Advanced Materials Bi-propellant Rocket (AMBR)
Engine Information Summary for Discovery Missions
April 2010

Summary: The Advanced Material Bi-propellant Rocket (AMBR) engine is a high performance (I_{sp}), higher thrust, radiation cooled, storable bi-propellant space engine of the same physical envelope as the High Performance Apogee Thruster (HiPAT™). To provide further information about the AMBR engine, this document provides details on performance, development, mission implementation, key spacecraft integration considerations, project participants and approach, contact information, system specifications, and a list of references. The In-Space Propulsion Technology (ISPT) project team at NASA Glenn Research Center (GRC) leads the technology development of the AMBR engine. Their NASA partners were Marshall Space Flight Center (MSFC) and the Jet Propulsion Laboratory (JPL). Aerojet leads the industrial partners selected competitively for the technology development via the NASA Research Announcement (NRA) process.

1. Introduction

1.1 Background

While the need generally exists for higher performance propulsion systems, the component technologies have to mature for any new, successful engine development to take place. In September 2006, a NASA/industry joint effort was initiated to boost the (specific impulse, I_{sp}) performance of Aerojet's HiPAT™ engine. The motivation was to attain a more efficient, storable bi-propellant engine that will benefit future NASA's planetary science missions. By increasing the specific impulse and thrust, the more efficient engines can enable near-term missions, enhance their science capability and returns, reduce mission cost, and cut transit time. The developmental effort is called AMBR, which stands for the "Advanced Material Bi-propellant Rocket" where the "advanced material" refers to the iridium (Ir)-coated rhenium (Re) combustion chamber fabricated using the EL-Form™ process.

The AMBR engine development aims for two major objectives:

1. higher specific impulse engine performance
2. lower fabrication cost for the iridium/rhenium combustion chamber

To initiate the effort, NASA Marshall Space Flight Center (NASA-MSFC) and NASA Jet Propulsion Laboratory (NASA-JPL) conducted mission-level and system-level studies to translate the target engine performance into spacecraft performance. Four conceptual missions were selected and used for the analyses based on the current scientific interest, launch vehicle capability, and trends in spacecraft size:

- GTO to GEO, 4800 kg, ΔV for GEO insertion only ~1830 m/s
- Enceladus Orbiter (Titan aerocapture) 6620 kg, ΔV ~2400 m/s.
- Europa Orbiter, 2170 kg, total ΔV ~2600 m/s
- Mars Orbiter, 2250 kg, total ΔV ~1860 m/s

Applying the original target AMBR specific impulse of 335 seconds (approximately seven seconds higher than the state-of-the-art), the study shows a 23 percent payload gain for the Mars Orbiter mission. Similar payload gains are also evident for the other missions. Additional AMBR engine improvements include the increased thrust level as compared to the 100 lbf baseline engine HiPAT™ with expected performance benefits for deep gravity well missions. The chamber fabrication is estimated to be reduced by 30 percent relative to the HiPAT™ engine, achieved through the higher production yield rate and lower rhenium materials cost associated with the combustion chamber. Additional cost savings are anticipated due to other design and processing changes that have not yet been quantified.

Hotfire performance verification for the AMBR flight-like, developmental prototype engine took place in October of 2008. It was followed by environmental (shock and vibration). Additional envelope testing took place in February and long duration testing in June 2009**.** The development plan was completed by the end of FY2009 with the AMBR engine prototype demonstrated in a relevant-ground environment to TRL 6 for a range of SMD missions.

The NASA In-Space Propulsion Technology Project Office contracted the AMBR engine development through a NASA Research Announcement (NRA) Cycle 3a contract (contract number NNM06AA93C) with the Aerojet Company at Redmond, WA. Other contributors to the effort are:

- Jet Propulsion Laboratory performed the mission and benefits analysis and the prototype shock test.
- NASA Marshall Space Flight Center performed the AMBR propulsion system analysis, and the high temperature refractory metal material analysis and testing.
- NASA Glenn Research Center managed the AMBR development since late 2006.
- Plasma Process, Inc., Huntsville, AL (PPI) performed the Ir/Re chamber fabrication.

