the MDO frameworks. It should be noted that the design process is limited to technological and geometrical constraints. For example, available structural technology does not allow the wall temperature to be more than 2000 K. Length and diameter of the thrust-chamber and fuel and oxidizer tanks radius (allowed by the installation in the upper stage) are geometrical constraints, which are considered based on overall upper stage specifications. Modeling of the bi-propellant propulsion system consists of employing a suit of analysis modules based on a combination of physical and empirical models. In the present paper, all critical system performance characteristics are computed using combustion, nozzle geometry, thrust chamber, cooling, pipelines, propellant tanks, pressurizing system, structure, and mass disciplines. Furthermore, over one hundred of variables (including design variables, coupling variables, state variables) and parameters were used in modeling the conceptual design of presented system, and the most important ones are briefly described as follows.

2.1 Analysis modules

2.1.1 Combustion

Modeling and simulation of the combustion process, is one of the most important requirements in determining the performance of a bi-propellant propulsion system. For this purpose, many computational tools (such as CEA, CEC, GASEQ, and CANTERA) have been developed in different industries. In this research, NASA Glenn's computer code CEA (Chemical Equilibrium with Applications) is applied to determine the properties of the combustion products. CEA is a fast and accurate combustion code, which is usually used to analyze and validate combustion processes (Chen et al. 2012). Minimization of free energy approach to chemical equilibrium calculations has been used in all versions of this program (McBride and Gordon 1996). In this analysis module chemical equilibrium analysis was performed for modeling adiabatic combustion. Three main inputs of the combustion discipline are oxidizer and fuel type, combustion chamber pressure, and oxidizer to fuel ratio. These design variables are the key aspects in design of bi-propellant systems. Combustion chamber pressure affects the size and Isp of the thruster. Generally, propellant type and optimum oxidizer to fuel ratio are determined based on many major factors (i.e., propellants density, cooling considerations, and start capability). It is obvious that deviations from these values will penalize vehicle performance. In the case of low thrust bi-propellant thrusters, hypergolic propellants are usually used, because of weight consideration, simplicity, and reliability issues. In this study, N_2O_4/UDMH and N_2O_4/MMH combinations were considered in modeling the design problem. The variation of combustion temperature, specific heat ratio, and molecular mass, computed in the combustion discipline with respect to oxidizer to fuel ratio and combustion chamber pressure for N_2O_4/UDMH are shown in Figs. 2, 3 and 4. As shown in these figures, oxidizer to fuel ratio has a big impact on the properties of the combustion products.

2.1.2 Thrust chamber

The thrust chamber provides an appropriate space for mixing of oxidizer and fuel, and complete chemical combustion. The thrust chamber analysis module uses gas dynamics equations to calculate overall system characteristics. During the preliminary design, allocations and assumptions were considered to

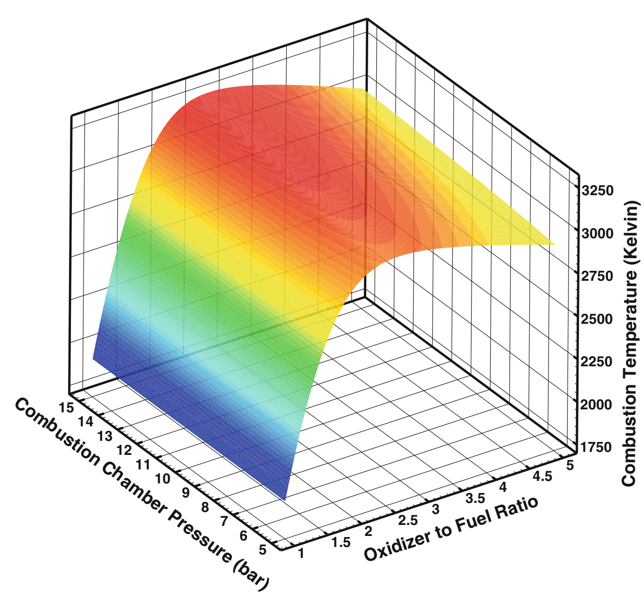


Fig. 2 Combustion temperature versus oxidizer to fuel ratio and combustion chamber pressure for N2O4/UDMH

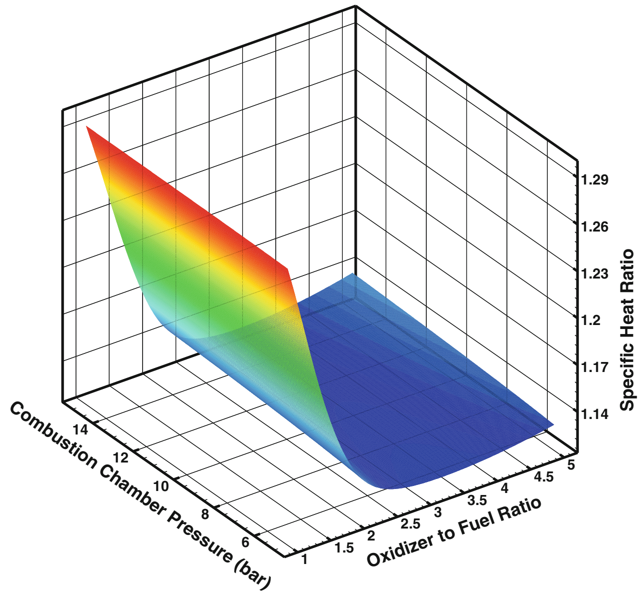


Fig. 3 Specific heat ratio versus oxidizer to fuel ratio and combustion chamber pressure for N2O4/UDMH

simplify the design process. The most important assumptions are adiabatic combustion; one dimensional isentropic fluid flow; complete combustion; and homogeneous mixing during combustion. Inputs of this analysis module consist of a set of design variables, design parameters, and coupling variables. The main input variables of this discipline are thrust, burn time, chamber pressure, exit pressure, combustion products properties, propellants type, and characteristic length. So far, various combustion chamber shapes such as spherical, cylindrical, and conical have been used in design of bi-propellant systems. Each of these shapes has many advantages and disadvantages which can affect other design disciplines. Historically, in bi-propellant thrusters, cylindrical chambers are used. (Sutton 2006) Therefore, in the present paper, a cylindrical shape was selected for design of the combustion chamber. Moreover, characteristic length method was used to estimate chamber geometry. (Huzel et al. 1992) This method is based on engine test data and gas-dynamic considerations. The thrust chamber analysis module computes specific impulse, chamber geometry, and other performance specifications. In the current study, for calculating real Isp, several Isp losses (e.g., combustion and nozzle) based on empirical and statistical data have been considered. (Coats 2004) Variation of specific impulse with respect to the oxidizer to fuel ratio, combustion chamber pressure, and exit pressure of a bi-propellant N2O4/UDMH thruster is shown in Fig. 5.

