Multi-Mission Suitability of the NASA Ames Modular Common Bus

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ABSTRACT

The obvious advantages of small spacecraft - their lower cost structure and the rapid development schedule - have enabled a large number of missions in the past. However, most of these missions have been focused on Earth observation from low Earth orbits. In 2006, the Small Spacecraft Division at the NASA Ames Research Center began the development of the Modular Common Bus, a spacecraft capable of delivering scientifically and technically useful payloads to a variety of destinations within 0.1 AU around the Earth. The core technologies used in the Common Bus design are a composite structure with body-mounted solar cells, an integrated avionics unit, and a high performance bipropellant propulsion system. Due to its modular approach, the Common Bus can be adapted to fit specific mission needs while still using a standardized and qualified set of components. Additionally a number of low cost launch vehicles are supported, resulting in overall mission costs of around $150M including the launch vehicle but excluding the science payloads. This significant reduction in cost and the shorter development time would enable NASA to conduct more frequent exploration missions within its budget and timeframe constraints, compared to the status quo.

In this paper the suitability of the Common Spacecraft Bus for four different exploration scenarios is analyzed. These scenarios include a lunar orbiter, a lunar lander, a mission to a Sun-Earth Libration Point, and a rendezvous mission to a Near Earth Object. For each scenario, a preliminary design reference mission is developed and key design parameters for the spacecraft are determined.

INTRODUCTION

In the last two decades small spacecraft with a total mass of up to 500 kg have proven to be a valuable possibility for different space applications, due to their lower costs and shorter development times. These applications include Earth and space sciences, where the lower cost results in a larger number of flight opportunities for science instruments [1, 2]. Even complex exploration missions, like planetary landers, have been studied extensively for the last ten years [3, 4, 5]. However the costs for exploration missions using small spacecrafts are still substantial due to their unique design, which might not always be necessary since modular spacecraft buses that can be adapted for a variety of missions could be used instead [6, 7].

In June 2006, the Small Spacecraft Office at the NASA Ames Research Center started to work on the design of the Modular Common Bus (MCB), a small spacecraft bus specifically designed for exploration missions beyond LEO [8]. For NASA, small spacecraft can greatly facilitate achievement of the agency's vision by undertaking critical precursor missions. The design goal for the Common Bus was to develop a spacecraft that is capable of delivering scientifically and technically useful payloads to a variety of orbits or even the lunar surface. Due to its modular approach, it can be adapted to fit specific mission needs while still using a standardized and qualified set of components. This will significantly reduce the costs and development times for new missions. This paper presents a short overview of the

Common Bus technical design and four design reference missions that demonstrate the suitability of the design which only requires minor modifications for each different mission.

THE MODULAR COMMON BUS

The design process for the NASA Ames Modular Common Bus was driven by the desire to develop a baseline spacecraft with sufficient capabilities for a large number of exploration missions while still reducing the costs compared to the traditional spacecraft design approach. This was achieved by avoiding unnecessary complexity, the use of integrated systems, and maximizing the use of components with flight heritage. Furthermore redundancy is only applied where necessary for a specific mission. An overview of all major modules and possible combinations for orbiter or lander missions is shown in Figure 1.

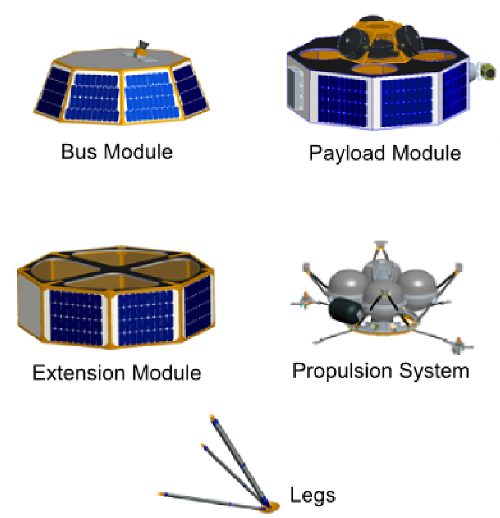

Figure 1: Major modules of the NASA Ames Modular Common Bus spacecraft.