The NASA Science Mission Directorate (SMD) funds the In-Space Propulsion Technology (ISPT) Project Office located in the NASA John H. Glenn Research Center in Cleveland, Ohio.

In 2007, SMD directed the project to close out the AMBR developmental activities while allowing the remaining effort to take the final product to a potential TRL of 6. Decision was made to eliminate the NTO/MMH engine performance demonstration in favor of more TRL 6 validation activities for the NTO/N2H4 propellants, reason being that the advancements in latter can be transferred across the propellant combinations. The direction change also added environmental and long-duration testing for the NTO/Hydrazine engine.

1.2 System Summary

The AMBR engine is a high performance bipropellant engine using the iridium/rhenium chamber technology in an attempt to obtain 335 seconds specific impulse (I_{sp}) with nitrogen tetroxide (NTO) and hydrazine (N2H4) propellants. AMBR engine aimes to benefit significantly interplanetary missions by enabling reduced launch weight and/or increased payload and reducing propulsion system cost. Figures 1 and 2 are a line drawing containing physical dimensions and a color graphic of the AMBR thruster.

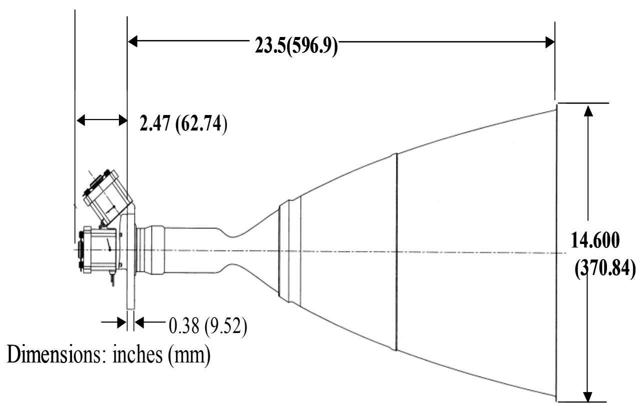

Figure 1. AMBR Thruster Physical Dimensions

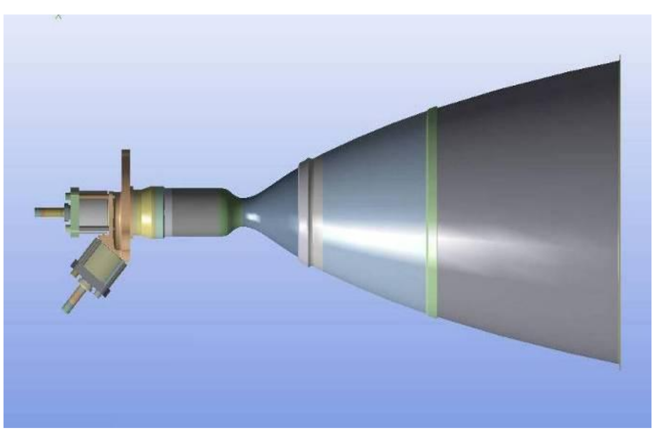

Figure 2. A Color Rendition of the AMBR Thruster

AMBR engine development originally targeted the following specifications:

- 335 seconds steady-state I_{sp} with NTO/N2H4 (by test)
- 3-10 years mission life (by analysis & similarity)
- one hour operating (firing) time (by test)

By July 2009, the single-iteration AMBR design has been hot-fire tested and results are given in Table 1 which shows AMBR design characteristics, the test-result, and the Aerojet's HiPAT™ Dual Mode engine, which is the baseline for AMBR development.