2.1.3 Nozzle geometry

The nozzle plays an important role in producing propulsion system thrust. Generally, this component consists of diverging and converging parts. The design of the nozzle is influenced by many design considerations such as weight, performance, manufacturing, and geometry constraints etc. The final design would be based on a tradeoff among the benefits of improved performance and penalties of increased weight and greater complexity. Whereas the bell-shaped nozzle with its proper features (i.e., good performance, light weight, and small

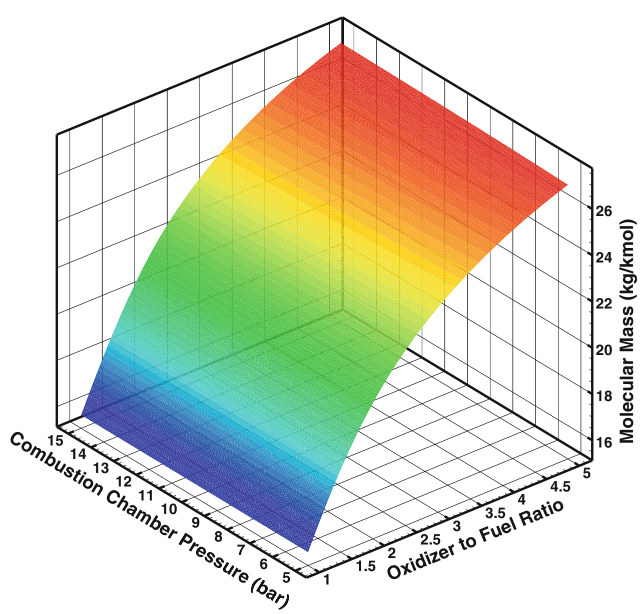


Fig. 4 Molecular mass versus oxidizer to fuel ratio and combustion chamber pressure for N_2O_4/UDMH

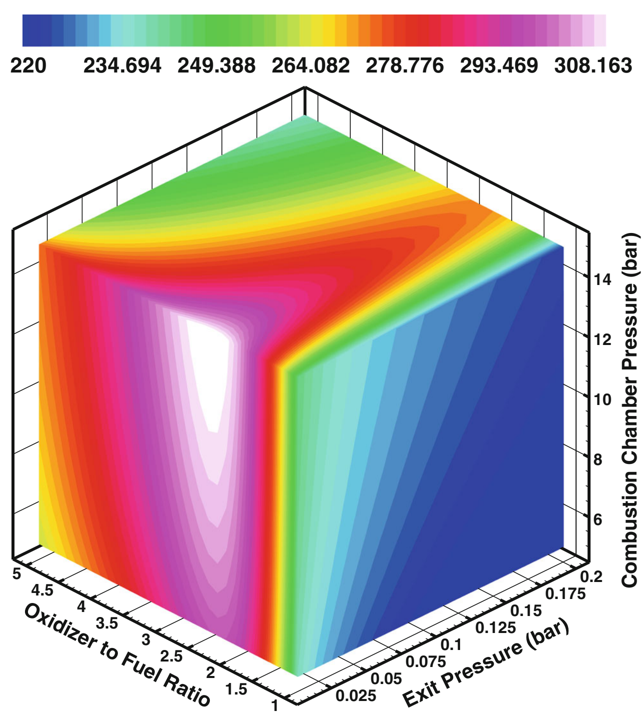


Fig. 5 Specific impulse performance of a bi-propellant N_2O_4/UDMH thruster versus oxidizer to fuel ratio, combustion chamber pressure, and exit pressure

losses), is popular for bi-propellant thrusters, the exact analysis of bell nozzles requires complex computation using CFD-based methods, which increase computation time and cost. Hence, these methods are seldom used for preliminary design. In this paper, parabolic geometry approximations (Humble et al. 1995) were used to estimate the bell-nozzle dimensions. This technique works well enough and its accuracy is acceptable in the conceptual design phase. Geometry constraints are the most important outputs from this discipline.

2.1.4 Cooling

Technically, thrust-chamber cooling is a major design issue, because of high combustion temperatures and high heat-transfer rates from hot gases to the chamber wall. So far, various thrust-chamber cooling techniques (e.g., Radiation, Regenerative, Ablative and Film cooling) have been developed and tested by researchers. (Huzel et al. 1992) There are certain drawbacks such as manufacturing complexity and weight issues associated with the use of Regenerative cooling technique in low thrust bi-propellant thrusters. Ablative cooling is used especially for short duration systems like booster engines with a limited operation time. Radiative cooling is a simple and efficient technique, but is commonly used when thermal stresses are low, such as nozzle extensions. Film cooling has been used, particularly for high heat fluxes, either alone or in combination with other cooling techniques. In this research, a combination of film cooling and radiation cooling was used to control the wall temperature of the thrust-chamber. In the film cooling technique, chamber wall surfaces are protected from immoderate heat by a thin layer of cooling fluid produced through orifices around the injector plate, while in the radiation cooling technique, the heat from the combustion gases is radiated away from the surface of the outer thrust-chamber wall. Several approaches were explored for modeling heat transfer in the thrust-chambers. In the present paper, a set of analytical and empirical relationships were used to calculate heat transfer rates (Howell et al. 1969; Huzel et al. 1992; Shine et al. 2012). The hot gas heat transfer coefficient was computed using Bartz correlation (Macdonald and Badescu 2014). The main inputs of the cooling discipline consist of the composition of fuel/oxidizer mixtures and flow rates, combustion gas properties, thrust-chamber geometry, chamber pressure, and film-coolant mass flow rate ratio, which were obtained from other integrated disciplines.

2.1.5 Pipelines

Most of the propulsion system components are connected to each other through pipelines. Technically, the objective of pipeline design is to make the pressure drops as low as possible. For early decisions, as a good approximation, incompressible flow relationships were used to estimate pressure drops. Pressure drops and pipeline dimensions are outputs from this discipline, which are integrated with the structure and tanks disciplines.

2.1.6 Propellant tanks

In design of bi-propellant systems, configuration of propellant tanks depends largely on many design factors (such as fuel and oxidizer type, oxidizer to fuel ratio, chamber pressure, location and layout of space propulsion montage, and volume of tanks). Spherical and cylindrical are the most common tank shapes. In most cases, vehicles of relatively low thrust range, high tank pressures and less stringent space conditions will use spherically-shaped tanks. These tanks have the most volume for a given surface area and tend to be lightest. Based on the aforementioned notes, spherical shape was considered in modeling the propellant tanks subsystem (Wagner and United States. National Aeronautics and Space Administration. 1974). The volume of the fuel and oxidizer tanks is estimated according to the operational propellant volume, trapped volume, reserve and unusable propellant volume, and ullage volume. Propellant tanks dimensions are selected as one of the design constraints which are ensured by the system-level optimizer.

2.1.7 Pressurizing system

Typically, pressurizing systems are used to keep the propellant tanks at the desired operating pressure. Chemical reaction, evaporation, and external stored gas are three main approaches in design of these systems. Stored gas systems are widely used in low thrust bi-propellant systems. In this approach, the gas is typically kept in a tank at an initial pressure ranging up to 60 Mpa, and delivered to the propellant tanks at a specified pressure regulated by a regulator. In this article, the stored gas system was used to pressurize the propellant tanks. In addition, helium and nitrogen were selected as pressurant gas in modeling of the pressurizing system. In this discipline, the design methodology for the pressurizing system was adopted from (Humble et al. 1995; Huzel et al. 1992; United States. National Aeronautics and Space Administration. 1976). In design of pressurizing system, the volume and mass of the stored gas are determined in an iterative process. In this approach, based on initial guess of the pressurant tank volume, pressurizing system specifications such as required mass of pressurant are computed. Then, with respect to this specifications, the required pressurant tank volume is evaluated. This process continuous until this algorithm has converged.

2.1.8 Structure

In this discipline, structural analysis relationships described in (Hart 1959; Huzel et al. 1992; Orlando et al. 1967) were applied to calculate components thicknesses. The criteria in analysis of the components' structure are no yield at the applied load and no failure at the ultimate load. As a rule, the wall thickness of bi-propellant system components is first approximated from stresses caused by internal pressure loads and discontinuities. Then, the thickness of the cylindrical or spherical tanks is determined using the hoop-membrane formula. Moreover, head (or end) thicknesses of the tanks are calculated by an equivalent thickness, which is an average value of the knuckle-crown thickness. It should be noted that safety factors for components' structural design were obtained from (Hart 1959). The results of this approach are used in the mass analysis module to improve the accuracy of prediction of components masses (see section 2.1.10). The main inputs of the structure discipline are components material, components dimensions, maximum acceleration, operational loads, and safety factors.

