MODERN APPROACH TO LIQUID ROCKET ENGINE DEVELOPMENT FOR MICROSATELLITE LAUNCHERS

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ABSTRACT

Microsatellites have been carried to space as secondary payloads aboard larger launchers for many years. However, the secondary payload method does not offer the specificity required for modern day demands such as increasingly sophisticated small satellites that have unique orbital and launch-time requirements. Furthermore, to remain competitive the launch cost must be as low as $7000/kg. With this in mind, SoftInWay Inc. is developing a system engineering approach focused on the design of both the liquid rocket engine turbopump and the entire engine which reduces the duration and cost of development.

In this paper, the approach applied to engineering system development is explained in detail, including modern aerothermodynamics simulation models and structural analysis of the pump and turbine of a turbopump; procedures for turbopump layout selection; secondary flows system simulation; determination and balancing/mitigation of axial and radial loads on the rotor; bearings simulation; rotor dynamics analysis; and mass and dimensions minimization procedures.

The example of the approach application to the design of the liquid rocket engine for microsatellite launcher is presented, including estimations of the performance, mass, and preliminary engine design duration.

INTRODUCTION

Modern satellites could be different in size and application. According to SpaceWorks[1] microsatellite is a satellite with a mass in the range from 10 to 100 kg (Exhibit 1). Some examples of the nano/microsatellites applications are: Earth Observation (HEDE-1, 45 kg); Communication (3-Diamond, 6 kg ea); Scientific (DIDO 1, 5 kg); Technology (D-Sat, 5 kg) etc[1].

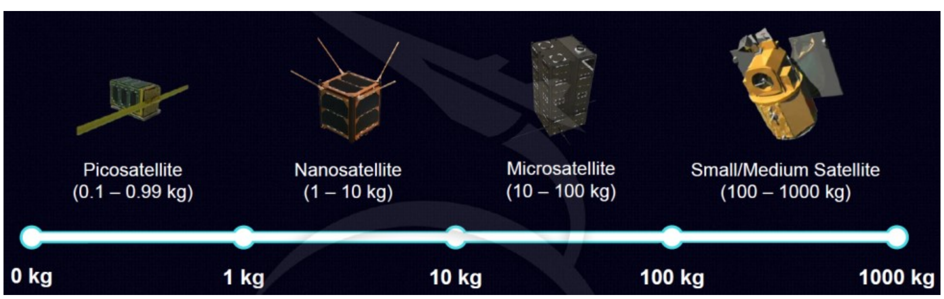


Exhibit 1: Nano/Microsatellite definition[1]

In 2017, over 300 nano/microsatellites were launched. This is a 205 % increase compared to 2016. In turn, SpaceWorks' 2018 forecast predicts 263 nano/microsatellites launches (Exhibit 2). The competition in the launch industry is getting more and more tight and dynamic. For example, there are 25+ companies pursuing the development of new small satellite vehicles. The claimed target launch price is in the range from $12K/kg to $57K/kg[1]. Thus the reduction of the cost is very important to make the vehicle more competitive on the market and attractive to the investors, especially for microsatellites launchers.

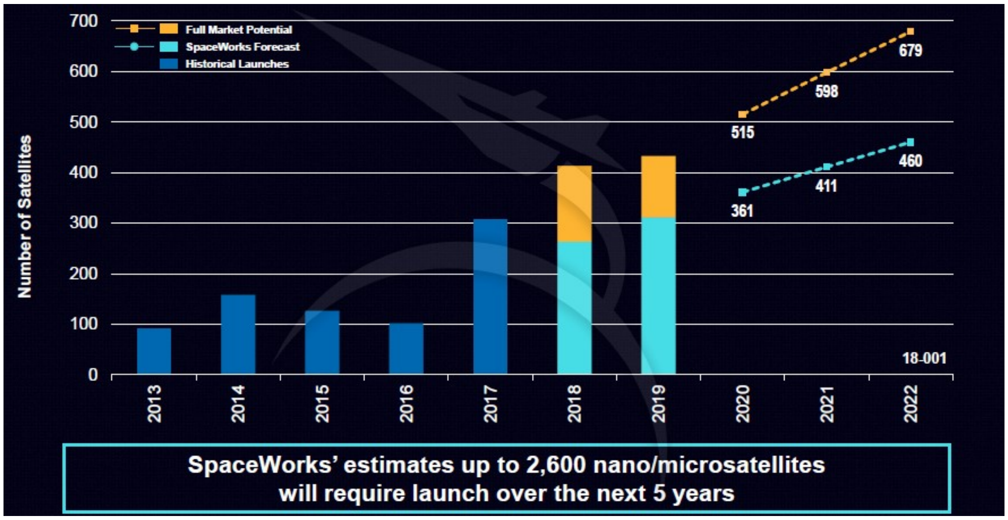


Exhibit 2: 2018 Nano/Microsatellite Launch History & Market Forecast (1-50 kg)[1]

It is obvious that the cost of the launch vehicle consists of the cost of the components. The example of launch vehicle cost breakdown for the first and second stages are presented in Exhibit 3. The Exhibit 3 shows that engine cost for the first stage is the major contributor to the cost of the first stage with structures in second place. For the second stage the avionics & electrical, structures, and the engine contribute approximately equally to the stage cost. In the scope of this paper, the authors will be focused on the reduction of engine development cost.

