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Selection of a Propellant Feed System for the LPRE of a Small Upper Stage

Abstract. The design of small upper stages must meet many propulsion requirements: repeated restarts, stable operation at low mass flow rate, strict limits on mass and volume, and compatibility with limited ground infrastructure. The central engineering task is a justified selection of the liquid propellant rocket engine feed architecture for a given mission profile and propellant pair, since this choice affects specific impulse, service life, risk, and ground processing effort. This work systematizes propellant feed schemes for a small upper stage and proposes a unified analytical framework that links engine cycle and feed method with propellant selection and tank pressurization modes. Pressure fed and turbopump schemes are treated as mature solutions; pump cycles are used mainly with cryogenic propellants and in high energy demand cases that require high efficiency and compact hardware. In parallel, electro pump schemes are actively studied as a promising direction due to lower mechanical complexity, precise control, and straightforward integration with modern control systems. The analysis shows how chamber pressure, allowable throttling range, restart capability, and the mass and volume metrics of tanks and hardware bound the rational domain of each scheme. The outcome is a set of criteria for early design that maps mission requirements to feed system architecture and supports a technologically feasible choice for small upper stages.
Keywords: pressure-fed system, turbopump-fed supply system, small upper stages, rocket engines, liquid rocket engine.

Introduction

Small upper stages (SUS) are means of deploying spacecraft, installed on the upper stages of a launch vehicle, intended for precise insertion of the payload into the target orbit or for performing inter-orbital maneuvers.

One of the key problems in designing SUS is the selection of an optimal propellant feed system for the liquid-propellant rocket engine (LRE). The correctness of this decision determines not only energy efficiency but also the reliability of the entire system that ensures delivery of the payload to the assigned orbit. The difficulty is that different feed schemes have fundamentally different design features, operational limitations, and efficiency indicators, which makes the choice non-trivial [1, 2].

Several types of feed schemes exist, the principal ones being the pressure-fed (gas-pressurized) system, the turbopump system, and the electric-pump systems that have developed in recent decades. Each has its own design and operational features: the pressure-fed scheme is characterized by simplicity and reliability; the turbopump scheme provides high efficiency when operating with cryogenic propellants and under increased energy demands; whereas electric-pump systems are considered a promising direction due to reduced structural complexity, the possibility of flexible control, and the use of modern battery and power-electronics technologies [3, 4].

The aim of this study is, on the basis of a theoretical review, to substantiate the choice of a feed system for SUS, drawing on scientific and technical sources and the accumulated global experience in the development and operation of space-rocket technology. Particular attention is paid to analyzing the applicability of different feed schemes under constraints on mass, overall dimensions, and cost.

When selecting a propellant feed system for a small upper stage (SUS), not only energetic performance matters, but also factors such as design simplicity, reliability, and technological feasibility under limited manufacturing capability. In world practice three main types are used: pressure fed, turbopump, and electro pump systems. A comparative analysis of these solutions makes it possible to determine the most appropriate propellant feed option for a liquid propellant rocket engine (LRE).

Pressure fed systems have clear advantages in the low thrust segment. They do not require complex turbines, gas generators, or precision pumps, which makes them manufacturable even with constrained tooling and materials capability. Tank mass in such systems is higher than in turbopump systems, but for a SUS where the main LRE thrust does not exceed tens of kN this limitation is not critical. Moreover, hypergolic pairs traditionally used in pressure fed schemes provide simple ignition and multiple restarts, which is ideal for multi burn mission profiles of small upper stages. Practice shows that such solutions underpin attitude control thrusters, apogee engines such as the S5.92 on the Fregat upper stage, and a number of other small upper stages [5, 6].

The principal drawback of the pressure fed system is the need to keep tank pressure close to the operating pressure in the combustion chamber. The propellant tanks become part of the high pressure circuit, so their walls must be thickened or reinforced. This increases the structural mass of the tanks and reduces the stage mass fraction. As a result, part of the payload capacity is consumed by the need to carry heavy pressurant bottles and stronger tanks. For small launchers this trade off can be considered an acceptable price for simplicity, but for larger launch vehicles it becomes unacceptable [7].

Turbopump systems allow operation at much higher chamber pressures than pressure fed schemes. This provides an increase in specific impulse, more complete use of the fuel energy, and a reduction in tank mass, which is especially important for heavy launch vehicles and large upper stages. At the same time the turbopump assembly is one of the most complex elements of a liquid propellant rocket engine. It requires precise rotor balancing, effective cooling, the use of heat resistant materials capable of withstanding extreme temperatures, and a well-developed test infrastructure. For a small upper stage this means a significant increase in cost and technical risk for a relatively modest gain in payload mass. In such conditions the turbopump scheme proves to be excessive [8].