2.1.9 Mass

One of the most difficult steps in the conceptual design phase is to estimate the system mass. There are two main approaches to estimate the mass of the propulsion system: analytical and statistical (or empirical) approaches. Analytical models are not easy to develop for estimating the mass of system due to their complexity and lack of sufficient data in the conceptual design phase. In addition, statistical models need to establish a database for estimating the mass of system. In the present study, for elimination of weakness of these approaches, a combination of analytical and empirical models is employed in the mass module. The mass of the entire system is broken down into combustion chamber, nozzle, pipelines, injector system, propellant tanks, pressurizing system tanks, operational propellant, reserve propellant, unusable propellant, and stored gas parts. In the design process, the mass of the components is computed based on many parameters such as dimensions, components materials, wall thicknesses, and technology level factors.

2.1.10 Model validation

Model validation is an important step in the engineering design process. It is utilized to determine whether a system model is an accurate representation of a real system. In this step, the designers ensure that the model meets its proposed requirements in terms of the results obtained and the methods employed. In the present paper, the system model was created by connecting analysis modules (described in previous sections) to each other. The system model has been validated using propellant tanks data sheets and design data of the bi-propellant propulsion systems. Because of the importance of Vacuum Isp and dry mass in the system-level, they were selected for validation of the design model. Validation results for the vacuum Isp, thrust chamber mass and tanks mass are shown in Figs. 6, 7 and 8 and their average error is 1.52, 8.06 and 6.40 % respectively. These low average errors increase the confidence in the level of system model accuracy and are adequate to conceptual design phase.

3 Multidisciplinary Design Optimization (MDO)

3.1 Generic MDO formulation

Before implementing the design problem in the MDO framework, we should describe the fundamental notions of an MDO process:

Definition (Sobieszczanski-Sobieski and Haftka 1997) define MDO as a "methodology for design of systems in which strong interaction between disciplines motivates designers to simultaneously manipulate variables in several disciplines".

In general MDO problems, three main categories of variables are defined. Design variables are independent quantities that are controllable from the designer's point of view. Typically, design variables can be classified into continuous, discrete (including integer and categorical), and Boolean types. In the MDO frameworks, they are always under the explicit control of an optimizer. State variables represent analysis results of the disciplinary analysis, and depend on the design variables and state equations. In the MDO process, analysis modules are connected with each other by coupling

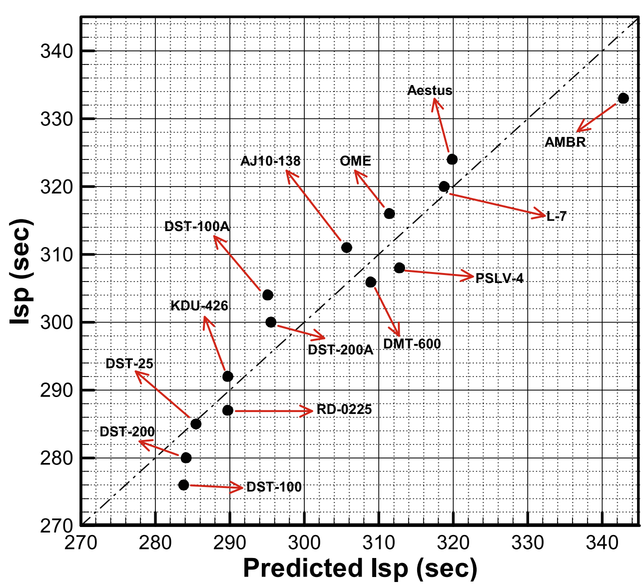


Fig. 6 Accuracy of the design model for predicting vacuum Isp

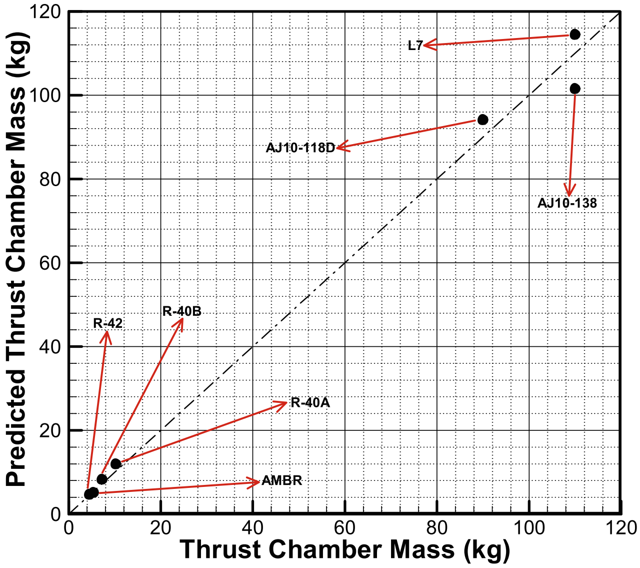


Fig. 7 Accuracy of the design model for predicting thrust chamber mass

variables. An MDO problem can be formulated in standard form as: (Balesdent et al. 2012)

$$
\begin{array}{ll}
\textit{Minimize} & f(x,y,z) \\
\textit{Subject to} & g(x,y,z)\leq 0 \\
 & h(x,y,z)=0 \\
 & \forall i, R_i(x_i,y_i,z_i)=0 \\
 & \forall i,\ \forall j\neq i,\ y_i=\left\{c_{ji}\left(x_j,\ y_j,\ z_j\right)\right\}_j \\
 & i=1,\ \ldots,\ n \\
\textit{With respect to} & x=\left\{x_{sh},\ \overline{x}_k\right\}
\end{array}
\quad (1)
$$

Where x is vector of design variables. x_{sh} symbolizes the variables which are shared between different subsystems (global variables) and $\overline{x}_k$ denotes the variables which are specific to one subsystem (local variables). z is the vector of the state variables; y is the vector of the coupling variables; Also, $f(.)$ is the objective function (i.e., cost function). The inequality constraints are described by $g(.)$ and $h(.)$ represents the equality constraints. $c_{ji}(.)$ symbolizes coupling functions which calculate the coupling variables from the subsystem i to the subsystem j. $R_i(.)$ characterizes the residual functions for the subsystem i, which quantify the satisfaction of the state equations, and n is number of the subsystems. $(.)_i$ represents functions or variables that apply to subsystem i. Generally, MDO architectures can be classified into two categories: monolithic formulations, and distributed formulations. Each of the MDO architectures has many advantages and drawbacks. Selection of proper MDO architecture depends on many factors such as the nature of design problem. In the presented design problem, because of strong coupling between disciplines, MDF and CO architectures were selected for solving the optimization problem. These frameworks are described as follows.