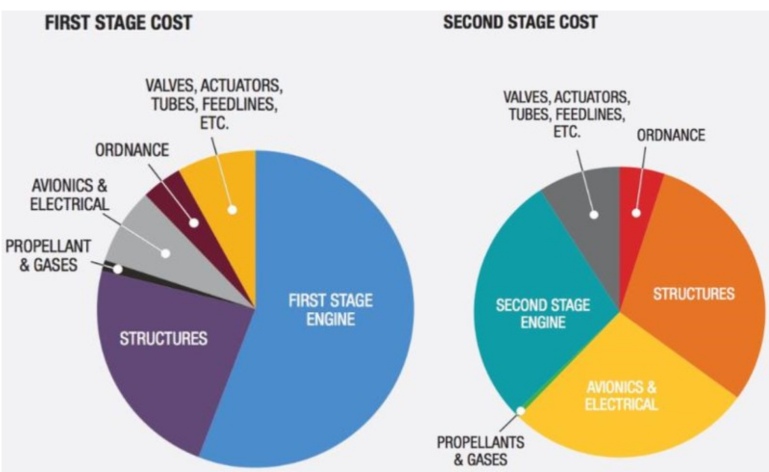


Exhibit 3: Launch Vehicle Cost Breakdown by Major Elements[2]

Together the labor and facilities cost are roughly 50 % of the engine development cost. At that, the facilities cost and labor cost both depends on the time of development. Thus the reduction of engine development duration will obviously decrease the resultant engine cost significantly. The development duration becomes extremely important for minimization of launch cost and for the supply of the specific launcher for a specific need in shortest time. Even the high performance and cost-efficient vehicle may become useless if not supplied on time in such a competitive and dynamic market.

The system design approach applied to rocket engine design is one of the potential ways for development duration reduction. For example, the system design approach based on multidisciplinary optimization technique has already been applied for hybrid propellant rocket engines[3].

The authors of this paper, utilzing system engineering approach, started development of the design system focused on the preliminary design of the entire liquid rocket engine and its components, including turbopump. The system allows reducing the duration and cost of development.

PRELIMINARY ENGINE SPECIFICATION

This paper describes the study of the applicability of the developed design system for the first stage liquid rocket engine preliminary design for microsatellite application. Authors selected a 100 kg payload to SSO as a requirement for the launch vehicle. Gas generator cycle has been selected, due to its simplicity (Exhibit 4).

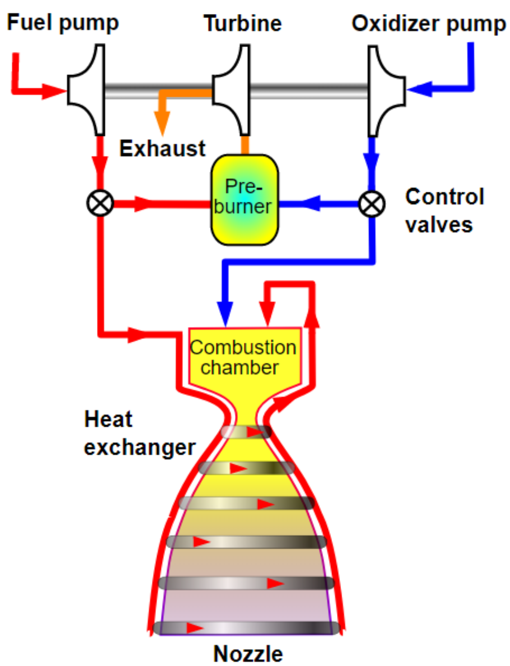


Exhibit 4: Gas-generator liquid rocket engine cycle[4]

For the vehicle with payload of 100 kg to SSO, the estimated thurst for the engine is 50 kN.

LOX-kerosine(RP-1) has been selected as a propellants pair. The choice of propellants is driven by their resulting specific impulse, thrust-levels and tankage-to-propellant mass ratios. Hence, for lower stages, high-density propellants are preferred which yields into both reduced tankage volume and geometrical expansion ratio. For this reason, LOX/kerosene is prefered for first stages over LOX/LH2. Kerosene can be easily stored and refueled, and is a cheap fuel available worldwide. This propellant combination is the most chosen one for private run companies and startups all over the world (Space-X, Rocket Lab Ltd., Virgin Orbital, PLD Space etc.).

The design pressure for exhaust of thrust nozzle has to be selected to maximize specific thrust of the stage and minimize the mass of the engine. Lower values of the design exhaust pressure are good for maximization of specific thrust, however, the dimensions of and specific mass of the engine are getting higher in this case. Generally speaking, this pressure has to correspond to trajectory averaged pressure. In order to determine the averaged pressure, it is necessary to know the expected mission, i.e. trajectory and operation time of the vehicle. For simplicity, the 60 kPa has been selected as a design exhaust pressure. 8 MPa has been assigned as a chamber pressure, which is typical for gas-generator cycles. Oxidizer excess factor (α_ox) has been selected to have maximum Isp at the selected chamber pressure 8 MPa (Exhibit 5). It was determined that optimum α_ox is 0.774. it is not equal to 1, due to the presence of dissociation of the combustion products. The higher the combustion chamber pressure the closer optimum α_ox would be to a stoichiometric one. Thrust chamber Isp at the selected α_ox is 316.59 m/s.