The high complexity of turbopump systems also makes them one of the most vulnerable subsystems of a liquid propellant rocket engine. The turbopump assembly requires high manufacturing precision, complex engineering, and lengthy test campaigns. All elements such as the gas generator, the control system, the pump, the turbine, and other systems must operate synchronously, which increases the probability of failure. A significant share of failures of liquid propellant rocket engines is associated with turbopump modules, including turbine damage, pump cavitation and breakage, shaft seal leaks, and other causes. Therefore the reliability of the pump scheme is influenced by numerous factors, and to achieve the required reliability level, especially for crewed launches, the control system must be made significantly more complex and redundancy must be introduced, which leads to an increase in cost and mass. For a small upper stage the added complexity can be unjustified, and the elevated risk of technical failure negates the performance gain [9, 10].

Electro pump systems are a modern approach that eliminates the gas generator and the turbine and replaces them with electric motors. This reduces structural complexity but requires high specific energy battery packs. The mass of such batteries grows rapidly with engine burn duration and required thrust, which sharply limits their applicability for main propulsion of liquid propellant rocket engines even on small upper stages. In practice such systems are justified in light small sat launchers, for example the Rutherford engine on the Electron launch vehicle by Rocket Lab, where the payload is on the order of tens of kilograms [11].

Material and Methods

Classification of small upper stages

At present, with the rapid development of technology, the trend toward reducing the size of an individual spacecraft (SC) has created a new, in-demand class of hardware on the market [12, 13].

To insert payloads into the target orbit, upper stages equipped with an autonomous control system and a restartable engine are used, which makes it possible to implement complex flight profiles up to multi-burn sequences. Such upper stages are employed on medium- and heavy-class launch vehicles (LV) carrying at least 500…1000 kg of payload [14]. In contrast to larger solutions, a small upper stage (SUS) is developed under strict constraints on mass, packaging volume, and power consumption, while still having to perform the full set of orbital-maneuver tasks such as multiple engine restarts, extended autonomous coast phases, and precise thrust metering. These features make the SUS a critical element in the architecture of very-light-class launch vehicles, where every subsystem must combine minimal dimensions with high reliability and functional completeness [15, 16].

This study focuses on a SUS intended for use as part of a very-light-class launch vehicle, whose mission is to deliver a 300-kg payload to LEO and/or a 150-kg payload to SSO.

The task of placing the SC into its operational orbit is accomplished by an upper stage, which can be classified by purpose, mass, type of target orbit, and propulsion system (Table 1).

Table 1 – Classification of upper stages

Upper stage units	**Purpose**
By appointment	Booster stages of the Launch Vehicle
	Upper stage, performing maneuvering
	Injection (transfer) stages
By type of orbit	Upper stages inserting payloads into LEO
	Upper stages inserting payloads into SSO
	Upper stages inserting payloads into GEO
	Upper stages inserting payloads onto interplanetary trajectories
By weight	Light (up to 3000 kg)
	Medium (3000...10000 kg)
	Heavy (over 10,000 kg)
By engine type	Upper stage with liquid rocket engine (LRE)
	Upper stage with a solid fuel rocket engine (SFRE)
	Upper stage with hybrid rocket engine (HRE)
	Upper stage with electric jet engine (EJE)

When selecting a propulsion system for an upper stage, the key parameters are mass, energy efficiency (specific impulse), reliability, and manufacturability. Their balance determines the success of a mission performed by a small upper stage (SUS) [16, 17].

In our case, the object of analysis is a liquid propellant rocket engine (LRE) considered as the basis for the SUS. Such a unit consists of a combustion chamber in which fuel and oxidizer react, a nozzle that forms and accelerates the flow, propellant tanks for storing the components, and a structural subsystem that ensures strength, stiffness, and reliability [18]. All these parts are integrated by the propellant feed system, which connects the tanks to the combustion chamber and sets the pressure, flow rate, and feed stability. This subsystem largely determines the mass, reliability, and performance of the engine, and for a SUS, where mass constraints and the need for a multi burn mission profile are critical, its selection becomes a key condition for mission success [19].

Table 2 shows the types of LRE propellant feed systems used in upper stages.

Table 2 – Types of LRE propellant feed systems

Feeding system class	**Varieties**	**Typical propellant pairs**	**Thrust range**
Pressure-fed	Cold-gas pressurization	UDMH + NTO (N_2O_4, dinitrogen tetroxide); Hydrazine (monopropellant), RP-1 + LOX	From tens of newtons up to ~20–30 kN
	Hot-gas pressurization		
Pump-fed	Gas-generator (open cycle)	LOX + LH_2; RP-1 + LOX; LOX + LCH_4 (LOX + CH_4); UDMH + NTO (N_2O_4, dinitrogen tetroxide)	From 10 kN to several MN
	Staged-combustion (closed cycle)		
Electric-pump-fed	Electric motor-driven pumps (batteries, fuel cells)	LOX + RP-1; LOX + CH_4; N_2O (nitrous oxide) + RP-1	From hundreds of newtons up to ~25–30 kN

The composition of a two-component fuel-air mixture is determined by the ratio of its oxidizer and fuel components.