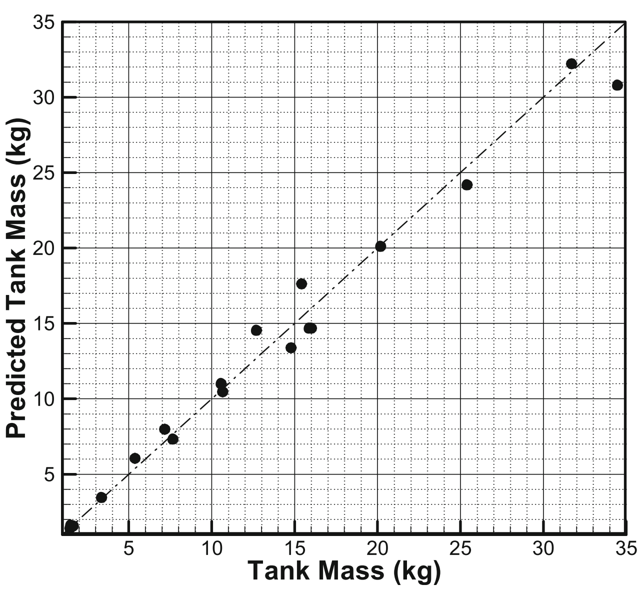


Fig. 8 Accuracy of the design model for predicting tanks mass

3.2 Multidisciplinary feasible (MDF) framework

In most cases of large-scale design problems, disciplinary analysis modules affect other subsystems and in fact, designers are faced with a tightly coupled system. Multidisciplinary Feasible (MDF) Framework is a single level architecture which handles the system analysis. MDF is the most general MDO formulation and has comprehensive industry acceptance, but is commonly restricted to small design space problems. The MDF, moreover known as fully integrated optimization and "all-in-one (AiO)" solves simultaneously the optimization problems with different subsystems. In this framework, a system analyzer coordinates all of the subspace analyzers and the system level optimizer controls the design process, ensuring that the global objective is achieved while the design constraints are satisfied.

MDF performs a complete system analysis at every optimization iteration. In the design problems that deal with coupled systems, some analysis methods (i.e., Fixed Point Iteration and Newton–Raphson) are regularly employed within an MDF approach. Compared with other monolithic formulations, the major benefit of MDF framework is that the dimension of optimization problem is as small as it can be for a monolithic formulation. Another advantage of this framework is that in each optimization iteration, MDF returns a solution that always satisfies the consistency constraints, but it may be time-consuming. An obvious disadvantage of the MDF is its limitation for parallelization the design process. This can increase the computational time and cost. The MDF architecture for the bi-propellant space thruster design problem (which was mentioned in Section 2) and coupling relationships of the disciplines are described with a Design Structure Matrix (DSM) shown in Fig. 9. As shown in this figure, a feedback

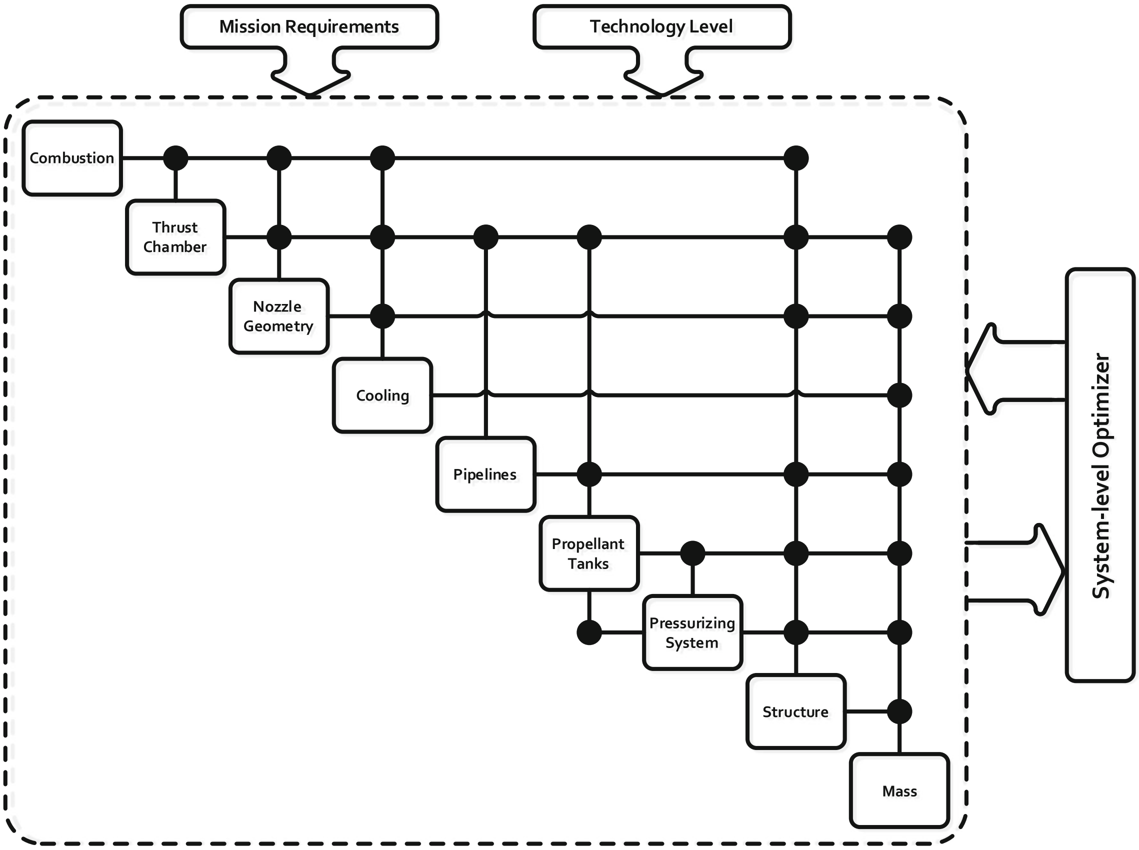


Fig. 9 MDF architecture for the bi-propellant space thruster design problem

coupling exists between pressurizing system and propellant tanks analysis modules. In order to solve this coupling, the Fixed Point Iteration was employed in Multidisciplinary Analysis (MDA) procedure.

Generally, in the design optimization process of bi-propellant thrusters, a large number of parameters can be considered as design variables. The design must be optimized to yield the design variables and design parameters for the disciplines. In order to decrease computation time, only most important parameters that have the greatest impact on design objectives and constraints (Huzel et al. 1992) were selected as the design variables. Therefore, thirteen design variables were used in the MDO setup. Table 1 shows the selected design variables. Discrete design variables are related to

Table 1 Design variables for the design problem in the MDF architecture

No.	Design variable	Type	Symbol	Unit	Range
1	Propellant type	Discrete	*prop*	–	N2O4/UDMH, N2O4/MMH
2	Pressurant gas	Discrete	*press*	–	Helium, Nitrogen
3	Tanks material	Discrete	*mat*	–	Aluminum 2219, Titanium Ti-6Al-4 V, Steel AISI 4130
4	Chamber pressure	Continuous	P_{cc}	*bar*	7–12
5	Oxidizer to fuel ratio	Continuous	O/F	–	1.2–3.5
6	Propellant mass	Continuous	M_P	*kg*	20–100
7	Burn time	Continuous	t_B	*sec*	30–120
8	Exit pressure	Continuous	P_e	*bar*	0.005–0.2
9	Characteristic length	Continuous	L^*	*m*	0.6–0.8
10	Convergent half angle	Continuous	θ_{conv}	*deg*	15–30
11	Initial gas pressure for pressurizing tank 1	Continuous	$P_{i\ Ox}$	*bar*	200–400
12	Initial gas pressure for pressurizing tank 2	Continuous	$P_{i\ Fu}$	*bar*	200–400
13	Film coolant mass flow rate ratio	Continuous	λ	–	0.03–0.15

fundamental decisions (e.g., propellant selection) in the design process. Continuous design variables determine system specifications, such as performance, mass, and configuration. For example, chamber and exhaust pressures determine the expansion ratio of the nozzle, and affect the Isp and the size of thruster. The important design parameters are listed in Table 2, and they are related to many factors such as spacecraft mission, structural analysis, and cooling. Design constraints for the design problem in the MDF architecture are shown in Table 3. Using the MDO architecture presented in Fig. 9, the optimization problem can be formulated as follows:

$$\begin{array}{ll} \textit{Minimize} & \textit{Total wet mass} \\ \textit{Maximize} & \textit{Total impulse} \\ \textit{With respect to} & \left\{prop,\ press, mat, P_{cc}, \frac{O}{F}, M_P, t_B, P_e, L^*, \theta_{conv}, P_{i\ Ox}, P_{i\ Fu}, \lambda\right\}^T \\ \textit{Subject to} & 3000 \leq Th \leq 5000 \\ & 285 \leq Isp \\ & T_{max} \leq 2000 \\ & L_T \leq 0.8 \\ & D_e \leq 0.4 \\ & r_{ox\ tank} \leq 0.3 \\ & r_{fu\ tank} \leq 0.25 \end{array} \quad (2)$$