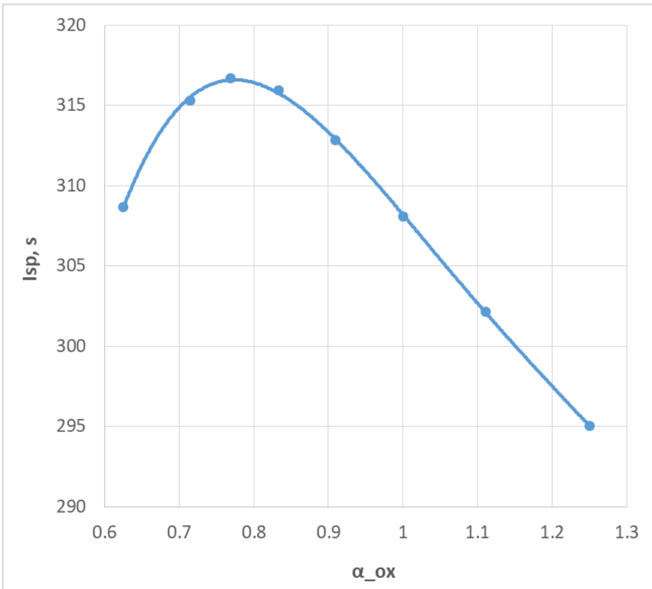


Exhibit 5: Thrust chamber specific impulse vs oxidizer excess factor

Preliminary assessment of pressure drops on the components of the engine was performed. The required discharge pressure magnitudes were determined at the fuel pump, oxygen pump, and turbine inlet. Turbine inlet temperature and gas generator mixture ratio also calculated. These assesments were embedded into the developed design system. The summary of the engine parameters selected for the study are presented in the table below (Exhibit 6).

Parameter	Unit	Magnitude
Payload	kg	100
Thrust	kN	50
Ambient pressure	MPa	0.06
Chamber pressure	MPa	8
Isp_thurst_chamber	s	316.59
Total propellants mass flow rate	kg/s	16.10
RP-1 mass flow rate	kg/s	4.43
Oxygen mass flow rate	kg/s	11.68
Fuel tank pressure	MPa	0.3
Oxygen tank pressure	MPa	0.3
Pressure at turbine exhaust	MPa	0.15
Turbine inlet temperature	K	1046.74
Turbine inlet pressure	MPa	6.46
Oxygen pump discharge pressure	MPa	9.4
Fuel pump discharge pressure	MPa	12.6

Exhibit 6: Preliminary engine specification

DESIGN SYSTEM DESCRIPTION

It is well known that for gas generator cycle engines, the specific impulse of the engine is less than thrust chamber specific impulse[6]. The designer's goal is to determine the highest possible engine specific impulse to maximize payload. In order to find the engine specific impulse, it is necessary to perform a preliminary design of the turbopump of different configurations and select the optimum one.

Preliminary design of the turbopump envisages following activities:

- Preliminary selection of the configuration
- Oxidizer pump preliminary design
- Fuel pump preliminary design
- Turbine preliminary design
- Turbopump preliminary layout development
- Rotor mass/inertia parameters preliminary determination
- Estimation of axial and radial forces on bearings, bearings simulation and rotor dynamics analysis
- Secondary flows (leakages) system analysis and determination of the required amount of propellant for each bearing branch
- Preliminary stress analysis of turbomachinery components

It is obvious that at the step of the preliminary design of pumps it is necessary to take into account leakages flow rates which are not known before secondary flows system analysis. In turn, secondary flows system cannot be simulated if the preliminary layout of the turbopump is not known as well as forces on the bearings. The turbine cannot be designed if the power required to drive pumps is not known and so on and so forth. This logic obviously leads to an iterative process, which starts with some initial guesses of unknown but required parameters that then iteratively determined until there is convergence throughout the entire system.

Taking into account the considerations stated above, the engineering system for preliminary engine design has to integrate the variety of tools for design/simulation of each specific component or subsystem of the turbopump including thermodynamic simulation of the engine in a single iterative process.

The list of the software tools utilized in the flowchart is presented at the end of the paper.

It should be noted that in the scope of this study the combustion chamber design, gas generator design, thrust nozzle design, plumbing and valves design will not be presented as well as the other types of liquid rocket engine cycles.

Process flowchart

The process flowchart integrating all design and analysis processes is presented in Exhibit 7.

The green and red blocks are self-explanatory. Orange rectangles represent script implementation in order to perform additional calculations. For turbopump design, the scripts were used to determine maximal shaft rotational speed calculation that provides an absence of cavitation at inducer and impeller flow path of pumps, total axial loads calculation, heat quantity at bearings and propellant evaporation check in the secondary flow system. Yellow rectangles represent the specific solver required to perform components simulation, such as pumps and turbine preliminary design[*], rotor dynamics[†], 1D hydraulic network[‡], etc. Orange diamond is a condition block required for implementation of loops to converge required parameters with the desired accuracy. Blue-red circles represent sub-processes of pumps design and calculation of mass inertia criteria of pumps blades. Red cube represents CAD tool.