Table 1: AMBR Characteristics Demonstrated Compared with the Baseline HiPAT™ Thruster

ENGINE CHARACTERISTICS	AMBR Design	AMBR Tested	HiPAT™ DM
Thrust (lbf)	200	140	100
Specific Impulse (sec	335	333	328
Inlet Pressure (psia	275	250	250
Oxidizer/Fuel Ratio	1.2	1.1	1.0
Expansion Ratio	400:1	400:1	375:1
Physical Envelope		Within existing HIPAT envelope	
Propellant Valves		Existing R-4D valves	

Due to insufficient cooling in the first- and single-iteration prototype, AMBR could only achieve 140 lbf of thrust at a lower oxidizer inlet pressure of 250 psia. Serendipitously, this lower inlet pressure accommodates the use of existing, available propellant tanks and subsystems and thus improves AMBR's nearer-term applicability for New Frontiers and Discovery class missions. The original 200 lbf thrust goal was not achieved.

AMBR is capable of operating at a temperature of 2470ºK.[4] The iridium/rhenium combustion chamber enables radiation cooling which sustains efficiency. It is fabricated using the advanced and cost reducing electroform process called EL-Form™. This process was selected after evaluating a group of candidates--Chemical Vapor Deposition (CVD), electroforming (El-Form), Low Pressure Plasma Spray (LPPS) and Vacuum Plasma Spray (VPS).

Of the processes listed above, the well understood CVD is the incumbent process used to fabricate the R-4D-15 HiPAT™ thrust chambers. The only other process that has been used to fabricate an Ir/Re chamber for a bipropellant engine was El-Form™. It was used successfully in 2004 to fabricate and test the Aerojet's development engine R-42DM. Finally, neither LPPS nor VPS were ever used; therefore, they were dropped from consideration due to the lack of technical maturity.

The Figures of Merit used for the decision matrix were:

- Cost – Nonrecurring
- Cost – Recurring
- Schedule – Nonrecurring
- Schedule – Recurring
- Producibility
- Performance – Mechanical Properties
- Performance – Thermal
- Performance – Oxidation Resistance
- Performance – Mass
- Heritage/Risk – Design
- Heritage/Risk – Manufacturing

Weighting factors were assigned to the Figures of Merit based on the primary performance goals of the program.

For AMBR, the PPI El-Form™ process was finally down-selected due primarily to the lower development unit costs and production cost estimates. The El-Form™ process carries more process risk than CVD since its development is less mature than CVD. However, the added risk is deemed worth the potential rewards in reduced costs.

Figure 3 below is a top-level schematic for a representative dual mode AMBR propulsion system. The system is "dual mode" because the same spacecraft fuel system supplies both the main engine and the Attitude Controlling System (ACS) thrusters (specific impulse of 210 seconds). Because of the similarity, the name ACS is interchangeable with Reaction Control System (RCS). This AMBR system and its components are designed and sized to enable assessments for potential mission benefit brought by the system. [4]