NSGA-II is one of the successful non-decomposition multi-objective evolutionary algorithms, and in the present study, this approach is chosen to solve the Multi-objective Multidisciplinary Design Optimization (MMDO) problem, because of approximately simplicity, robustness, elitism, efficiency, and proper spread in its optimized solutions (Arias-Montano et al. 2012; Konak et al. 2006). NSGA-II is a computationally efficient algorithm accomplishing the concept of a selection method based on classes of dominance of all solutions. This paper does not intend to address the details of NSGA-II and only presents the results of applying it to solve the MMDO problem. The NSGA-II parameters for this study are reported in Table 4. After implementing the multi-objective optimization problem on the MDF framework, it was solved by the NSGA-II and the results are discussed in following section.

3.3 Collaborative Optimization (CO)

Collaborative Optimization (CO) is a bi-level optimization framework developed for the design of multidisciplinary and complex systems that was originally proposed in 1994 (Kroo et al. 1994). The key concept in the CO is the decomposition of the design problem into two levels, namely discipline level and system level optimization. The CO is designed in such a way that it supports disciplinary autonomy while maintaining interdisciplinary compatibility, thus providing added design flexibility. These features make CO well suited for use in a practical multidisciplinary design environment such as space propulsion systems. In this framework, the discipline level sub-problems are made independent of each other by using copies of the coupling and shared design variables. These copies are shared between disciplines during every

Table 2 Important design parameters for the bi-propellant propulsion system design problem

No.	Design parameter	Unit	Value
1	Spacecraft acceleration	m/sec^2	50
2	Feed lines losses	bar	0.4
3	Velocity of propellant flow through feed lines	m/sec	10
4	Number of fuel tanks	–	2
5	Number of pressurizing tanks	–	2
6	Film cooling efficiency	–	0.5
7	Stefan–Boltzmann constant	W/m^2K^4	5.67×10^{-8}
8	Total emissivity of outer wall surface	–	0.9
9	Factor of safety	–	1.5
11	Initial gas temperature	K	273.15

Table 3 Design constraints for the design problem in the MDF architecture

No.	Design constraint	Symbol	Unit	Lower bound	Upper bound
1	Specific Impulse	Isp	sec	285	–
2	Thrust	Th	N	3000	5000
3	Thrust-Chamber diameter	D_e	m	–	0.4
4	Thrust-Chamber length	L_T	m	–	0.8
5	Fuel tank radius	$r_{fu\ tank}$	m	–	0.25
6	Oxidizer tank radius	$r_{ox\ tank}$	m	–	0.3
7	Maximum wall temperature	T_{max}	K	–	2000

optimization iteration. Equality constraints are required to ensure that matching of discipline level and system level variables at each optimal design.

If the design problem has a great number of coupling variables, CO framework can be inefficient. The CO architecture for the proposed design problem is shown in Fig. 10. In the present investigation, in order to decrease the coupling variables of the proposed design problem in the CO framework, all analysis modules are grouped into two super-disciplines based on degree of coupling. Each of these super-disciplines uses the multidisciplinary analysis from the MDF framework with a modified objective function.

In the system level optimization problem, only vectors of coupling variables y^s and system design variables x^s could be changed. In addition, in the k-th discipline level optimization problem, only y_k and x_k could be changed. In this design procedure, when the optimization problem converges, the system level variables will get close to discipline level variables. The system level optimization of the presented design problem can be stated as below:

$$\begin{array}{ll} Minimize & Total\ wet\ mass \\ Maximize & Total\ \ impulse \\ With\ respect\ to & \{prop^s, press^s, mat^s, M_P{}^s,\ t_b{}^s,\ O/F^s,\ \lambda^s,\ P_{inj}{}^s,\ M_t{}^s\ \}^T \\ Subject\ \ to & J_{sup\ dis\ 1}{}^s \le 10^{-5} \\ & J_{sup\ dis\ 2}{}^s \le 10^{-5} \end{array} \tag{3}$$

Where $P_{inj}{}^s$ and $M_t{}^s$ are the injector pressure and thrust-chamber mass respectively; $J_{sup\ dis\ 1}{}^s$ and $J_{sup\ dis\ 2}{}^s$ are compatibility constraint functions in the system level optimization problem which are described as same as the functions $J_{sup\ dis\ 1}$ and $J_{sup\ dis\ 2}$ in discipline level optimization problems. Other system level variables are the same as MDF (see Table 1). Since the dimension of design space for discrete variables was small, these variables were solved at the system level. In the discipline level optimization problems, discrete variables were considered as system parameters. The discipline level optimization problems can be formulated as (4) and (5). For solving the problem more efficiently, a combination of global objective and compatibility term was considered as the subspace objective function.

$$\begin{array}{ll} Minimize & J_{sup\ dis\ 1} + \dfrac{W_1}{Total\ impulse} \\ & J_{sup\ dis\ 1} = \left(1-\dfrac{M_P}{M_P{}^s}\right)^2 + \left(1-\dfrac{t_b}{t_b{}^s}\right)^2 + \left(1-\dfrac{O/F}{O/F^s}\right)^2 \\ & + \left(1-\dfrac{\lambda}{\lambda^s}\right)^2 + \left(1-\dfrac{P_{inj}}{P_{inj}{}^s}\right)^2 + \left(1-\dfrac{M_t}{M_t{}^s}\right)^2 \\ With\ respect\ to & \{P_{cc}, P_e\ , M_P,\ t_b, O/F, \theta_{conv},\ \lambda,\ L^*\}^T \\ Subject\ \ to & 3000\ \le\ Th \le 5000 \\ & 285\ \ \le\ Isp \\ & T_{max} \le\ 2000 \\ & L_T \le\ \ 0.8 \\ & D_e \le\ \ 0.4 \end{array} \tag{4}$$

$$\begin{array}{ll} Minimize & J_{sup\ dis\ 2} + \ W_2.\ Total\ \ wet\ \ mass \\ & J_{sup\ dis\ 2} = \left(1-\dfrac{M_P}{M_P{}^s}\right)^2 + \left(1-\dfrac{t_b}{t_b{}^s}\right)^2 + \left(1-\dfrac{O/F}{O/F^s}\right)^2 \\ & + \left(1-\dfrac{\lambda}{\lambda^s}\right)^2 \\ With\ respect\ to & \{M_P,\ t_b, O/F, P_{i\ Ox}, P_{i\ Fu},\ \lambda\}^T \\ Subject\ \ t\ o & r_{ox\ tank} \le\ 0.3 \\ & r_{fu\ tank} \le 0.25 \end{array} \tag{5}$$