[*] AxSTREAM® Preliminary Design
[†] AxSTREAM® Rotor Dynamics
[‡] AxSTREAM® NET

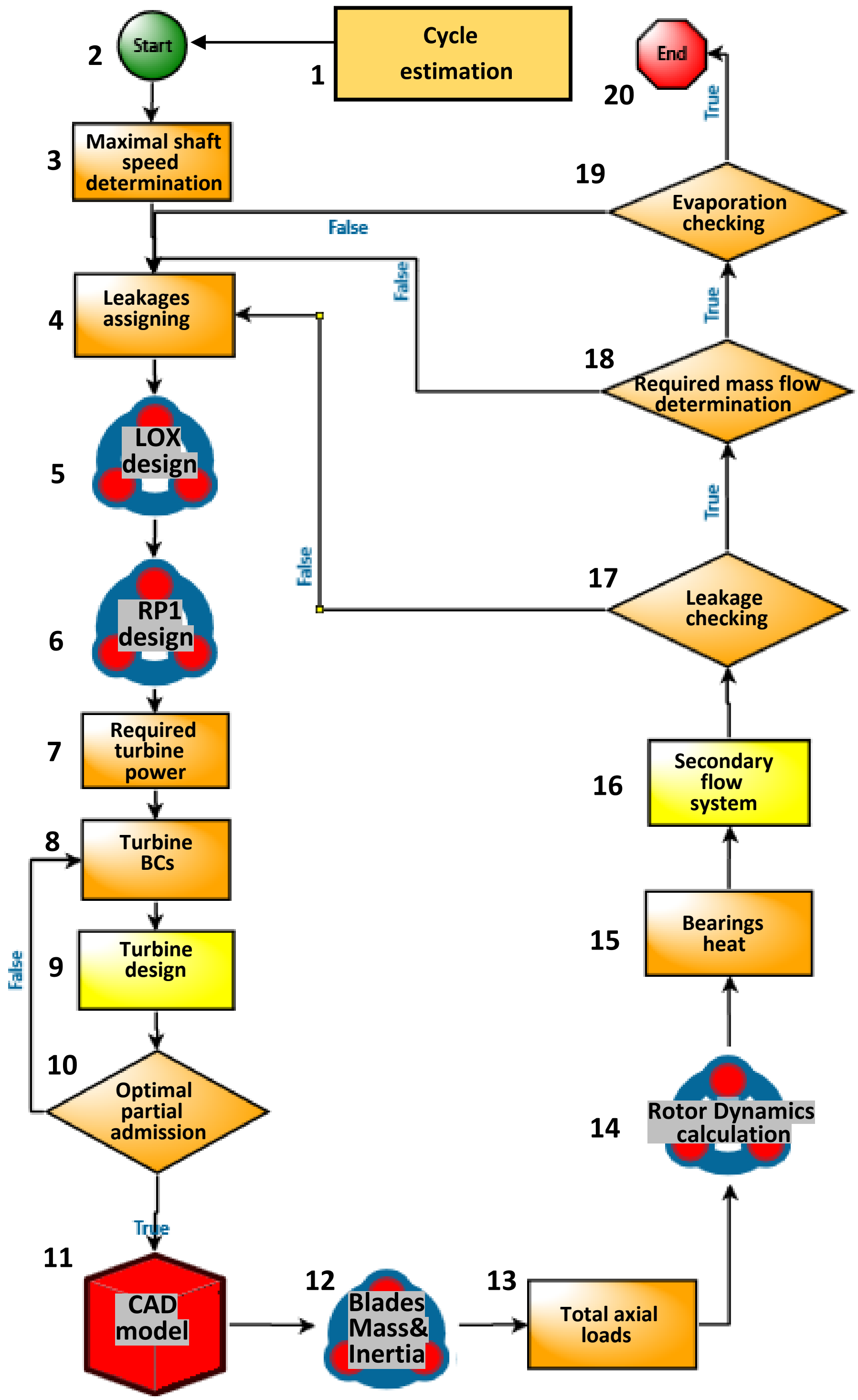


Exhibit 7: System execution processes flowchart

Subprocesses description

The developed system included all design/simulations activities listed in the previous subsection.

The flowchart starts with the cycle thermodynamic simulation (block 1). After the cycle calculation is done the cycle data are transferred to the preliminary design of the turbopump.

Turbopump preliminary design process starts with maximal shaft rotational speed determination (block 3) that satisfy the minimal turbopump sizes satisfying anti-cavitation requirements[5]. As a rule, the oxygen pump is more susceptible to a cavitation[6]. Despite this, the maximal shaft rotational speed in the presented algorithm is performed for both oxygen and RP-1 pumps. The resultant magnitude of shaft speed is determined by the magnitude of the smaller of the two maximum allowable speeds on the pumps. Higher speed is preferable because it allows reducing dimensions and mass of the turbopump.

At the first iteration the leakages mass flow rate (MFR) values at the discharge of both pumps and partially recirculating to the pump inlet, were equal to zero (block 4). They were then iteratively determined through the execution process (block 17).

Next step is the liquid oxygen and RP-1 pumps preliminary design (block 5 and block 6) using preliminary design 1D solver[7]. The values of total pressures at inlet and outlet rotational speed were taken from previous blocks of the flowchart. It should be noted that the design MFR at the first iteration is equal to the value from the cycle analysis and then, on the other iterations the design MFR differs from cycle level one, taking into account secondary flows. The result of pump preliminary design is the geometry and the performance of the pump. Cavitation absence was controlled throughout the entire convergence process.