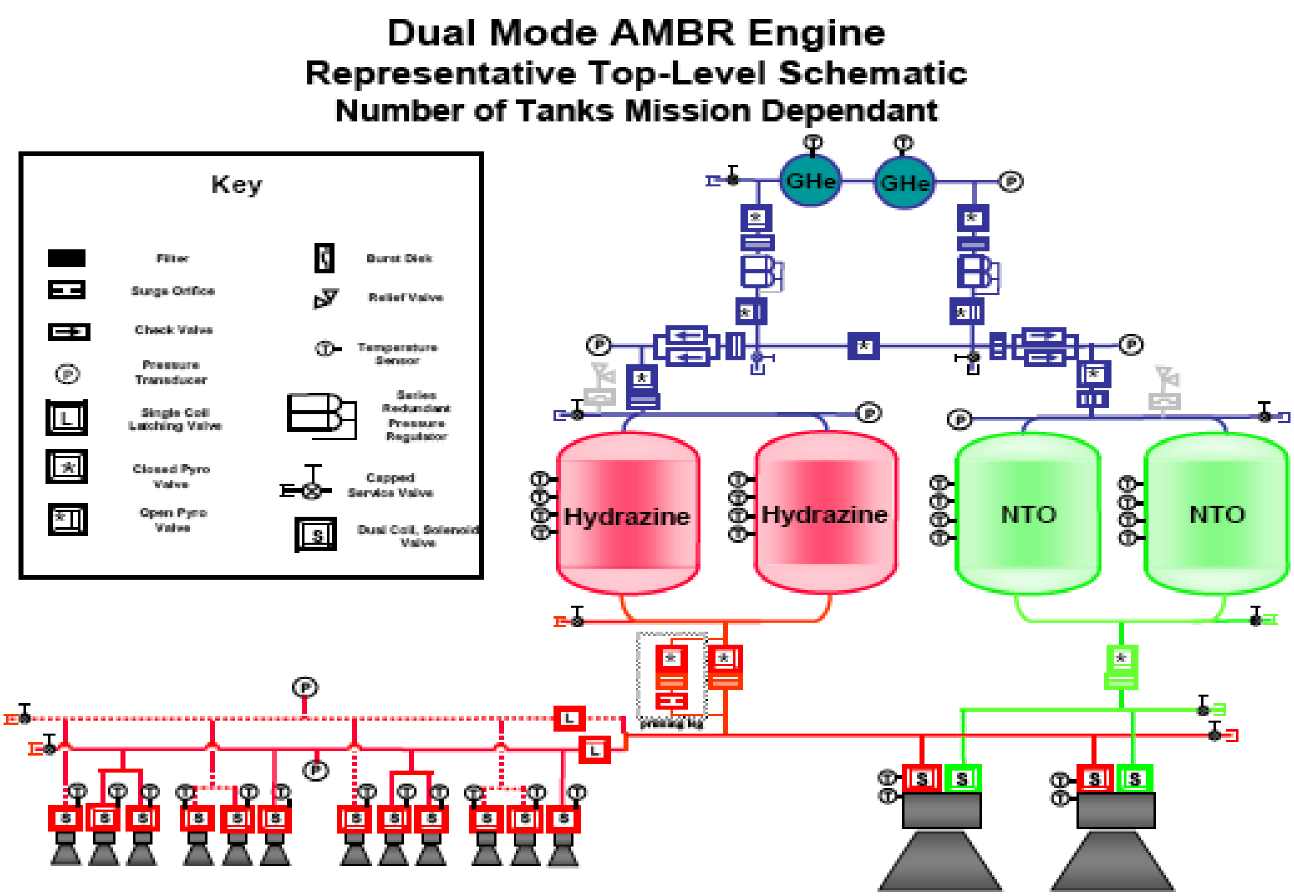

Figure 3. Example of AMBR Propulsion System

This representative AMBR propulsion system is single-fault-tolerant based on flight proven HiPAT™ design and can use a significant amount of the HiPAT™ heritage hardware. Most of the system hardware is at Technology Readiness Level (TRL) of 9 (flight proven through successful mission operations). The component masses (e.g., valves, regulators, filters, etc.) are based on those onboard flight proven spacecraft like the Mercury Messenger and Space Shuttle. A ten percent design contingency is used for the hardware.

1.3 Subsystem Summaries

For the representative AMBR system shown above, the pressure vessel characteristics are:

- Factors of Safety:
  - Propellant Tanks – 1.5
  - Pressurant Tanks – 1.5
- Materials:
  - Propellant Tanks – Ti (6Al-4V)
  - Pressurant Tanks – COPV

- Operating Pressures
  - Propellant Tanks – 2.6 MPa (400 psia)
  - Pressurant Tanks – 31 MPa (4,500 psia) down to 5.5 MPa (800 psia)
- Anti-slosh/propellant management device ~10 percent Propellant-tank shell mass
- Propellant tank ullage – 5 percent (regulated)
- Propellant residual – 1 percent
- Helium pressurant – sized for isothermal blowdown

Based on an AMBR propellant inlet pressure of 275 psia (an interim test result), the tank pressure is set at 275 psia and the tank is designed with a safety factor of 1.5. Tank material is fixed as titanium (6Al-4V), ullage volume at five percent and a surface tension propellant management device (PMD) is assumed to add ten percent to tank weight with one percent of the initial propellant load unusable.