Where W_1 and W_2 are weighted coefficients and discipline level variables range are presented in Table 1. In the discipline level optimization problem, the variables without superscript “s” are changeable during the optimization process, while the system level variables keep constant. In the described CO problem, for achieving the Pareto-optimal solutions, the NSGA-II was applied in the system level optimization. This optimization technique is known to obtain good global

Table 4 Genetic Algorithm (GA) parameters for MDF and CO architectures

No.	Mode/Parameter	Value
1	Maximum Generations	1000
2	Population size	70
3	Crossover	0.8
4	Mutation	0.06
5	Maximum constraint violation	0.02
6	Percent penalty	0.25
7	Stall generation limit	50

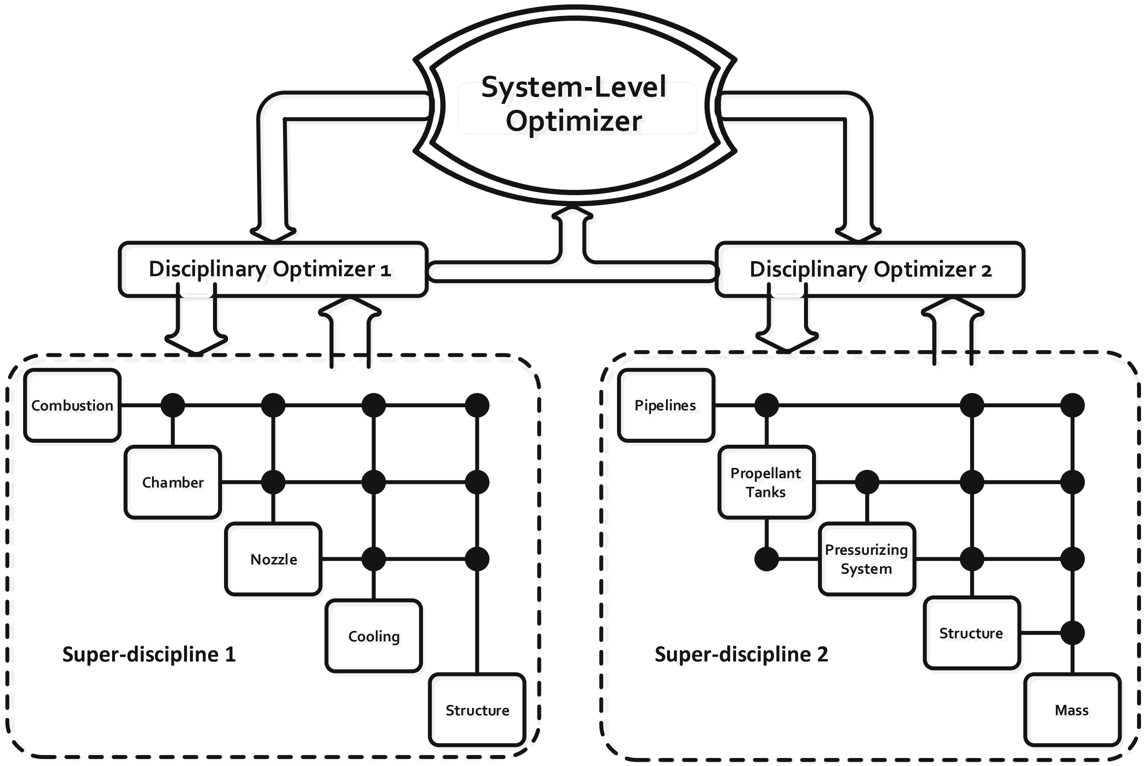
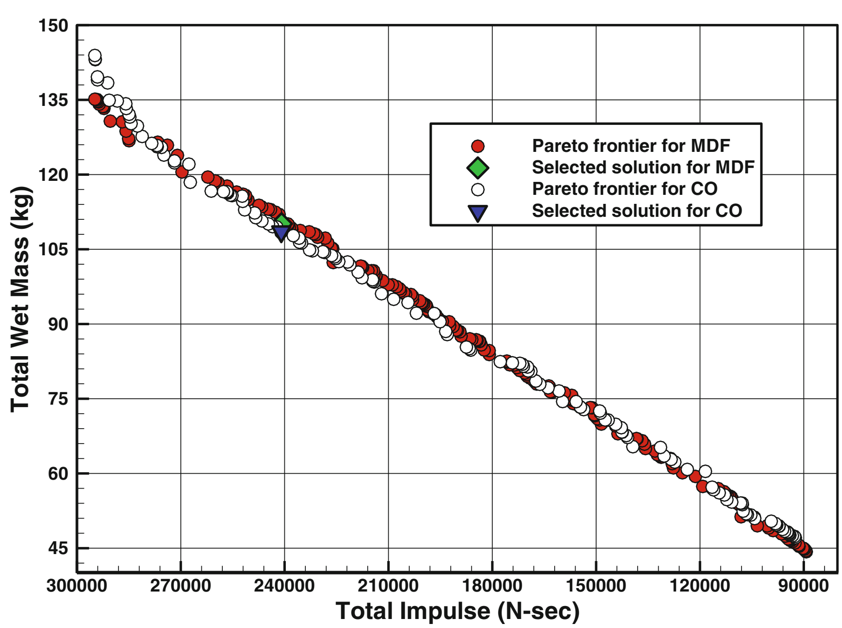


Fig. 10 CO architecture for the bi-propellant space thruster design problem

solutions and can allow us to consider discrete design variables. NSGA-II parameters for the CO architecture are the same as MDF (see Table 4). In the discipline level, in order to decrease computation time, Sequential quadratic programming (SQP) method was used to solve all optimization problems. The SQP is one of the most successful methods for solving nonlinear constrained optimization problems. This optimization method finds a search direction by solving an approximate problem based on linear approximations of the constraint functions and a quadratic approximation of the objective function. During optimization process in the CO framework, NSGA-II produces system level variables that are transferred to the discipline level optimizers (SQP). These variables are then compared with optimal discipline level variables to produce compatibility constraints.

Fig. 11 Pareto frontiers for the bi-propellant propulsion system design problem within the MDF and CO architectures



4 Results and discussion

4.1 Optimization results

The study of the final Pareto frontier and also design constraints diagrams can help the system engineers to better understand the behavior of the design space. The system engineer can evaluate how much the objectives can be improved and how important are the design variables and quantify also the correlations among the variables themselves. In this case, the designer must choose an optimum design point among these solutions based on customer and mission requirements. Final Pareto optimal sets for the design problem within MDF and CO architectures are presented in Fig. 11 . As can be observed from Fig. 11, the designer is confronted with a set of optimal solutions. In design of propulsion systems, designers usually want to maximize performance and minimize system mass, which are then tempered by some design considerations. In the obtained Pareto frontier, the objective functions are against each other. In other words, when the total

