VERTEX PROPULSION SYSTEMS
TECHNICAL SPECIFICATION DOCUMENT

Aerospace Propulsion Subsystem

Design Documentation

Nimbus-X Liquid Methane/Oxygen Engine Thrust Chamber Design

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Abstract

This document presents the thermodynamic, aerodynamic, and physical design specifications for the thrust chamber of the **Nimbus-X** liquid oxygen and liquid methane rocket engine. Developed by **Vertex Propulsion** for **Nexa Corporation's Synapse-3** vehicle upper-stage, the engine targets a nominal vacuum thrust of 350 kN and specific impulse of 355 s. The design employs a contraction ratio of 3.5, a throat diameter of 158.9 mm, and an 80% bell nozzle contour with an expansion ratio of 80 to optimize vacuum performance. Standard Method of Characteristics (MOC) contour calculations and 1D isentropic relationships are presented alongside detailed dimensional tables and a chamber cross-section. Regenerative cooling thermal margins are also verified under peak throat heat fluxes of $85.0\ \mathrm{MW/m^2}$, proving structural viability.

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1 Introduction

The development of high-performance launch vehicles requires optimized liquid rocket engines capable of high reliability and high specific impulse. Under the Nexa Corporation launch vehicle program, the **Nimbus-X** engine is being developed by **Vertex Propulsion** (a division of **Apex Aerospace**) as the primary propulsion subsystem for the second stage of the **Synapse-3** medium-lift launcher.

The Nimbus-X is designed as a liquid oxygen (LOX) and liquid methane (LCH_4) gas-generator cycle engine, combining modern clean-burning propellant benefits with moderate chamber pressures to reduce complexity and system weight [4]. Methane was selected over liquid hydrogen (LH2) due to its higher density (reducing tank volume), lower boil-off rates, and suitability for long-duration orbital coast phases.

1.1 Document Scope

This document details the engineering specifications, thermodynamic calculations, and physical design configurations for the Nimbus-X thrust chamber. It covers:

- System requirements and performance specifications.
- Detailed combustion chamber sizing and thermochemical analysis.
- Nozzle contour design and supersonic expansion characteristics.
- Thermal management and coolant flow distributions.

1.2 Fictional Design Context

The development program is managed under the **Meridian Aerospace Alliance**, with testing conducted at the **Nimbus Test Range** in New Mexico. The engineering data and parameters listed in this document represent the baseline design freeze (Revision 4.2), intended for verification by the systems engineering board.

2 Subsystem Specifications

The operational requirements for the Nimbus-X engine are derived from the Synapse-3 upper-stage mission profile. The engine must provide high specific impulse in vacuum while maintaining restart capability for orbital insertion and circularization maneuvers.

2.1 Performance Parameters

Table 1 outlines the primary design specifications for the Nimbus-X rocket engine at its nominal operating point. These values serve as the inputs for the thermodynamic and geometric calculations detailed in subsequent sections.

Table 1: Nimbus-X engine design specifications and operational limits.

Parameter	Symbol	Value	Unit
Vacuum Thrust	F_{vac}	350.0	kN
Sea-Level Thrust (Nominal)	F_{sl}	305.5	kN
Vacuum Specific Impulse	$I_{\text{sp,vac}}$	355.0	s
Combustion Chamber Pressure	P_c	10.0	MPa
Nozzle Exit Pressure	P_e	8.5	kPa
Oxidizer-to-Fuel Ratio	O/F	3.40	–
Total Propellant Mass Flow Rate	$\dot{m}$	100.56	$\mathrm{kg\,s^{-1}}$
Chamber Gas Temperature	T_c	3450	K
Nozzle Expansion Ratio	ϵ	80.0	–
Design Burn Duration	t_{burn}	450.0	s

2.2 Propellant Feed and Ignition System

The engine utilizes a dual-impeller turbopump driven by a single-shaft gas-generator turbine. Liquid oxygen and liquid methane are pressurized and routed to the injector head. Ignition is achieved via a spark-torch igniter situated in the center of the injector face, providing reliable multiple-restart capability.

The injector utilizes a co-axial shear design with 180 elements arranged in concentric rings. This configuration ensures rapid mixing and atomization of the cryogens, achieving a combustion efficiency (η_c) of 98.5% under nominal chamber conditions.

3 Thermodynamic Calculations & Chamber Sizing

This section details the primary aerodynamic and thermodynamic calculations performed to establish the baseline geometry of the Nimbus-X thrust chamber.