Table 2 shows a component list for the representative AMBR engine system. It contains mass estimates for a Europa orbiter mission.

The assumption is made that the spacecraft propellant requirements will determine the tank size. This assumption

Table 2: Representative Component List for an AMBR Engine System (Europa Orbiter Mission)

	Europa Lander	Unit Mass		Total Mass		Comments:
Quantity	Propulsion System Components	(kg)	(lbm)	(kg)	(lbm)	
2	Pressurant Tank (COPV)	10.5	23.1	21.0	46.2	Calculated Hardw
3	Fill and Drain Valve, Hign Press He	0.1	0.2	0.3	0.7	Messenger Hardw
6	Filter, He	0.11	0.2	0.7	1.5	Messenger Hardw
7	Pyro Valve, Pressurant	0.2	0.4	1.4	3.1	Messenger Hardw
2	Pressure Regulator	2.31	5.1	4.6	10.2	STS OMS
1	High Pressure Transducer	0.23	0.5	0.2	0.5	Messenger Hardw
4	Check Valves	1.36	3.0	5.4	12.0	STS OMS
4	Transducer, Low pressure	0.23	0.5	0.9	2.0	Messenger Hardw
0	Burst Disk	0.1	0.2	0.0	0.0	STS OMS
0	Relief Valve	2.31	5.1	0.0	0.0	STS OMS
4	Ground Checkout Hand Valve	0.07	0.2	0.3	0.6	Messenger Hardw
2	Propellant Tanks, Fuel (w/ PMD)	18.1	39.8	36.2	79.6	Calculated Hardw
1	Propellant Tanks, Oxidizer (w/ PMD)	25	55.0	25.0	55.0	Calculated Hardw
3	Pyro Valve, Propellant	0.2	0.4	0.6	1.3	Messenger Hardw
2	ISO Valve, Propellant, RCS	0.65	1.4	1.3	2.9	Messenger Hardw
6	Fill and Drain Valve, Propellant	0.15	0.3	0.9	2.0	Messenger Hardw
3	Filter, Propellant	0.29	0.6	0.9	1.9	Messenger Hardw
6	Transducer, Low pressure	0.23	0.5	1.4	3.0	Messenger Hardw
12	RCS Thruster (22 N, 5 lbf thrust)	0.65	1.4	7.8	17.2	Aerojet MR-106E
2	AMBR Thruster (91 N, 200 lbf thrust)	5.5	12.0	10.9	24.0	
	Miscellaneous Hardware	10%		12.0	26.4	
	Design Contingency	10%		13.2	29.0	
	Total Dry Weight			145.0	318.9	
	Propellant: Usable			1111.7	2445.8	
	Residuals			11.1	24.5	
	Pressurant: Helium			1.7	3.8	
	TOTAL PROPULSION SYSTEM			**1269.6**	**2793.0**	

The assumption is made that the spacecraft propellant requirements will determine the tank size. This assumption may not be practical as it is often preferable to select an existing flight proven tank, even though the size may not be optimal, to avoid the developmental cost for a new tank.

Pressurant tanks are the next largest mass element of a propulsion system. Propellants are pressure fed from the tanks to the engine, so a composite-overwrapped helium pressure vessel was selected with size calculated assuming adiabatic blowdown of gas initially at 4500 psia down to a minimum regulator inlet limit of 800 psia.

For MSFC's system model, component masses are based on the mass of existing hardware that is flight proven in the space environment (TRL 9) in spacecraft like the Mercury Messenger or Space Shuttle. Additionally, ten percent design contingency is applied to ensure that system mass is not under-estimated.