Primary loss prediction of pump screw inducer is carried out by loss model that includes an accounting of skin friction, flow path diffusion, incidence angle influence, tip clearance loss and secondary (end wall) losses. For impeller design the adapted for pump calculation Aungier loss model was used[8]. The author validated the loss model results with NASA test data and confirmed the model's high ability of loss prediction[8].

Turbine design (block 9) was performed for boundary conditions from cycle thermodynamic analysis to cover pumps consumed power. The turbine was designed according to the methodology[9]. Optimal partial admission ratio and the number of drilled nozzles were determined to provide maximal efficiency of the turbine (block 10).

The preliminary layout of the turbopump was automatically generated in CAD tool[§] (block 11). The example of the generated layout according to the determined dimensions of the pumps and the turbine utilizing some statistical relations for the main turbopump dimensions is presented in Exhibit 8. The algorithm allows generating preliminary layouts of the turbopump of different configurations (single stage and 2-row VC turbine, pumps orientations, single flow, and double flow options). The developed sketch was utilized in the algorithm for mass/inertia parameters determination, secondary flow system dimensions generations and for the visualization of the turbopump configuration. The layout was automatically refined at every iteration.

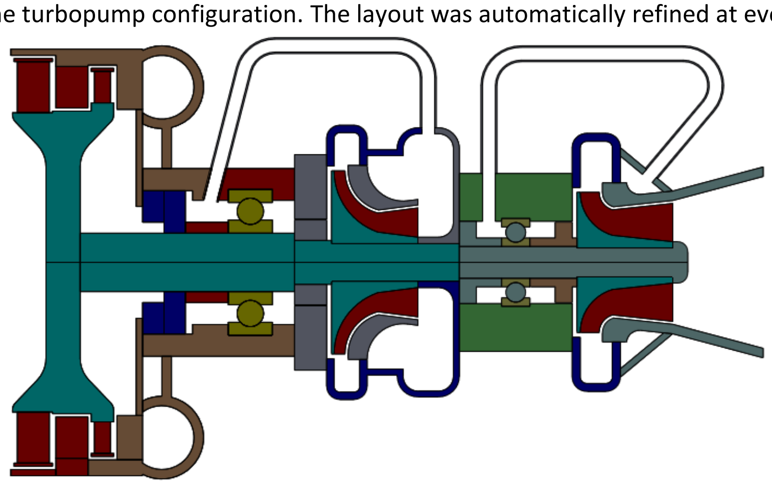
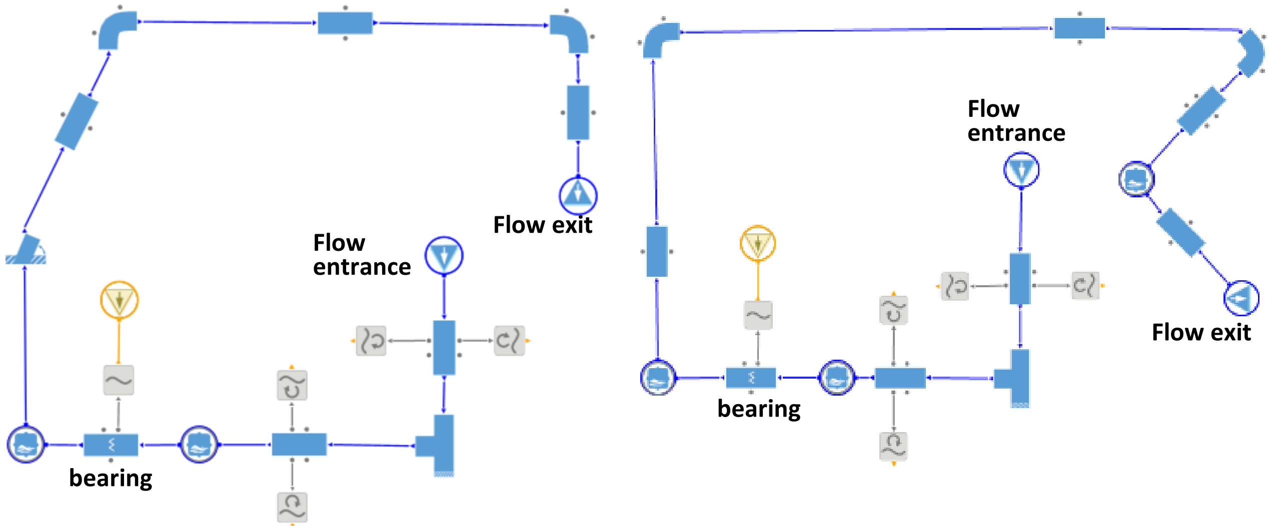


Exhibit 8: Example of preliminary layout of turbopump

[§] Different CAD tools can be used for this purpose. For example SolidWorks, NX etc.

Secondary flow system was modeled to determine the fluid mass flow rate that provides sufficient cooling for reliable work of bearing. For secondary flow system calculation (block 16) was used the hydraulic network analysis tool[10]. The example of calculation scheme of the secondary flow system is presented in Exhibit 9.