3.1 Mass Flow Rates and Exhaust Velocity

Using the design vacuum thrust and specific impulse, the total mass flow rate $\dot{m}$ and effective exhaust velocity v_e are calculated. The effective exhaust velocity in vacuum is:

$$v_{e,\text{vac}} = I_{\text{sp,vac}} \cdot g_0 = 355.0 \text{ s} \cdot 9.80665 \text{ m/s}^2 = 3481.36 \text{ m/s} \tag{1}$$

The total propellant mass flow rate is:

$$\dot{m} = \frac{F_{\text{vac}}}{v_{e,\text{vac}}} = \frac{350{,}000 \text{ N}}{3481.36 \text{ m/s}} = 100.535 \text{ kg/s} \tag{2}$$

With an oxidizer-to-fuel ratio $O/F = 3.40$, the individual mass flow rates for the oxidizer ($\dot{m}_{\text{ox}}$) and fuel ($\dot{m}_{\text{fuel}}$) are determined by:

$$\dot{m}_{\text{fuel}} = \frac{\dot{m}}{1 + O/F} = \frac{100.535}{4.40} = 22.849 \text{ kg/s} \tag{3}$$

$$\dot{m}_{\text{ox}} = O/F \cdot \dot{m}_{\text{fuel}} = 3.40 \cdot 22.849 = 77.686 \text{ kg/s} \tag{4}$$

3.2 Throat Area Sizing

The throat area A_t is sized to choke the flow under nominal combustion chamber conditions. The thrust coefficient C_F is calculated from compressible flow relations as a function of the specific heat ratio γ, expansion ratio ϵ, and pressure ratio [2]. For the LOX/Methane combustion products at $O/F = 3.40$, the specific heat ratio is modeled as $\gamma = 1.22$. The theoretical vacuum thrust coefficient $C_{F,\text{vac}}$ is given by:

$$C_{F,\text{vac}} = \sqrt{\frac{2\gamma^2}{\gamma - 1}\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \epsilon\left(\frac{P_e}{P_c}\right) \tag{5}$$

Substituting $\gamma = 1.22$, $\epsilon = 80$, $P_c = 10.0$ MPa, and $P_e = 8.5$ kPa:

$$C_{F,\text{vac}} = 1.765 \tag{6}$$

Thus, the throat area required is:

$$A_t = \frac{F_{\text{vac}}}{P_c \cdot C_{F,\text{vac}}} = \frac{350{,}000 \text{ N}}{10{,}000{,}000 \text{ Pa} \cdot 1.765} = 0.01983 \text{ m}^2 \tag{7}$$

The corresponding throat diameter is:

$$D_t = \sqrt{\frac{4A_t}{\pi}} = 0.1589 \text{ m} = 158.9 \text{ mm} \tag{8}$$

3.3 Combustion Chamber Geometry

The combustion chamber volume is characterized by the characteristic length L^*, which represents the chamber volume divided by the throat area:

$$L^* = \frac{V_c}{A_t} \tag{9}$$

For liquid oxygen and liquid methane, an L^* value of 1.15 m is selected to ensure complete combustion and vaporization of the droplets within the chamber [3]. From Eq. (9), the chamber volume is:

$$V_c = L^* \cdot A_t = 1.15 \text{ m} \cdot 0.01983 \text{ m}^2 = 0.0228 \text{ m}^3 = 22.8 \text{ L} \tag{10}$$

The contraction ratio $\epsilon_c = A_c/A_t$ is chosen as 3.5 to limit gas velocities at the injector face and prevent combustion instabilities. This yields:

$$A_c = \epsilon_c \cdot A_t = 3.5 \cdot 0.01983 = 0.06941 \text{ m}^2 \tag{11}$$

$$D_c = \sqrt{\frac{4A_c}{\pi}} = 0.2973 \text{ m} = 297.3 \text{ mm} \tag{12}$$

The chamber length L_c (excluding the convergent section) is then computed as 0.28 m based on standard conical convergence angles.

4 Nozzle Contour & Aerodynamic Design

The expansion of the combustion products from the throat to the exit plane is designed using the Method of Characteristics (MOC) to produce a shock-free, optimized parabolic contour (Rao Bell nozzle) [1]. This minimizes divergence losses while keeping nozzle length and mass within stage limits.