2. PERFORMANCE SUMMARY

2.1 Performance Characteristics

The resultant AMBR engine peak performance characteristics are thrust level of 140 lbf and I_{sp} of 333 seconds. A nominal operating condition for AMBR, most likely at a slightly reduced thrust and performance, would need to be defined based on the level of margin required for the application. A notional AMBR engine operating box based on performance data is shown in Figure 4. No combustion instability was observed in the hotfire testing using a substitute copper combustion chamber.

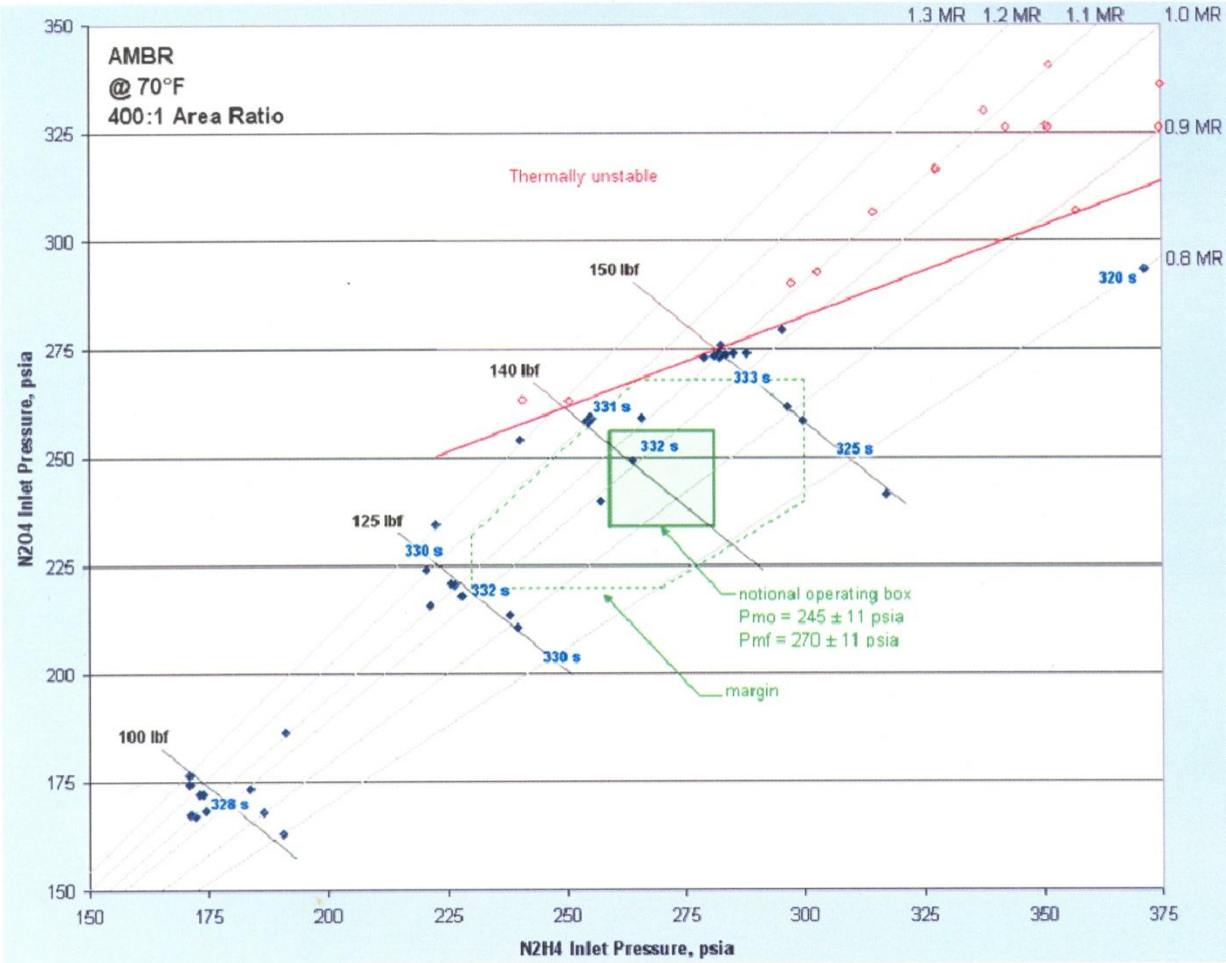

Figure 4: Notional operating box for AMBR engine

At the outset of the AMBR developmental effort, NASA MSFC and NASA JPL conducted mission level and system level studies to extrapolate improved engine performance into spacecraft requirements and performance. JPL chose four reference missions for this analysis based on scientific interest, current launch vehicle capability, and trends in spacecraft size (at the time when the analysis was performed 2006-2007 using the 200 lbf thrust baseline). Table 3 shows a summary of the results of the analyses. The Delft University of Technology provided the propulsion requirements for a GEO-sat, extrapolated for 15-year service life. [1]

Table 3: Summary of the Reference Missions

Mission	ACS ΔV	Axial ΔV, m/s	Launch Mass, kg	Deployed/ shed mass, kg
GTO-to-GEO	1,170 m/s	1,830	4,800	0
Europa Orbiter	23.4 kg	2,215	2,170	0
Mars Orbiter	20.0 kg	2,064	2,250	0
Titan-Enceladus Orbiter	50.0 kg	2,368	6,633	1,298, 59.2, & 345

The performance analysis assumes a dual-mode propulsion system. In a dual mode system, Attitude Control System (ACS) thrusters share the hydrazine monopropellant with the main engine which also uses the hydrazine from the same supply system as fuel for combustion with an oxidizer.

Appendix A describes the method used to derive the mission information.