RP1 secondary flow branch LOX secondary flow branch

Exhibit 9: Secondary flow system model

The influence of wall rotation and the heating of bearings were accounted for during secondary flow calculations.

Heat quality due to bearings friction (block 15) is determined using turbopump axial load (block 13) and bearings reaction in the radial direction. The radial reaction of bearing was determined using rotor dynamics simulation tool[11] taking into account radial forces induced by circumferentially non-uniform flow admission on the turbine (block 14). For this, the rotor geometry was transferred from the turbopump parameterized CAD model (block 11). The example of rotor model for bearing reaction determination is given in Exhibit 10.

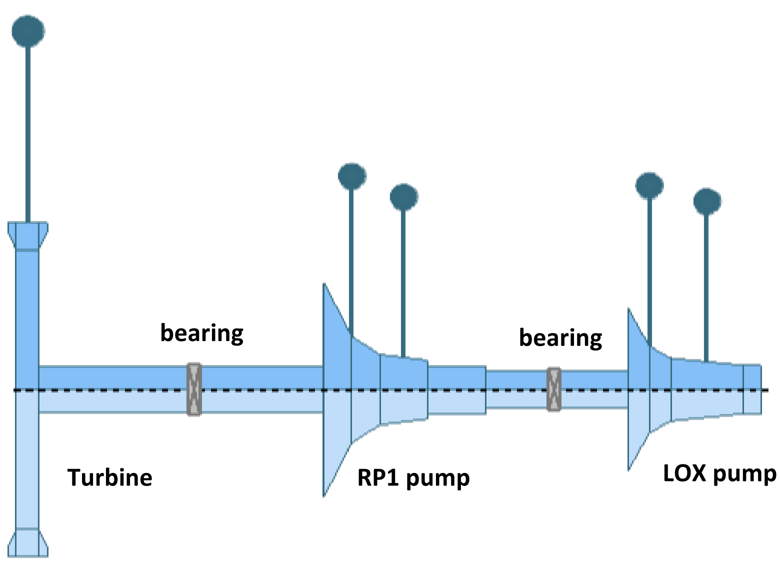


Exhibit 10: Rotor dynamics model

Blue cylindrical and conic sections represent the rotor geometry (Exhibit 10). Dark blue circles simulate the turbine and pumps blades mass and inertia momentum.

Calculation of the blades mass and inertia momentum (block 12) is carried out automatically using CAD tool, and 3D geometry of blades were generated after each pump and turbine flow path redesign (Exhibit 11).

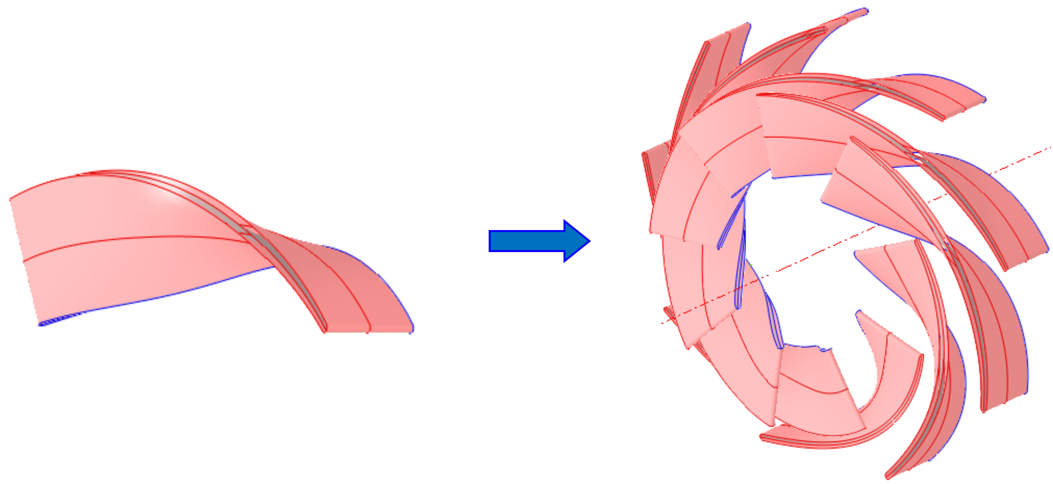


Exhibit 11: Blade export to CAD tool

Preliminary FE stress analysis for the turbomachinery components (presented in block 5 and block 6) was also included in the algorithm. The example of stress analysis for the pump is presented in Exhibit 12. The results of the stress analysis were used for refinement of turbomachinery components geometry.