4.1 Geometric Profile

The expansion ratio $\epsilon = A_e/A_t = 80.0$ yields a high area expansion suited for vacuum operation. The exit diameter D_e is calculated as:

$$D_e = D_t\sqrt{\epsilon} = 158.9 \text{ mm} \cdot \sqrt{80} = 1421.2 \text{ mm} = 1.421 \text{ m} \tag{13}$$

The nozzle convergent section uses a standard 60° half-angle convergence, and the divergent section starts at the throat with an initial expansion angle $\theta_i = 18.2°$ and terminates at the exit plane with a terminal angle $\theta_e = 8.5°$. The overall length of the nozzle divergent section is 1.85 m, representing an 80% bell nozzle.

4.2 Thrust Chamber Contour Schematic

Figure 1 illustrates the cross-section of the Nimbus-X thrust chamber, detailing the injector plane, throat section, and divergent bell profile.

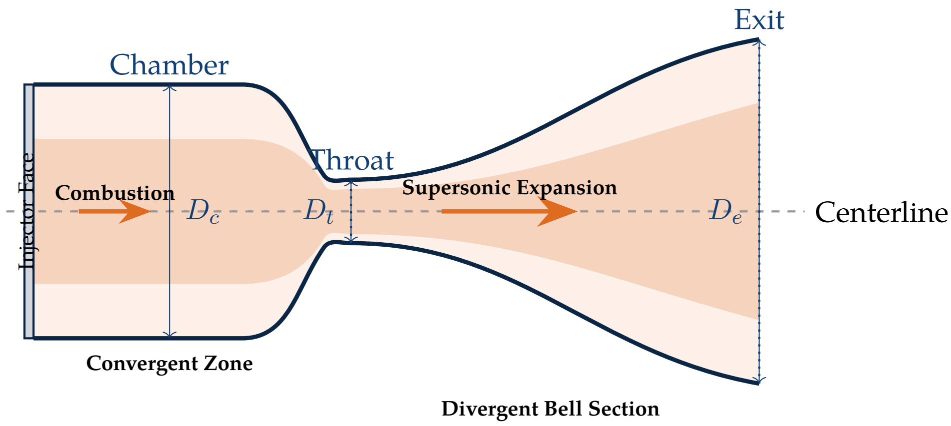

Figure 1: Cross-sectional schematic and flow path of the Nimbus-X thrust chamber.

4.3 Mach Number Profile

As the combustion products expand through the convergent-divergent nozzle, the flow transitions from subsonic ($M < 1$) in the chamber to sonic ($M = 1$) at the throat, and supersonic ($M > 1$) in the divergent section. The relationship between Mach number M and area ratio A/A_t is determined by the 1D isentropic relation:

$$\frac{A}{A_t} = \frac{1}{M}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{\frac{\gamma+1}{2(\gamma-1)}} \tag{14}$$

For $\gamma = 1.22$ and exit area expansion $\epsilon = 80.0$, this yields an exit Mach number of:

$$M_e = 4.62 \tag{15}$$

resulting in high kinetic energy conversion and directed momentum.

5 Thermal Management & Implementation Plan

The extreme thermal environment in the combustion chamber ($T_c = 3450$ K) and throat area requires active cooling. The Nimbus-X utilizes a regenerative cooling scheme where the liquid methane fuel is routed through 120 milling channels inside a copper-alloy liner (NARloy-Z) before entering the injector head.

5.1 Heat Flux Profile and Cooling Design

At the throat, the heat flux reaches a peak value of $q = 85.0$ MW/m^2. Calculations show that a coolant velocity of 22.5 m/s through the NARloy-Z channels is sufficient to keep the wall temperature below the structural limit of 900 K. The outer structural jacket of the thrust chamber is constructed of electroformed nickel to withstand the high mechanical loads and operating pressures ($P_c = 10.0$ MPa).

5.2 Development Schedule & Verification Milestones

The Nimbus-X development program is currently in the detailed design phase. Table 2 details the upcoming milestones leading to flight qualification on the Synapse-3 launch vehicle.

Table 2: Nimbus-X developmental schedule and testing milestones.

Milestone	Description	Target Date	Status
PDR	Preliminary Design Review	Nov 2025	Completed
CDR	Critical Design Review	Mar 2026	Completed
HFT-01	Subscale Injector Hot-Fire Testing	Oct 2026	In Preparation
SFT-01	Full-Scale Engine Static Fire (Sea Level)	Feb 2027	Planned
VFT-01	Vacuum Chamber Certification Test	Jul 2027	Planned
FQR	Flight Qualification Review	Oct 2027	Planned

5.3 Document Sign-Off & Approval

This design documentation package is approved by the leading propulsion architects and systems engineering managers for the Synapse-3 program.

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