Table 4 shows a summary of the propellant mass estimates calculated for reference missions at various main engine I_{sp} values. The baseline I_{sp} is 320 seconds for the GEO missions and 325 seconds for the planetary missions. AMBR thruster is given the original target I_{sp} of 335 seconds. The analysis assumes an ACS I_{sp} of 230 seconds for monopropellant hydrazine.

Table 4: Propellant Estimates for the Four (4) Reference Missions

Mission	Total Propellant Load (kg) by Main Engine I_{sp}				
	320 sec	325 sec	330 sec	332.5 sec	335 sec
GTO-to-GEO	3,204	3,189	3,176	3,170	3,163
Europa Orbiter	N/A	1,131	1,120	1,116	1,109
Mars Orbiter	N/A	1,320	1,307	1,300	1,293
Titan-Enceladus Orbiter	N/A	2,969	2,942	2,928	2,914

These reference missions show the mass performance and benefits of the AMBR engine.

2.2 Benefits Over SOA

The baseline engine for AMBR's development is HiPAT™. The latter is currently the highest performing biprop engine in the 100-lbf thrust range with a specific impulse I_{sp} of 328 seconds. In contrast, the improved AMBR engine yields up to 140-lbf thrust, and its specific impulse is 333 seconds. Because of the increased specific impulse and thrust, AMBR would clearly offer mass benefit to missions.

AMBR's increased thrust at 140 lbf enables better Thrust Vector Control (TVC). For example, a single engine is preferred for spacecraft with 140 lbf thrust operating for transit or orbit insertion. Compared to multiple engines supplying the same thrust, a single engine simplifies the gimbals and thrust vector control.

Higher thrust level also provides options for descent and ascent in terms of the capability to carry a heavier load or a spacecraft design using fewer engines.

AMBR's utilization as a dual-mode engine, allows integration with the spacecraft RCS and ACS, using the same propellant, and simplifying the propulsion system design and operation.

2.3 Potential Application to Candidate Science Missions

The In-Space Propulsion Technology Project Office routinely performs high level assessments of the AMBR technology's applicability towards candidate science missions. The applicability is considered to be high.