All spacecraft subsystems except the propulsion subsystem are part of the uppermost, or "bus" module. The side walls are slanted in order to allow power generation of a wide range of sun orientations, which is especially important for planetary lander missions. To avoid the complexity and cost of a deployable mechanism, only body-mounted 28.5% BOL triple junction solar cells are used on all modules. The avionics unit integrates the command and data handling system and the electrical power system into a single box, which significantly improves the compactness and reduces the amount of harness required. Next to the avionics box are the transmitter, receiver, and power box for the S-Band communication system, which uses two evolved broad beam antennas mounted on the top and bottom of the spacecraft. The guidance, navigation, and control system is also part of the bus module and very modular in itself. Possible sensors include up to twelve coarse sun sensors, a star tracker with two camera heads, and an inertial measurement unit. All selected sensors have spaceflight heritage. Up to four reaction wheels can then be used to adjust the spacecraft attitude.

The payload module on the other hand is reserved solely to accommodate science instruments which can be mounted within the module or on the outer surfaces. The remaining surface area is filled with solar cells and additional radiator surface if required.

The bottom extension module includes the high performance bipropellant propulsion system, which uses Monomethylhydrazine (MMH) and Mixed Oxides of Nitrogen (MON) as propellants. This system enables the Modular Common Bus to be also used for planetary lander missions by using components with a high thrust to weight ratio [9]. While a high thrust Divert Attitude Control System (DACS) thruster is used for orbit correction maneuvers and landing, six smaller thrusters in two triplet configurations on opposite sides of the extension module are used for attitude control. Each thruster triplet is composed of a pair of thrusters in a "bow-tie" assembly and third thruster oriented downward, in the same direction as the main thruster. The two "bow-tie" pairs are the minimum sufficient for attitude control, while the two downward-pointing thrusters can provide the capability for fine velocity control, nutation control of a spinning cruise stage, and a limited degree of redundancy, depending on the mission requirements. If the delta-v capacity of the 27 kg of propellant in the four tanks of the bipropellant system is insufficient for a particular mission, a cruise stage with a solid rocket motor (SRM) can be mounted beneath the extension module to provide additional delta-v. For planetary landers, four lightweight legs can be added as well.

The number of modules is to some extent determined by the mission parameters. A "minimal" MCB suitable for a very small lander or other high delta-v mission would consist of only a bus module and an extension module containing the propulsion system, with a very small payload installed in the bus module. On the other hand a "maximal" MCB could consists of two payload modules, or a double extension module to carry a bulky propulsion system.

The Modular Common Bus was designed to be used with dedicated small lift launch vehicles like the Minotaur or Falcon. By doing this, the launch date and insertion are more flexible and can be optimized for a specific mission scenario. This optimized use of the launch vehicle and the simplified launch operations also reduce the total mission cost. Furthermore being the secondary payload on a launch vehicle usually results in a number of constraints especially regarding the use of propulsion systems. As currently designed, the MCB structure is slightly too large to fit in a Pegasus payload fairing. However, the structure has the capability to be shrunk to the degree required with minimal loss in structural mass efficiency.

MISSION 1: LUNAR LANDER

Probably the most challenging exploration mission within the scope of low cost small spacecraft is a planetary lander. Much of the initial MCB design was done using a planetary lander as a reference mission, in the belief that a vehicle which could meet the demands of this mission could be more easily adapted to others than the reverse.

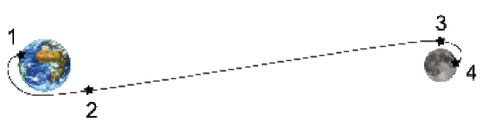

Figure 2: Mission scenario for lunar lander, showing the launch (1) with a C_3 of -1.89 km^2/s^2, the midcourse Correction (2) with a delta-v of 50 m/s, the Lunar Insertion (3) with 2,350 m/s, and the descent (4) with 450 m/s.