The stoichiometric ratio of components is defined as the ratio of moles of oxidizer to moles of fuel in which the valences of the combustible elements are completely replaced by the valences of the oxidizing elements, provided that complete chemical interaction (combustion) of the combustible and oxidizing elements of the fuel occurs. This number of moles of oxidizer per mole of fuel is denoted by æ° and is called the molar stoichiometric ratio of fuel components.

According to the chemical formulas of the conditional molecules of fuel and oxidizer, the molar stoichiometric ratio of components is determined as follows:

$$æ^{\circ} = \frac{\sum_{i=1}^{m} b_{i\,F} V_i}{\sum_{i=1}^{m} b_{i\,Ox} V_i} \tag{1}$$

where m – the number of chemical elements in the fuel composition;

$b_{i\,F}, b_{i\,Ox}$ – the number of atoms of the i-th chemical element in the conventional molecule of the combustible and oxidizer, respectively;

V_i is the highest valence of the i-th chemical element.

The mass and volume stoichiometric ratios of the components are written as follows:

$$K_m^o = æ^{\circ} \frac{\mu_{Ox}}{\mu_F}, \tag{2}$$

$$K_V^o = K_m^o \frac{\rho_F}{\rho_{Ox}}, \tag{3}$$

where μ_F, μ_{Ox} – are the molar masses of the fuel and oxidizer; ρ_F, ρ_{Ox} – are their densities, respectively.

For a specific chemical rocket fuel, $æ^{\circ}, K_m^o, K_V^o$ are constant values.

A fuel mixture with a stoichiometric ratio of components is called stoichiometric.

In a real rocket engine, the stoichiometric ratio of fuel components in the chamber does not, as a rule, provide the required parameters and characteristics of the engine.

The actual (real) mass ratio of fuel components in the engine:

$$K_m = \frac{\dot{m}_{Ox}}{\dot{m}_F}, \tag{4}$$

where $\dot{m}_{Ox}$ and $\dot{m}_F$ – are the mass flow rates of the oxidizer and combustible, respectively.

Using equations (2) and (3), it is easy to determine the actual volume and molar ratios of the components $K_V = \frac{V_{Ox}}{V_F}$ and æ, where V_{Ox} and V_F are the volume flow rates of the oxidizer and combustible, respectively.

In practice, it is more convenient to determine the combustible of a two-component fuel by the excess oxidizer coefficient:

$$\alpha_{Ox} = \frac{K_m}{K_m^o} = \frac{K_V}{K_V^o} = \frac{æ}{æ^{\circ}} \tag{5}$$

When $\alpha_{Ox} = 1$ we have a stoichiometric ration of components in the fuel. If $\alpha_{Ox} > 1$, there is excess of combustible in the fuel, and if $\alpha_{Ox} < 1$, there is an excess of oxidizer.

The mass and volume fractions of oxidizer and combustible in the fuel will be:

$$g_{ok} = \frac{K_m}{1 + K_m} = \frac{\alpha_{ok} K_m^o}{1 + \alpha_{ok} K_m^o}, \tag{6}$$

$$g_{\Gamma} = \frac{K_m}{1 + K_m} = \frac{\alpha_{ok} K_m^o}{1 + \alpha_{ok} K_m^o}, \tag{7}$$

$$r_{ok} = \frac{K_V}{1 + K_V} = \frac{\alpha_{ok} K_V^o}{1 + \alpha_{ok} K_V^o}, \tag{8}$$

$$r_{\Gamma} = \frac{1}{1 + K_V} = \frac{1}{1 + \alpha_{ok} K_V^o}. \tag{9}$$

Pressure fed propellant feed system

In a pressure fed scheme, propellant tanks are made of high strength, leak tight materials, in particular aluminum alloys, metal lined composites, or stainless steel, and are fabricated by welding or brazing to ensure complete structural hermeticity [20]. Sealing of joints and interfaces is an important manufacturing task; to ensure tightness, automated welding or plasma welding, multilevel seals, and vacuum leak testing are used. When expulsion membranes of bellows or flat type are used to isolate the liquid from the pressurant gas, metallic membranes made of corrosion resistant steel or aluminum are the most effective. Some systems use expansion or accumulator tanks, including a spring-loaded bladder made of corrugated elastomer or aluminum foil, which provides closed loop fuel expulsion. These solutions scale poorly, add mass, and are usually incompatible with cryogens, therefore they are not used inside tanks in turbopump assembly (TPA) and electro pump schemes. Figure 1 shows a diagram of the pressure fed propellant feed scheme [21]:

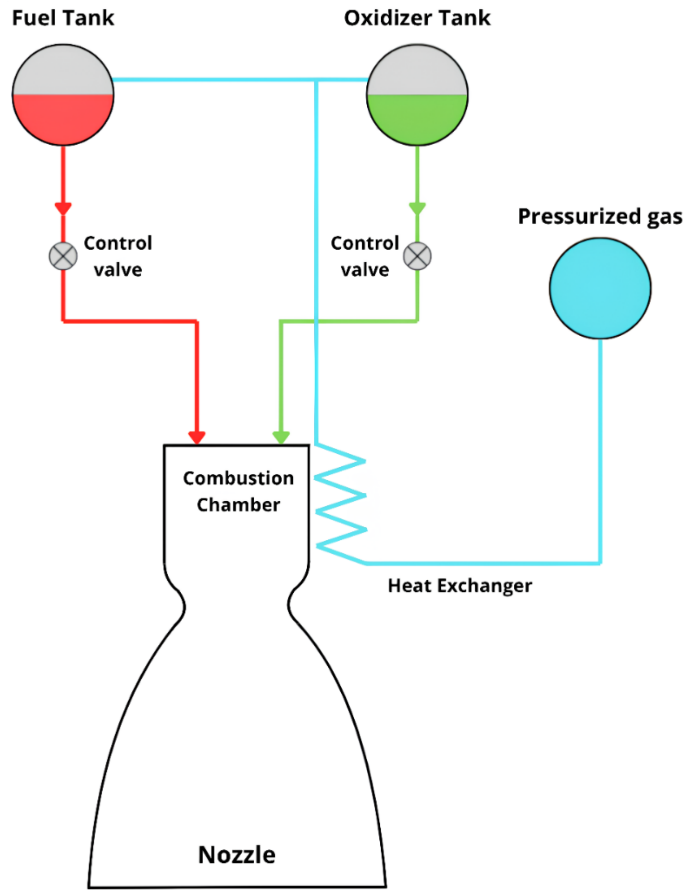

Figure 1 – Schematic of the pressure fed propellant feed system

To maintain pressure, high pressure gas cylinders with helium or nitrogen and gas liquid accumulators are used. Helium is more often used cold, being supplied from a cylinder through a pressure reducing regulator to all tanks of the propellant system. A pressure regulator, either a mechanical two stage unit or a porous pilot valve, maintains the required pressurant gas pressure regardless of gas flow and temperature variations. In a number of systems, preliminary heating of helium is used to prevent chill down and to ensure stable pressure. For this purpose, heat exchangers, regenerative loops in which part of the propellant circulates through a tank jacket or through external heaters, electric heaters, as well as thermal insulation of the tanks including multi-layer vacuum insulation, are used. These measures provide thermal stabilization and reliable operation of the pressure fed propellant feed system [22, 23].

Valves and regulators in the pressure fed system are designed for high precision of pressure control. Spring loaded check valves, orifices, and electrically actuated shutoff valves with minimal deadband and reliable sealing are used. Gas pressure reducers can withstand operating temperatures and the chemical aggressiveness of the gas; they are usually made of titanium or brass and are equipped with filtration and heating to avoid icing [24]. Modern solutions include regulators with an automatic bypass valve and dual sealing, designed with temperature gradients and vibration in mind. In general, the pressure fed system provides a comparatively simple design but requires careful selection of tank materials, membranes, and valves, as well as means of thermal stabilization for the pressurant and the propellant components [25].

Pressure fed propellant feed systems are among the simplest and most reliable solutions, widely used in low thrust engines and apogee liquid rocket engines. The classification is based on the type of working medium used to pressurize the tanks and on the method of its generation. In engine design, fuel feed systems are distinguished as operating on a cold working medium and on a hot working medium. In the first case, inert gases are used as the working medium, supplied either through pressure reducers or by direct expansion. A special group consists of vapor pressurization schemes where the working medium is formed by evaporation of dedicated liquids. Systems using a hot working medium, by contrast, are based on heated gas obtained in a gas generator or a chemical reactor, which provides higher tank pressure but increases structural complexity [22].

A structured presentation of the variants of pressure fed systems is given in Table 3.