3. DEVELOPMENT SUMMARY (UNDER ISPT)

3.1 Status

Initiated in year 2006, the AMBR effort has so far completed:

1. Baseline effort: by hot-firing a developmental HiPAT™ engine, the thermal, propellant flow and pressure dynamic information were successfully collected for use in the AMBR thruster design.
2. Injector risk mitigation: Based on the C* value obtained using a copper chamber firing for very short durations, the injector design was verified capable of achieving the developmental goal of 335 seconds I_{sp} at that time.
3. The AMBR engine completed the fabrication and the preliminary performance envelope hot fire testing in October of 2008 which demonstrated 150 lbf thrust and 333.5 seconds I_{sp}.
4. The AMBR engine completed the vibration and shock testing in January 2009.
5. Additional envelope testing was done in February and the long-duration hot fires completed in June 2009, with repeatable 140 lbf thrust and 333 seconds I_{sp}.

3.2 Key Activities (Summary)

Beginning in the latter part of year 2006, AMBR thruster development progresses via a number of stages:

- Stage 1: Baseline Hotfire Test: hotfire developmental HiPAT™ engine to collect thermal and dynamic information for use in designing the prototype. **Completed**
- Stage 2: Injector Design Verification/Risk Mitigation: verify injector design using a copper chamber; found design highly successful for meeting program goal. **Completed**
- Stage 3: Fabricate and test the AMBR prototype thruster: components needing fabrication include the complete injector assembly, combustion chamber, nozzle, and nozzle extension.
  Completed
- Stage 4: Perform shock and vibration environmental testing.
  Completed
- Stage 5: Perform post-environmental performance testing.
  Completed
- Stage 6: Perform engine inspection and analysis followed by additional performance and longer-duration testing.
  Completed
- Stage 7: Produce final drawings, design models (thermal and structural), work instructions and parts lists
  Completed

4. DISCOVERY MISSION IMPLEMENTATION

4.1 Subsystem Selection

Information needed for selecting subsystems is given throughout this document. Section 1.2 System Summary describes the AMBR system details which are the basis for subsystem selection.

4.2 Mission Success

The AMBR propulsion system development summarized here was done with rigor and depth of considerations required for a high performance bi-propellant system suitable for NASA planetary missions. The development is accomplished via a multi-year, multi-partner (NASA Centers, JPL, Aerojet Corporation, etc,).

The AMBR technology is an improvement upon the existing HiPAT™ engine, a member of the Aerojet Corporation's R-4D Family of thrusters. The R-4D family of thrusters has the following heritage: >1000 engines delivered, >650 flown, 100 percent success rate. [2]

5. CONTACT INFORMATION

Please direct all inquiries and requests related to the AMBR engine to the following individual:

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APPENDICES

A. METHOD FOR DERIVING THE REFERENCE MISSION INFORMATION:

For each mission, the mass of the spacecraft at launch is estimated based on the expected launch vehicle capability and the terminal velocity which the launch vehicle is obligated to impart. The spacecraft trajectory is planned, in some cases taking advantage of planetary momentum exchange to modify the spacecraft velocity. Main engine burns are an essential part of trajectory planning to keep the spacecraft on course. In one case, the scientific requirements of the mission require deployment of spacecraft elements such as a heat shield or independent landing craft, requiring accounting for the mass decrements. Demands placed on the attitude control system are modeled based on historical data, acceptable limits of spacecraft pointing and statistical distributions of spacecraft attitude perturbations due to internal and external influences. The calculated propellant load is increased by one percent to account for the inability of propellant tanks to completely discharge their contents. Finally, because of the uncertainties inherent in engineering, a five percent margin is added to the propellant load.

Once the accounting is in place for mass and velocity changes, assumptions are made regarding the efficiency of the propulsion system elements. These assumptions are based on a database of past engine performance or in this case on the goals for improved main engine performance. The propellant mass required to execute the velocity changes required by trajectory planning and ACS analysis are determined by means of the rocket equation or similar calculation.