A typical lunar lander mission, as shown in Figure 2, would start with the injection of the spacecraft into a lunar transfer trajectory using a dedicated small lift launch vehicle like the Minotaur V. Within 24 hours after insertion a midcourse correction maneuver (MCC), requiring around 50 m/s, is performed to correct for insertion errors. In contrast to other design studies for lunar landers [10, 11, 12] where the spacecraft first enters a low lunar parking orbit before starting its descent, the Ames spacecraft is directly inserted into the descent trajectory. The overall delta-v for insertion, descent, and landing is not affected by this change and remains around 2,800m/s, but the change allows shifting delta-v from the descent phase to the insertion phase of the maneuver. This makes it possible to increase the mass efficiency of the mission by using a powerful solid rocket motor (SRM) for the 2,120m/s insertion burn and then after jettisoning the SRM, the spacecraft uses its own liquid propulsion system for the remaining 705m/s of the descent and landing phase. During the approach a radar altimeter and a Digital Scene-Matching Area Correlator (DSMAC) is used, which compares real-time pictures taken of the surface with an onboard camera with a sufficiently detailed reference map of the landing area stored in the onboard computer [13]. To soften the actual landing, the four legs have crushable sections that reduce the final relative velocity to the surface to 4m/s. The mass distribution of the spacecraft and the orientation of the legs allow a landing on uneven terrain with a slope of up to 15 degrees without tipping over. After landing the 10kg of science payload can be operated for around two weeks using the onboard battery and the solar cells.

The overall configuration of the MCB for such a mission consists of the bus module, the extension module with the legs and the propulsion system, and the solid rocket motor for the insertion burn. While the radar altimeter and DSMAC are added to the GN&C system, no reaction wheels are required for a lunar lander mission. Currently, some of the prototype control software for a lunar lander is being developed and tested at the Hover Test Facility at NASA Ames [14].

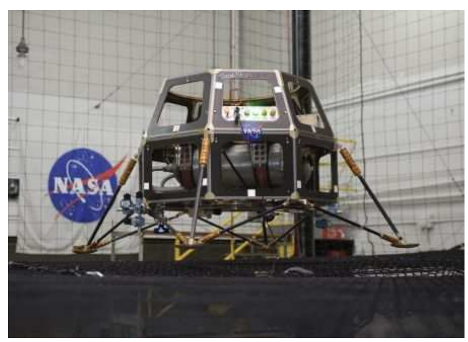

Figure 3: The Hover Test Vehicle at the NASA Ames Research Center.

The overall costs of such a mission are estimated to be around $150M including the launch vehicle, with a required development time of around 30 months.

MISSION 2: LUNAR ORBITER

A reference mission for a lunar orbiter to be used in a low lunar orbit, shown in Figure 4, is very similar to the lunar lander mission described above.

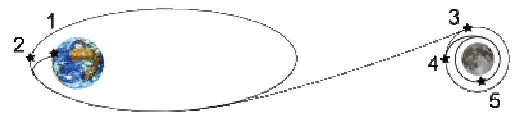

Figure 4: Mission scenario for lunar orbiter, showing the launch into the phasing loops (1) with a C3 of -2.6 km2/s2, the transfer trajectory insertion (2) with a delta-v of 50 m/s that includes 20 m/s for the midcourse correction, the insertion into the lunar checkout orbit (3) with 800 m/s, the Hohmann transfer from the checkout into the science orbit (4) with 80 m/s, and the orbit maintenance and reaction wheel desaturation maneuvers (5) on the science orbit.

While a direct injection into a transfer trajectory would be possible as well, it is more likely that the spacecraft will be inserted into phasing loops around the Earth requiring a C_3 of around -2.6 km^2/s^2. The following 21 day mission phase can then be used to checkout all necessary spacecraft functions before the spacecraft uses its onboard liquid propulsion system to perform the transfer trajectory insertion by increasing the apogee altitude to lunar distance. This approach requires an additional 30m/s and is therefore not feasible for the delta-v constrained lunar lander, but offers operational flexibility for the less constrained lunar orbiter. While the midcourse correction maneuver only requires 20m/s due to the more exact insertion, the 800m/s lunar orbit insertion burn is performed using a solid rocket motor again. Depending on the mission requirements, like calibration of science instruments on a higher altitude orbit, the spacecraft can either be inserted directly into its science orbit or into a higher altitude checkout orbit. These higher orbits require less orbit maintenance and have shorter eclipses, which allow spacecraft and instrument checkout in a more stable environment. After the checkout, the liquid propulsion system would be used to adjust the orbit altitude after the checkout phase. A total delta-v amount of 250m/s is available after the lunar orbit insertion, which should be sufficient for around three months of operation in an equatorial orbit or eighteen months of operation in a polar lunar orbit [15].