Table 3 – Classification of pressure-fed propellant feed systems

<table>
<tr><th>Classifications</th><th>System variants</th><th colspan="2">Features</th></tr>
<tr><td rowspan="4">Cold-gas pressurization</td><td rowspan="2">Stored-gas</td><td colspan="2">Regulated (pressure-reducing regulator)</td></tr>
<tr><td colspan="2">Direct expansion</td></tr>
<tr><td rowspan="2">Evaporative</td><td colspan="2">With stored pressurant liquid</td></tr>
<tr><td colspan="2">Self-pressurizing (self-displacement)</td></tr>
<tr><td rowspan="6">Hot-gas pressurization</td><td>Chemical</td><td colspan="2">Use of reaction products as the pressurant</td></tr>
<tr><td rowspan="3">Gas-generator-based</td><td colspan="2">Single-component (monopropellant)</td></tr>
<tr><td rowspan="2">Two-component (bipropellant)</td><td>oxidizer-rich</td></tr>
<tr><td>fuel-rich (reducing)</td></tr>
<tr><td rowspan="2">Heated-gas</td><td colspan="2">Heating of components</td></tr>
<tr><td colspan="2">Heating the pressurant to the required tank pressure</td></tr>
</table>

Turbopump propellant feed system

Turbopump propellant feed systems are among the most widespread solutions and are actively used in upper stages because they can provide high chamber pressure and thereby achieve high energetic performance of the engine [23]. Depending on how the gas generator working medium is used, such systems are divided into several variants of process organization. The most common are the open and the closed schemes. In the open scheme, the gas generator combustion products, after driving the turbine, are discharged to ambient and do not contribute to thrust, which simplifies the design and increases reliability but reduces the overall efficiency of the engine. In the closed scheme, the spent gas is routed to the combustion chamber for afterburning together with the main propellants, which makes maximum use of the fuel energy and increases specific impulse, but requires more complex technical solutions and increases sensitivity to operating conditions. The schemes for implementing the turbopump propellant feed system are shown in Figure 2 [21].

The turbopump assembly (TPA) includes axial or centrifugal pumps for fuel and oxidizer that are driven by a gas turbine on a common shaft or on separate shafts. In practice, coaxial dual pump layouts are used, an example being Centaur with the J 2S engine, where both impellers are mounted on a single shaft with two bearing supports. To reduce axial loads, symmetric pump arrangements with opposed flows or special balance devices that create counterpressure on the disk are employed. In a number of projects, in particular in dual shaft VK D-160 layouts, each pump is driven by its own turbine wheel through a gearbox. The TPA shaft is made of alloy steel or a titanium alloy with high mechanical properties. Gas or hydrostatic bearings are selected for minimal losses and speeds up to 100 to 200 thousand rpm [24].