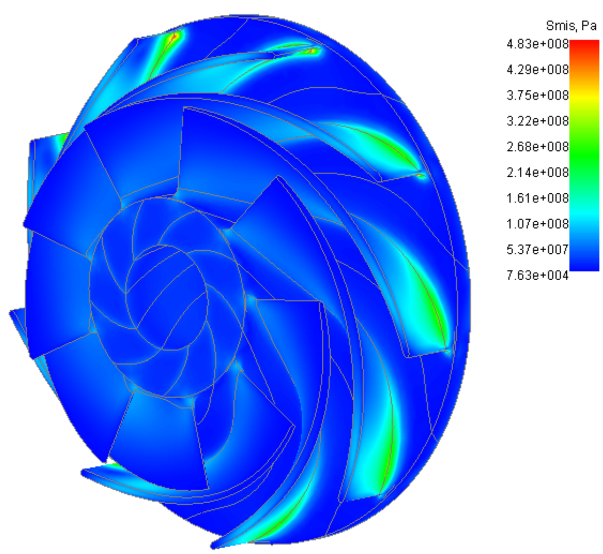


Exhibit 12: Stress analysis for the pump

In this study, 7 different configurations of the turbopump were considered. The developed approach allows switching from a configuration to configuration automatically performing preliminary design steps described above for each configuration and recording crucial performance parameters for the selection of the best configuration. The configurations are presented in Exhibit 13. All configurations have a single rotor, but differently oriented pumps, with different types of flow entry, single flow or double flow oxygen pump types, single stage impulse, and 2-row velocity compound turbine. It should be noted that the preliminary design sysytem is not limited to the presented configurations. Other configurations could be also added, for example dual rotor with gearbox or individual turbine for each pump, electromotor and other. For this respective CAD templates and leakages flow system should be developed.

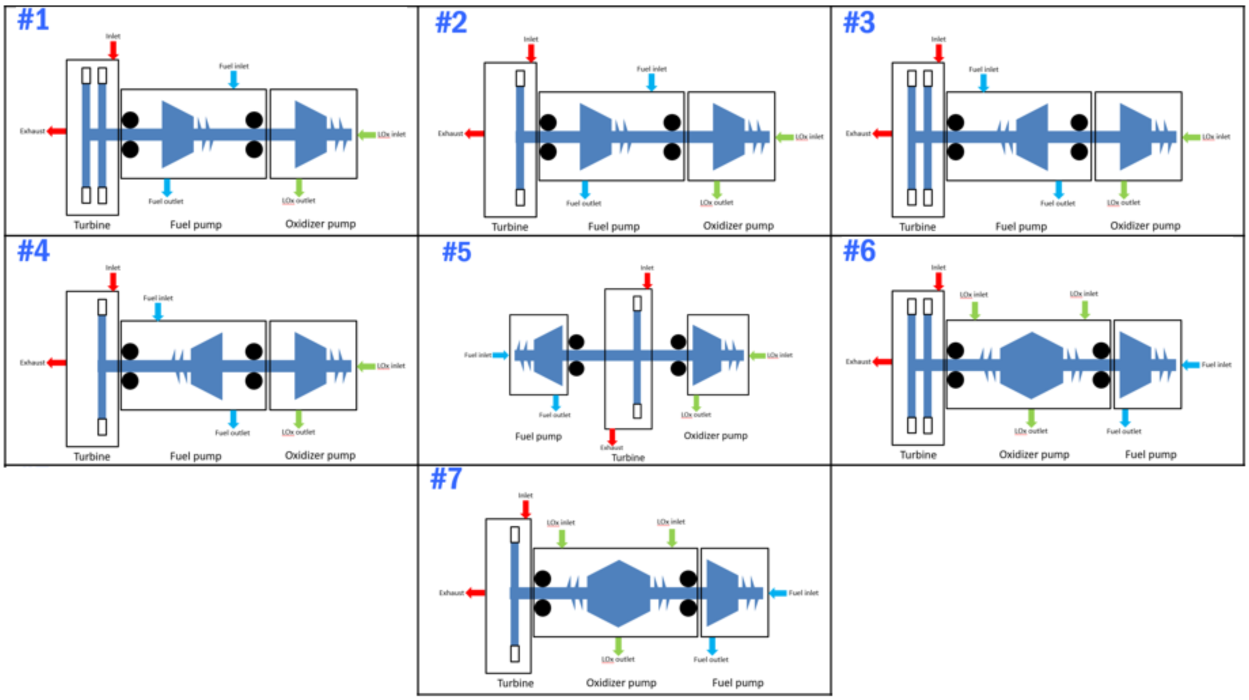


Exhibit 13: Configurations of turbopumps included in the study

TURBOPUMP DESIGN RESULTS

Utilizing the approach described above, the automatic preliminary design and configuration selection has been performed. The required gas generator mass flow rate (turbine mass flow rate) to drive turbopump, turbopump mass and engine specific impulse are presented in the table below (Exhibit 14). Since the chamber pressure and pressure ratio are the same, the chamber mass and dimensions will be the same. Thus the comparison was performed based on the mass of the turbopump and engine specific impulse.

Parameter	Unit	#1	#2	#3	#4	#5	#6	#7
Turbine mass flow rate	kg/s	0.2999	0.3763	0.3080	0.3870	0.3869	0.2983	0.3742
Axial load	N	-14420	-14400	4837	4857	4857	-5406	-5382
Turbopump mass	kg/s	11.97	11.16	12.25	11.81	11.93	12.40	12.15
Turbopump length	m	0.234	0.226	0.232	0.221	0.217	0.233	0.225
Turbopump diameter	m	0.141	0.138	0.141	0.138	0.138	0.106	0.104
Shaft speed	rpm	52485	52485	52485	52485	52485	74243	74243
Isp_engine	s	310.69	309.19	310.54	308.98	308.98	310.73	309.23