The lunar orbiter uses the baseline MCB configuration described earlier, including the payload module which can accommodate up to 50kg of payload. To enable accurate 3-axis stabilized pointing of the payload four reaction wheels are used. Using four instead of the minimum required three wheels decreases the momentum that each wheel has to store and therefore increases flexibility in scheduling of the reaction wheel desaturation maneuvers. Furthermore a medium gain antenna can be added to increase the amount of data that can be downloaded from the spacecraft. The required transmission time as well as the pointing accuracy also determines the amount of power available for the science payloads. The overall costs and required development time are similar compared to the lunar lander described earlier.

Figure 5: Artists impression of the Lunar Atmosphere and Dust Environment Explorer (LADEE).

A very similar mission, called the Lunar Atmosphere and Dust Environment Explorer (LADEE) is currently pursued by the NASA Ames Research Center with support from the NASA Goddard Space Flight Center [16]. LADEE carries three science instruments and a laser communication experiment. It is currently scheduled for launch in early 2012.

MISSION 3: SUN EARTH LIBRATION POINT

The Sun-Earth Libration point 1 is situated between the Sun and the Earth with a distance of around 0.1 AU from the Earth. A variety of spacecraft have used this point in the last decades because of its unique opportunities for heliospheric research. Since the libration point itself is unstable, a spacecraft needs to be placed in an orbit around it [17]. Depending on the orbit parameters, these are called halo or lissajous orbits. The SEL1 reference mission proposed here and shown in Figure 6 uses a large amplitude halo orbit, similar to the one used by ISEE-3 and SOHO [18]. These orbits require very low orbit maintenance and are therefore favorable for low cost small spacecraft missions.

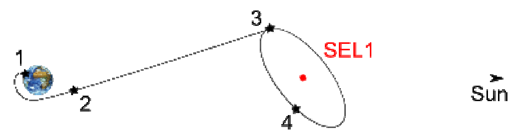

Figure 6: Mission scenario for Sun Earth Libration Point Mission, showing the launch into the transfer trajectory (1) with a C_3 of -0.6 km^2/s^2, the midcourse correction (2) with a delta-v of 50 m/s, the halo orbit insertion (3) with 50 m/s, and the orbit maintenance and attitude control maneuvers on the halo orbit (4) with 130 m/s.

Similar to the reference missions described above and all previous SEL missions, a direct injection into the transfer orbit using a dedicated launch vehicle is used. Potentially the spacecraft could also be placed in a Low Earth Orbit using even smaller launch vehicles, and then use an additional cruise stage with a solid rocket motor for injection into the transfer orbit. In case of a direct injection, no solid rocket motor is required at all. After the direct injection with a C_3 of -0.6km^2/s^2, the spacecraft needs to perform a midcourse correction requiring a delta-v of around 50m/s again. After around 120 days the halo orbit insertion is performed, the delta-v of which is a function of the transfer time and the orbit amplitudes. Around 50 m/s can be used as an estimate for a mission similar to ISEE-3 and SOHO [17, 20]. Due to the large amplitude of the halo orbit, only 10m/s are required for orbit maintenance per year, while the attitude control delta-v amounts to nearly 100m/s [11, 14, 15]. The resulting total mission delta-v including a large delta-v margin of 100 m/s is similar to the total delta-v for ISEE-3 and SOHO, which had 430m/s and 318m/s respectively [17, 21].