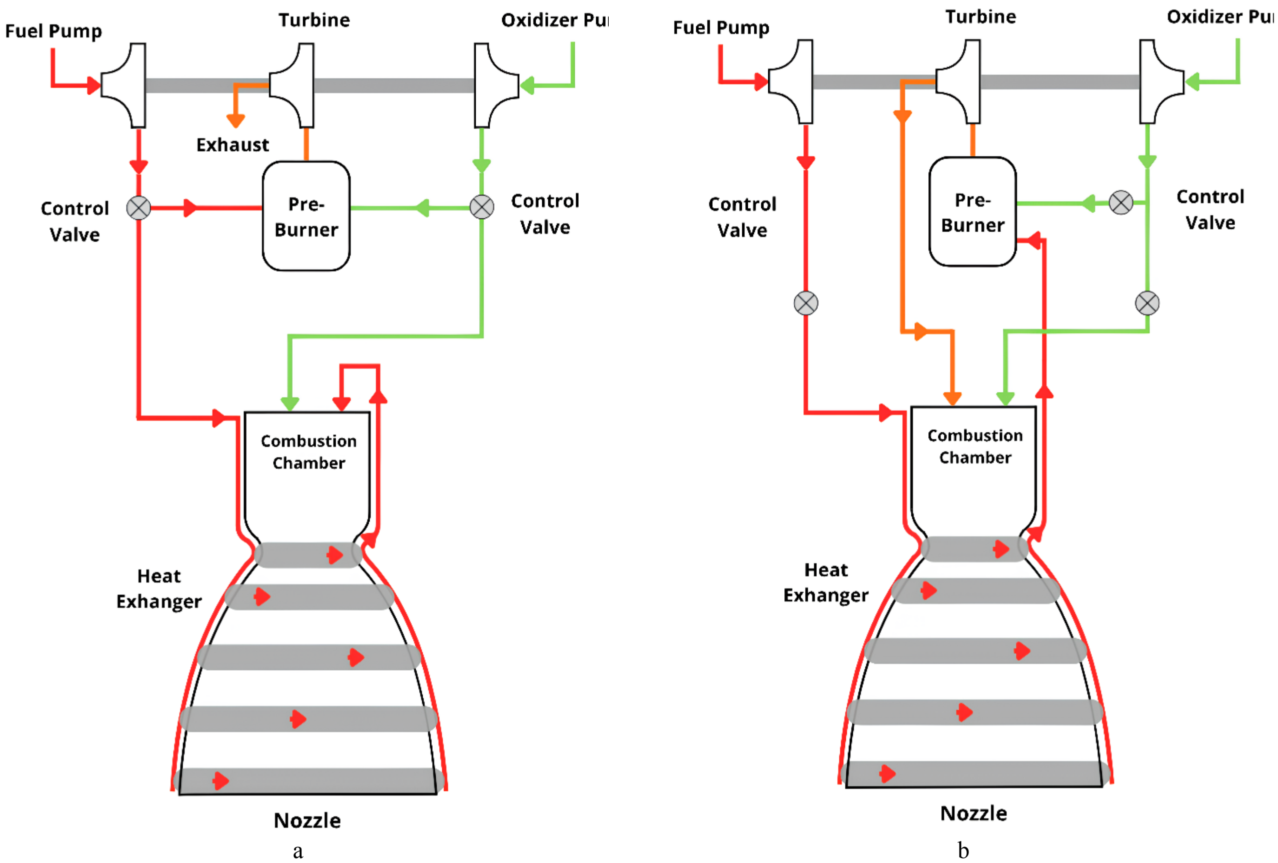

Figure 2 – (a) Schematic of the open cycle turbopump propellant feed system; (b) Schematic of the closed cycle turbopump propellant feed system

Pump impellers are made of heat resistant alloys. For pumping liquid oxygen and kerosene, stainless and nickel alloys are usually used, for example 12Kh18N10T and EI 868, and for pumping UDMH and other hypergolic propellants, titanium alloys such as VT5 and VT6 are used due to their low mass and sufficient strength [25, 26]. A screw centrifugal impeller that combines the functions of inducer and main stage is often used, which makes it possible to dispense with a gearbox. To thermally isolate the pump unit and prevent the pump from overheating from the turbine, shielding shrouds and thermal insulation coatings are used, up to multilayer metal glass ceramic. In liquid hydrogen engines, heat suppression and prevention of condensation are achieved by a vacuum jacket or vapor phase barrier coatings [26, 27].

Depending on the method of process organization, pump systems are divided into several varieties. The most common are gas generator schemes, open and closed, which differ in the use of gas generator products. Alternatives are expander cycles, where the working fluid for the turbine is formed due to evaporation of the fuel in the cooling path, as well as autogenous schemes that use heated components to drive the turbine or to pressurize the tanks. In addition, the classification takes into account the composition of the gas generator, single component or two components [28]. A structured summary of the varieties of pump systems is given in Table 4.

Table 4 – Classification of turbopump-fed propellant feed systems

Classification	System variants	Features
By use of gas-generator products	Open scheme (no afterburning)	Gas-generator products after the turbine are dumped overboard; simpler design, higher reliability, but lower overall engine efficiency
	Closed scheme (with afterburning)	Gas-generator products after the turbine are routed to the main chamber for afterburning; higher efficiency, increased design complexity.
By method of generating turbine gas	In a single-component liquid gas generator using fuel	Turbine gas produced by thermal decomposition of the fuel component.
	In a single-component liquid gas generator using an oxidizer	Gas generation via oxidizer decomposition.
	In a two-component liquid gas generator (fuel +oxidizer)	Conventional method; may be oxidizer-rich or fuel-rich (reducing)
	Propellant gasification (expander-based)	Evaporation/boil-off of liquid fuel/oxidizer used to drive the turbine
By handling of spent generator gas (for open cycles)	Discharge through an auxiliary nozzle	Generator gas contributes additional thrust
	Discharge overboard without nozzle	Simplified arrangement, reduced efficiency
By number of TPAs	Single TPA	Unified unit feeding both components
	Dual TPAs (separate for fuel and oxidizer)	Increased reliability and control flexibility
	Twin TPAs, clustered solutions	Used in high-thrust LPREs with multiple chambers

Electric-pump-fed propellant feed system

In electro pump schemes of engines, conventional pumps are driven by an electric motor, usually a brushless PMSM or BLDC, through a power drive and battery packs. The power electronics architecture includes an inverter, a PID speed controller, and an engine control unit (ECU). The electric motor can be mounted on the main flange of the engine or remotely with a shaft drive using a universal joint. Power is supplied by high energy rechargeable batteries, with preference given to lithium ion or lithium polymer cells due to their high specific energy. The batteries are equipped with monitoring and thermal protection, including a battery management system (BMS) with active cooling or thermal radiators, because overheating or overcharge can lead to failure. The thermal regime of the electrical power system is demanding. During operation the motors deliver power up to tens of kilowatts, which generates heat in the windings and in the electronic section. Liquid or forced air cooling of the electronics is usually used, and exposed engine surfaces are coated with a radiative thermal coating. A schematic of an LRE with an electro pump propellant feed system is shown in Figure 3 [29].