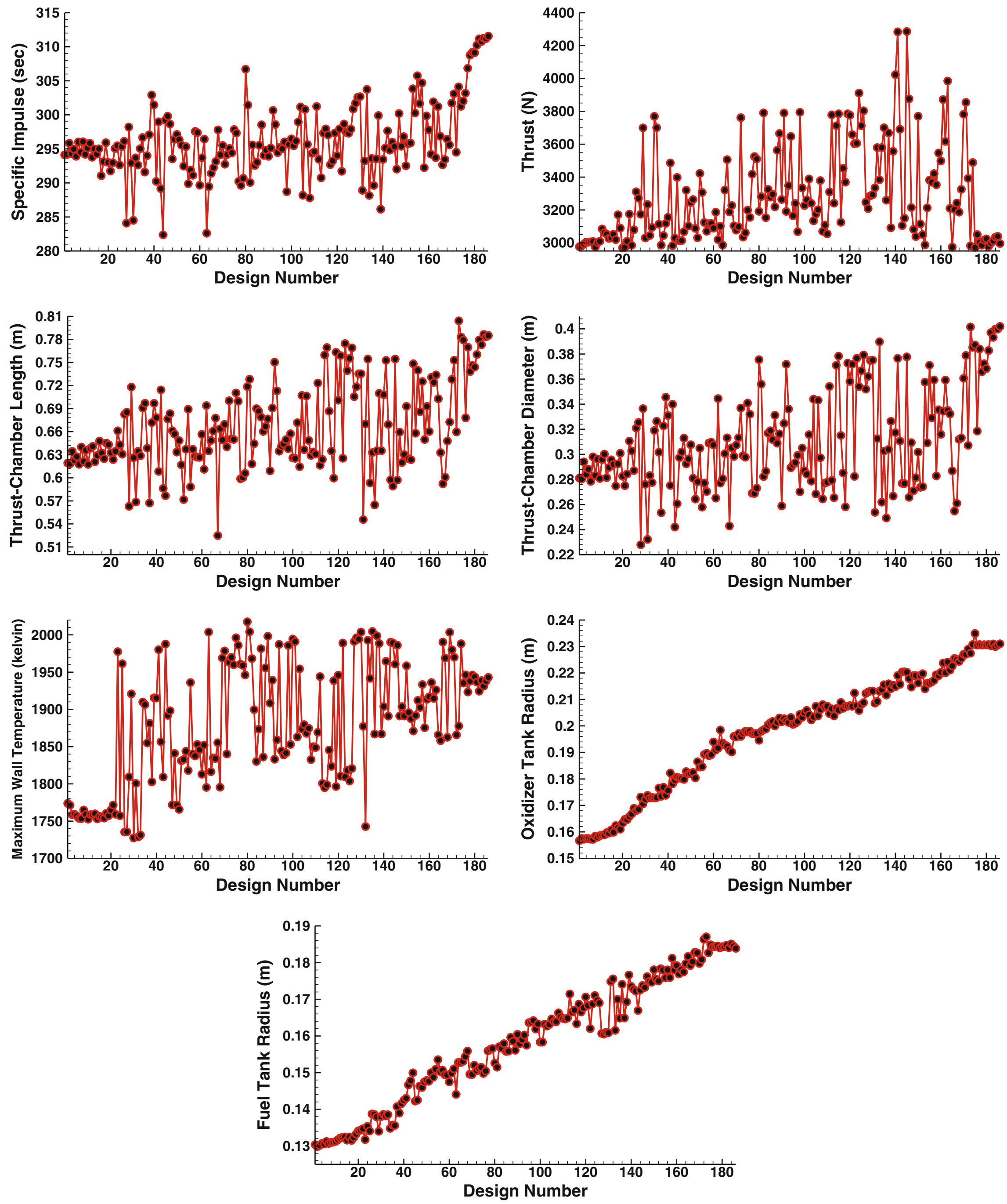


Fig. 12 Design constraints for the Pareto optimal solutions within the MDF architecture

impulse is increased, the total wet mass will increase and vice versa.

As shown in Fig. 11 the Pareto frontiers for both MDF and CO architectures have the same behavior. In the CO architecture, the accuracy of Pareto frontier depends on the value of compatibility constraints. Therefore, Pareto optimal sets for MDF could be more accurate than those generated within CO. The behavior of the design constraints for the Pareto optimal solutions within the MDF and CO architectures are shown in Figs. 12 and 13. As illustrated in these figures, all design constraints were satisfied in the design process. Compatibility constraints for the obtained Pareto optimal solutions within the CO architecture are illustrated in Fig. 14. It should be noted that lower values of these constraints can guarantee matching of discipline level and system level variables. As shown in Fig. 14, in the first discipline level optimization problem, because

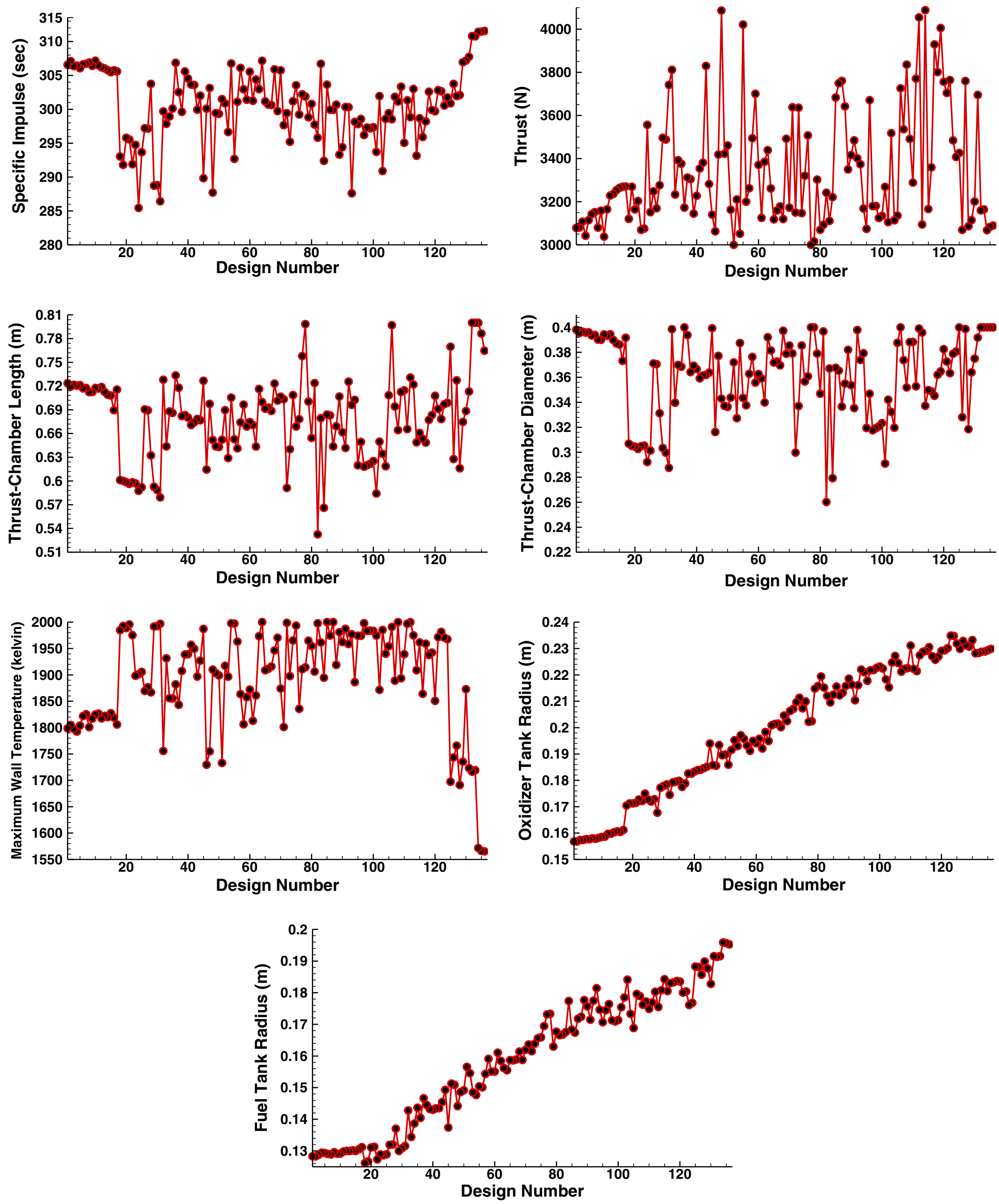
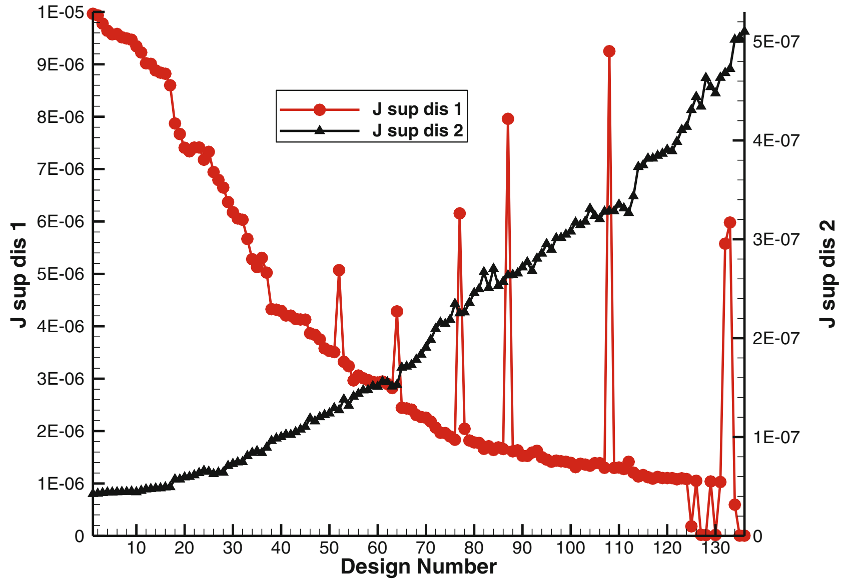