The Modular Common Bus only requires a limited amount of modification to accommodate this SEL1 mission. The three layer configuration is ideally suited for spin-stabilized operation similar to ISEE-3 and SOHO. This would mean that no reaction wheels are required, resulting in 5.6kg of mass savings compared with nominal design. Based on the experience of previous missions, the coarse sun sensors (CSS) used on the common bus should be upgraded to more accurate two-axis ones which are about 1.2kg heavier. Since the spacecraft will be constantly in sunlight at SEL1, a smaller 15Ah battery sufficient for launch and eclipses during transfer can be used, resulting in an additional 2.6kg of mass savings. The thermal subsystem should be able to handle the slightly higher solar constant without major modifications. Depending on the science payload, additional radiator surface might be required which can be achieved by replacing parts of the MLI on the bottom with silvered Teflon. A major modification is necessary on the communication subsystem due to the large distance between the spacecraft and the Earth, as well as the chosen spinning attitude. The similar ISEE-3 spacecraft used a medium gain antenna, with a pancake pattern, mounted on top of the spacecraft once the spacecraft had reached its final orbit [11], while the nominal low gain antennas still provided sufficient data rate for the cruise phase and for backup operations. The ground segment for the SEL1 mission requires using the 36m dishes of the Deep Space Network, while the previously discussed missions can use the smaller 18m dishes of the Near Earth Network.

Overall, a spacecraft based on the Modular Common Bus with the modifications described above and using a design reference mission similar to the one above, would be capable of delivering 50kg of payload to the Sun Earth Libration point for costs similar to the one of a lunar orbiter.

MISSION 4: NEAR EARTH OBJECT

The Mission Design Center at NASA Ames also explored the possibility of using the Modular Common Bus for a mission to the Near Earth Object (NEO) Apophis [22]. The spacecraft would be launched into a heliocentric trajectory with a C_3 of 8.3 km^2/s^2 using a medium lift launch vehicle like the Falcon 9 or Taurus 2. Similar to the lunar lander and orbiter concepts described above, a solid rocket motor would provide 80 to 90% of the delta-v required for the rendezvous burn, while the liquid propulsion system would be used for the remaining delta-v as well as orbit maintenance and attitude control. While the power, avionics, thermal, guidance, navigation, and control system are completely similar to the nominal Modular Common Bus, the communication subsystem needs to be modified to be suitable for the long distance between Apophis and the Earth at the time of rendezvous. This is achieved by adding a fixed parabolic reflecting High Gain antenna operating in X-Band, and using the 34m Beam Waveguide dishes of the Deep Space Network for the downlink, while still using the S-Band system for uplink of commands. The total payload capacity would be around 10kg with total costs of around $150M, again including the launch vehicle and excluding the science instruments.

CONCLUSION

As shown in the previous sections, a spacecraft based on the NASA Ames Modular Common Bus is suitable for a variety of exploration missions while requiring only a limited amount of modifications. This greatly reduces the development time as well as the number of personnel required, which in turn reduces the total mission cost. An overview of the estimated performance for four selected exploration missions is given in Table 1.

Table 1: Modular Common Bus configurations for different exploration missions. The total wet mass estimates for the Sun-Earth Libration Point mission and the Near Earth Object Mission contain additional mass margins, since these missions have not been analyzed in the same

Mission	*Lunar Lander*	*Lunar Orbiter*	*Sun Earth Libration Point*	*Near Earth Object*
Launch Vehicle and Trajectory				
Launch Vehicle	Small Lift	Small Lift	Small Lift	Medium Lift
C_3	-1.89 km^2/s^2	-2.6 km^2/s^2	0.6 km^2/s^2	8.3 km^2/s^2
Trajectory	Direct Transfer	Phasing Loops	Direct Transfer	Direct Transfer
Cruise Stage (SRM)	Yes	Yes	None	Yes
Spacecraft Design				
Modules	Top, Extension with Propulsion, Legs	Top, Payload, Extension with Propulsion	Top, Payload, Extension with Propulsion	Top, Payload, Extension with Propulsion
Subsystems modified compared to Baseline	Guidance, Navigation and Control	Communication	Guidance, Navigation and Control, Communication	Communication
Total Wet Mass	101 kg	191 kg	201 kg	175 kg
Payload Mass	10 kg	40 kg	40 kg	10 kg
Total Delta-V	2,874 m/s	1,142 m/s	380 m/s	2,681 m/s
Cost and Schedule				
Cost	$150M	$150M	$150M	$190M
Development Time	30 months	36 months	36 months	36 months

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