During rotation of the rotor of a high-power electric motor, significant gyroscopic torques and centrifugal forces arise. Therefore, the rotor mass and its inertia are carefully optimized. High coercivity magnets are used, and rotor balancing is performed with tolerances on the order of microns. Electromagnetic constraints include the need to shield power cables and to filter electromagnetic interference in order to prevent disturbances in avionics and navigation [30,31]. To reduce interference, common mode grounding together with ferrite filters is often used. In systems with several pumps, maintaining identical rotational speeds is achieved by synchronizing the controllers over a CAN or Ethernet bus, which ensures balanced propellant delivery. Methods for balancing the pumps include flow sensors in each main line and software based leveling of drive power [32].

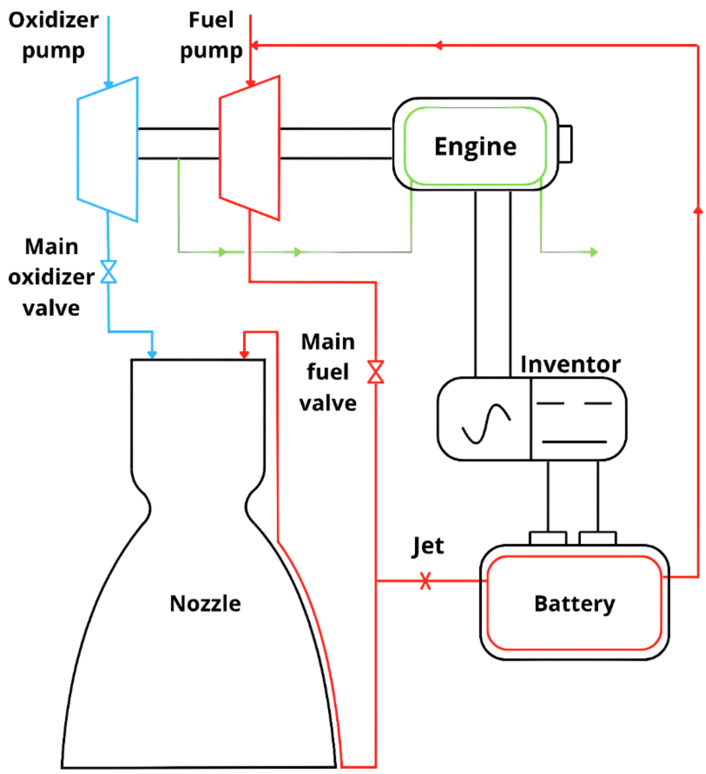

Figure 3 – Schematic of an LRE with an electro pump propellant feed system

A drawback of electro pumps is the high specific energy demand of the power supply, especially the batteries. In engines with electric drive, such as on Electron, it has been noted that the battery mass reduces the system mass margin. At the same time, electro pumps readily provide wide thrust control ranges by changing the voltage or frequency supplied to the electric motor, that is by an electronic throttle. In combination with compact electronics and fault tolerance with redundant cells and parallel strings, this makes electro pump schemes promising for small upper stages under limited ground infrastructure [33]

The energy balance model of the electric pump circuit additionally takes into account battery degradation due to two mechanisms: cyclic and calendar aging. The available capacity is described by the approximation:

$$C_a = \\ = C_0\left(1 - k_c * \sqrt{N} - k_{cal} * f(T_s) * t\right), \qquad (10)$$

where C_0 – is the nominal capacity of new cells;

N- is the number of deep discharge cycles;

t – is the storage time in the mission;

$f(T_s)$ – is the temperature factor;

k_c and k_{cal} – are empirical coefficients identified by the passport data of the selected chemistry.

The increase in internal resistance is calculated using the expression:

$$R_i = R_0\left(1 + \alpha\sqrt{N} + b * g * T_{Op}\right), \qquad (11)$$

where R_0 – is the initial resistance of the cells;

T_{Op} – is the operating temperature;

α and β – are coefficient reflecting the contribution of cycles and temperature, respectively.

Taking into account C_a and R_i the required installed battery energy is determined as:

$$P_e = \left(\dot{m} * \frac{\Delta p}{\eta_p}\right) * \frac{1}{\eta_m * \eta_i}, \qquad (12)$$

$$E_i = \gamma_r * \int P_e dt, \qquad (13)$$

where $\dot{m}$ – total mass flow rate, Δp – required pump head, η_p – pump efficiency, η_m – electric motor efficiency, η_i – power inverter efficiency, γ_r –

reserved factor, including degradation over the entire mission profile.

Results and discussions

Based on the analysis performed and the literature review, we systematized the three principal propellant feed schemes used in small upper stages (SUS): pressure fed, turbopump, and electro pump. For each scheme, we considered design features, operational constraints, and application experience in domestic and international projects. This section presents the results of a comparative analysis of these systems with emphasis on their applicability under the mass, volume, and power resource constraints of a small upper stage, and it also discusses their development prospects in the context of current and future technologies. Table 5 presents a comparative analysis of all three systems.