Exhibit 14: Engine performance with different turbopump configurations

As we can see, the maximum Isp was obtained for configuration #6. However, the difference of Isp between Configuration #6 and Configuration #1 is rather small and Configuration #1 has a 0.43 kg lighter turbopump. Assuming 200 s as a single firing duration configuration #1 will require 0.3 kg of total propellants mass more than configuration #6, which is less than the difference in turbopump mass. Thus, configuration #1 provides a slightly better combination of Isp and turbopump mass. Another factor to be taken into account is an axial load on the shaft of the turbopump. It is -14420 N for configuration #1 and -5406.34 N for configuration #6[**]. Configuration #6 has a smaller load because of the double flow oxygen pump. The special action[††] could be performed to balance the load for configuration #1 and it still could be considered as a better choice. However, configuration #6 can still be considered. The difference between thrust chamber Isp (Exhibit 6) and engine Isp (Exhibit 12) is 1.9 %. It should be noted that utilization of a dual flow oxygen pump allows to increase rotational speed and make configurations #6 and #7 pretty compact and light weight. The rotational speed for the other configurations is the same because the oxygen pump in all of them has single flow axial inlet, which gives the same maximum allowable speed based on anticavitation criteria.

It is important to notice that the performance of all steps shown in Exhibit 7 required no more than 30 minutes of calculation time for single configuration. Thus, the results shown above were obtained during approximately 3.5 hours. It is obvious that the completion of the iterative process of turbopump preliminary design, including both pumps design, turbine design, turbopump preliminary layout development, secondary flows simulation, bearings simulation, rotor dynamics and stress analysis would require a minimum of a 3 weeks of experienced engineer labor time for a single configuration. Seven configuration results in 21 weeks (840 hours) of a labor time. Of course, one or two configurations could be considered in this case, but it is possible to miss the potential opportunity for engine performance improvement at the very beginning of the engine development. Labor time reduction must not sacrifice the resultant performance of the engine. Thus, the labor time for the preliminary design of the liquid rocket engine was reduced by 240 times utilizing the developed approach. This time reduction not only decreases labor time but also decrease the associated cost and allows the engine to be supplied in a shorter period which is extremely valuable in such a dynamic market.

It should be noted that the proposed design system is easily expandable. It provides the opportunity to perform thrust chamber preliminary design, gas generator design, plumbing routes, turbopump orientation and mounting configurations considerations. After preliminary design, more detailed calculations can be performed as a part of the presented algorithm and reduce the duration of liquid rocket engine detailed design phase. The authors of the paper are planning to continue the work in this direction and present the further results in future papers.

OFF-THE-SHELF SOFTWARE TOOLS UTILIZED IN THE STUDY

The AxSTREAM® Platform was used in the design system, including:

- The AxSTREAM® Preliminary design for turbomachinery preliminary design
- The AxSTREAM NET™ 1D hydraulic networks analysis tool was used for leakage flows simulation
- The AxSTREAM Rotor Dynamics™ and AxSTREAM Bearings™ were used for rotor dynamics and bearings simulation
- The AxSTRESS™ was used for preliminary stress analysis of turbomachinery components
- The AxSTREAM ION™[12] was utilized for the development of the turbopump preliminary design system, including operation flowchart design, optimization, integration of the off-the-shelf and custom software tools, and execution.

ACKNOWLEDGMENTS

We wish to thank the many people from SoftInWay Inc. team who generously contributed their time and effort in the preparation of this work. The strength and utility of the material presented here is only as good as the inputs. Their insightful contributions are greatly appreciated.

[**] Negative sign means that axial load directed toward turbine.
[††] Back face seals radial position adjustment

CONCLUSIONS

1. Microsatellite launch trends and launch vehicles market analysis has been performed and showed that demand is very significant and the market becomes more and more tight and dynamic, so the development duration becomes extremely important as for minimization of launch cost as for the supply of the specific launcher for the specific need in the shortest time. Even the high performance and cost-efficient vehicle may become useless if not supplied on time in such a competitive and dynamic market.
2. The consideration of launch vehicle breakdown by elements was presented, which showed the significant potential for launch cost reduction, shortening the engine development duration and as a result, labor and facilities cost decrease.
3. The system for the automatic preliminary design of a liquid rocket engine was developed, which allows automatic iterative execution of rocket engine cycle analysis and turbopump preliminary design, including fuel pump design, oxidizer pump design, turbine design, turbopump preliminary layout development, secondary flows simulation, bearings simulation, rotor dynamics and stress analysis.
4. The proposed design system is easily expandable, which provides the opportunity to perform thrust chamber preliminary design, gas generator design, plumbing routes, turbopump orientation and mounting configurations considerations and even some additional and more detailed calculations after preliminary design as a part of the presented algorithm and reduce the duration of liquid rocket engine detailed design phase as well.
5. The example the application of the developed system to the preliminary design of a rocket engine, considering gas-generator cycle simulation and turbopump preliminary design of 7 different configurations is presented. It was determined that the configuration #1 provides a better combination of Isp (310.69 s) and turbopump mass (11.97 kg). The difference between thrust chamber Isp (Exhibit 6) and engine Isp (Exhibit 12) is 1.9 %.
6. The labor time for the preliminary design of the liquid rocket engine was reduced in 240 times utilizing the developed approach. This time reduction not only decreases labor time but also decreases the associated facilities cost and allows to supply the engine in a shorter period which is extremely valuable in such a dynamic market.

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