Fig. 13 Discipline level constraints for the Pareto optimal solutions within the CO architecture

Fig. 14 Compatibility constraints for the Pareto optimal solutions within the CO architecture



of higher number of coupling and design variables, satisfying compatibility constraint is more difficult.

4.2 Comparison of the MDF and CO architectures

As described in the previous sub-sections, multidisciplinary design optimization problem of a bi-propellant thruster was solved using MDF and CO architectures. In implementing the design problem, all design constraints were satisfied and a set of solutions in the Pareto frontier was achieved. Final results and selected design characteristics of one of the optimum design candidates for these architectures are presented in Table 5. As can be observed from Table 5, there are similarities between the selected solutions in both methods. For example, in the selected solutions Titanium Ti-6Al-4 V and N2O4/MMH were used as tanks material and propellant type. This leads to increasing performance and decreasing system mass. It should be noted that in the MDF, all design variables are under control by system level optimizer. Because of this feature, the MDF could not be efficient in the detail design of the large scale complex propulsion systems. Relative to the MDF architecture, the advantages of the CO include the ability of decomposition of optimization problem to its subsystems, inherent system modularity and flexibility, no analysis integration requirements, and a significant reduction in communication requirements. These practical advantages make the CO architecture well-suited for use in the large-scale space propulsion systems. Indeed, CO achieves a marked degree of disciplinary autonomy, and the elimination of the local discipline level variables from the system level optimization problem is an attractive feature of CO. Previous studies confirmed the possibility of convergence error in the CO architecture

Table 5 The most important design characteristics for the selected optimal design

No.	Description	MDF	CO	Units
1	Total impulse	240894	240963	N.sec
2	Total wet mass	110.13	108.65	kg
3	Thrust	2986	3663	N
4	Specific impulse (Isp)	303.83	298.63	sec
5	Thrust-chamber mass	6.03	5.653	kg
6	Chamber pressure	9.379	8.24	bar
7	Oxidizer to fuel ratio	1.71	1.895	–
8	Propellant type	N2O4/MMH	N2O4/MMH	–
9	Pressurant gas	Helium	Helium	–
10	Tanks material	Titanium Ti-6Al-4 V	Titanium Ti-6Al-4 V	–
11	burning time	80.65	65.76	sec
12	Expansion ratio	59.92	39.99	–
13	Propellant mass	90.02	90.55	kg
14	Total function calls	302446	2268120	–

(Alexandrov and Lewis 2002). It is necessary to mention that in many design problems, objective functions and constraints are non-smooth or gradients could not be calculated. Therefore, applying gradient-based optimization methods could result in finding a local optima and does not guarantee the global solution. Thus, in this study, for finding a global solution, the GA was used as a system-level optimizer and the SQP was used for optimization in the inner design loop in the CO architecture. The results of optimization with this method are presented in Table 5. In the case of solving MDO problems, function calls can be more dominant than the time of the optimization process. Therefore, in this study, comparison was made on the number of function calls, rather the CPU time. The total number of function calls includes the number of the function calls in the system analysis process and analysis process of the disciplines. For example in the case of CO, the number of function calls for each super-discipline is obtained by summation of the number of function calls in each discipline during discipline level optimization process. Then, the number of function calls for the discipline level optimization problems is calculated by summation of the number of function calls in each super-discipline. Finally, the total number of function calls is computed by adding the numbers of function calls for the discipline level optimization problems together during system level optimization process. In the MDF architecture, system analysis affects the number of function calls. In the presented design optimization problem because of the iterative nature of the system analysis in the MDF, the total number of the function calls was increased to 302,446. However, it was relatively smaller than the total number of the function calls in the CO. Nevertheless, because of the nature of CO architecture, it provides possibilities for solving complex multidisciplinary design problems such as the design of space propulsion systems when using monolithic formulations are difficult or very time consuming.

5 Conclusions

Space propulsion systems exemplify highly integrated systems that suffer from high levels of Computation efforts during design process. The elemental design philosophies for space propulsion systems need fundamental changes; each needs to move from a disciplinary design approach toward a completely integrated approach focused on the system design problem. MDO is the appropriate methodology for this transition. This paper illustrates MDO's successfully application to a spacecraft bi-propellant propulsion system design problem. The MDO methodology was applied to conceptual studies of the bi-propellant space propulsion systems in order to improve the mass and performance capabilities that can fulfill customer and mission requirements. Improving the mass capability means the increasing payload mass capability that can be transferred by the upper stage and also improving the performance capability means increasing the final velocity that can be achieved. Conceptual design of the mentioned system was performed by collaboration of several different analysis modules in the MDF and CO architectures. The results of this investigation show that, in the MDF, function calls are lower than CO, while CO provides the ability of decomposition of optimization problem to its subsystems. Furthermore, compared to monolithic MDO formulations, the presented CO architecture could potentially mitigate some of the difficulties that arise at later stages of the bi-propellant thrusters design process, and reduce design complexities by eliminating of the local discipline level variables from the system level optimization problem. The integration of the analysis modules in the MDF and CO architectures produced a large set of non-dominated optimal designs distributed along the Pareto fronts. This methodology provides an interesting decision making approach to design of complex systems under conflicting goals. In this article, engineering-level analysis codes were used in the conceptual design phase, which can be replaced by high fidelity analysis modules (such as three dimensional CFD or FEA codes) with more capabilities in the detail design phase. In the MDO application of space propulsion systems, because of large number of function calls, employing high fidelity computation tools may be time consuming. Therefore, in order to reduce computation time, expensive analysis codes in the MDO process are suggested to be replaced by surrogate models (such as response surface methodology). It should be noted that the application of MDO methodology in the field of space systems design is still a challenge because of high computation cost and time.

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