Table 5 – Comparative analysis of all three systems

Criterion	Pressure-fed system	Turbopump-fed system	Electric-pump-fed system
Structural mass	Tank mass is substantially higher because the tanks must withstand high pressure; acceptable for low-thrust applications.	Tank mass is lower, but a heavy turbopump assembly appears; optimal for medium and high thrust.	Tank mass is moderate, but a significant share goes to batteries.
Specific impulse (in vacuum)	285–315 c	320-465 c (depending on the fuel pair: LOX/RP-1 or LOX/LH_2)	300–340 c (LOX/RP-1, LOX/CH_4)
Thrust range	From tens of Newtons to 20-30 kN	From 10 kN to 3-4 MN	From 0.1 to 25-30 kN
Reliability	Very high, minimum moving parts	High if the design is well-developed, but sensitive to TPA failures.	Fairly high; depends on power-system and cooling reliability.
Manufacturability	Simple layout does not require a complex production base	High technological complexity requires a well-developed test and production base.	Simplified layout but limited by availability of high-energy power sources.
Multiple restarts	5-20 cycles (no battery limits)	5–10 cycles (limited by ignition system and TPA reliability).	3–6 cycles (limited by power budget and thermal loading).
Applicability to SUS	Most rational for ultralight launchers under a constrained technical base.	Effective, but excessive and risky for ultralight projects; justified for medium and heavy upper stages.	Promising for very small upper stages but limited by battery capacity and scalability.

The comparison of the three systems shows that, for the development of a small upper stage (SUS), the most rational choice is the pressure fed propellant feed system. It combines structural simplicity, high operational reliability, manufacturability, and the possibility of multiple restarts, which is especially important for a SUS. Despite limited energetic performance, this scheme provides practical feasibility of the project under constrained production and test infrastructure.

A parametric assessment of the applicability of three supply schemes was carried out, taking into account the characteristic conditions of the small booster mission. Low thrust modes, multiple restarts, long passive flight segments, and cryogenic temperatures were considered.

At low thrust, the key limitations of the electric pump scheme are heat dissipation in the electric machine and inverter during long periods of operation, as well as the energy capacity of the battery in the payload. For the turbo pump scheme, the requirements for cavitation margin at the pump inlet with a compact tank layout and for the minimum stable mode of the gas generator or turbine at low flow rates are dominant. For the displacement scheme, the main factor is the increase in the mass of the cylinders and tank walls as the working pressure increases, which limits efficiency during prolonged operation.

With multiple restarts, the electric pump scheme wins out with its flexible power profile and lack of a gas generator thermal cycle, but it requires consideration of battery degradation and the reliability of the power electronics. The turbopump scheme requires justification of stable ignition and repeated return to mode, while the displacement scheme places increased demands on thermal stabilization and valve service life.

During long passive periods in cryogenic modes, evaporation losses, component stratification, and the operation of liquid intake devices become decisive factors. The same assumptions regarding LAD, thermal insulation, and permissible gas phase levels were applied to all schemes. As a result, it was shown that the choice of scheme is determined not only by energy and mass metrics, but also by the cryogenic management architecture, which allows for reliable start-up after exposure.

Conclusions

The review and comparative analysis have shown that the selection of the propellant feed system is a key factor in the design of a small upper stage (SUS). The options considered, pressure fed, turbopump, and electro pump systems, have different advantages and limitations that determine their applicability. Turbopump schemes provide the best energetic performance, however their complexity, high cost, and demanding production base make them excessive for very light launch vehicles under limited technical resources. Electro pump systems are of interest as a promising direction that simplifies the design by eliminating the gas generator and the turbine, however their use is limited by the high mass of battery packs and by insufficient technological maturity. The most rational choice for a SUS is the pressure fed propellant feed system, which provides simplicity of implementation, high reliability, and the possibility of multiple starts. Despite comparatively modest energetic performance, this system matches the project goals, namely minimization of technical risks, adaptation to existing production capabilities, and assurance of successful operation to meet the stated objectives.

An electric pump circuit is preferable when there are restrictions on ground infrastructure, requirements for a flexible thrust profile, and multiple restarts, provided that the energy balance is maintained, taking into account the aging of the power source and ensuring the thermal regime of the power unit. The turbo pump scheme is rational at elevated pressures in the chamber and long active periods, provided that engineering and test bench facilities are available for the gas generator part and cavitation resistance is confirmed. The displacement scheme is optimal in scenarios with moderate pressures and limited duration of operation, where simplicity, high reliability, and minimal preparatory infrastructure are priorities.

Each of the recommendations is formulated in terms of a set of mission constraints, including total ΔV, number of restarts, duration of active phases, permissible energy and heat budgets, and ground infrastructure requirements.

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