Attitude and Orbit Control System of CubeSat Lunar Lander OMOTENASHI

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Abstract

OMOTENASHI is a 6U size, 14kg CubeSat lunar lander which will be launched by SLS EM-1 rocket. After deployment from the rocket, it is put into a lunar impact orbit by gas jet propulsion systems, and its orbital velocity is cancelled by a solid rocket motor just before its lunar impact. Its Attitude and Orbit Control System (AOCS) has the functions of Sun-pointing, three-axis pointing, orbital maneuver, angular momentum management, spin-up, etc., and those functions should be realized by ultra-small instruments. In this paper, the specification required for the onboard AOCS and the design results which meet the requirements are reported.

超小型月着陸機 OMOTENASHI の姿勢軌道制御系

OMOTENASHI は、6U サイズ、14kg の超小型月面着陸機であり、SLS EM-1 ロケットにて打ち上げられる予定である。ロケットから分離後、ガスジェット推進系を用いて月衝突軌道に投入され、月面衝突直前に固体ロケットモータにより軌道速度がキャンセルされる。姿勢軌道制御系（AOCS）は、太陽指向制御、３軸姿勢制御、軌道制御、角運動量制御、スピンアップ制御などの機能を有しており、これらの機能を超小型の機器により実現している。本稿では、搭載 AOCS へ要求される仕様と、それを満たす設計結果について報告する。

1. Introduction

OMOTENASHI (Outstanding MOon exploration TEchnologies demonstrated by NAno Semi-Hard Impactor) is a secondary payload for launch by NASA's SLS (Space Launch System) EM-1 (Exploration Mission-1) with the Orion crew vehicle in Dec., 2019 [1]. In August 2015, NASA invited international partners to participate and Japan Aerospace Exploration Agency (JAXA) responded to this invitation in October, 2015. In May, 2016, NASA finally selected two JAXA CubeSats, namely OMOTENASHI and EQUULEUS [2]. The main mission of OMOTENASHI is to demonstrate semi-hard landing to lunar surface by a CubeSat [3][4]. The technologies developed for OMOTENASHI enable wide-range distributed multiple observations by CubeSats. Moreover, in the near future, industry, academia, and even individuals will be able to easily participate in space exploration. OMOTENASHI will contribute very much to realize such a world.

Since it is too difficult to land a CubeSat softly on the lunar surface due to severe resource limitations, we have considered a semi-hard landing scheme. That is, the final deceleration for the landing is performed by a solid rocket motor as an open loop manner without any navigation sensors. To realize a spacecraft within 14 kg and of 6U CubeSat size, we had to develop some new technologies. For example, a propulsion system that would allow an entire 14-kg spacecraft to land on the lunar surface could not be realized within 6U size, so it is essential to minimize the landing mass. Current mass allocation of a surface probe, that is, the landing part, is 715 g. Considering the specific impulse (I_{sp}) and dry mass of the propulsion system, we use a solid rocket motor to decelerate from an orbital velocity of roughly 2.5 km/s. Moreover, even the ignitor of the solid rocket motor should separate just after ignition to reduce deceleration mass; a laser ignitor is introduced to realize this concept. Since the thrust of a solid rocket motor cannot be controlled after its fabrication, an error of a few tens of meters per second at impact on the lunar surface should be considered. To withstand a high-speed impact, shock absorption mechanisms are needed. We employ three kinds of technologies, an airbag, a crushable material, and epoxy filling.

This paper presents the mission sequence, the requirements for Attitude and Orbit Control System (AOCS) of the spacecraft, and the design results of the AOCS.

2. Mission sequence

During launch, each spacecraft will be housed inside an individual dispenser installed in Orion Stage Adapter (OSA) [5] and shall be powered off [6]. After leaving the dispenser, the OMOTENASHI spacecraft shown in Fig.1, which consists of an orbiting module (OM), a rocket motor (RM), and a surface probe (SP), will initiate an attitude acquisition sequence to make its body-mounted solar cells face toward the Sun. After a health check and tracking for orbit determination lasting roughly a day, an orbital maneuver to put the spacecraft into lunar impact orbit (DV1) will be performed by a cold-gas-jet system; the velocity increment at DV1 will be about 15 m/s. A trajectory correction maneuver (TCM) will be conducted if the DV1 execution error is large.

After precise orbit determination for a few days, several minutes before the lunar impact, the spacecraft will begin its landing preparation sequence, that is, an attitude change for the deceleration maneuver (DV2) and spinning up for the solid motor firing. DV2 will be conducted by the solid motor of the RM, and deceleration will be from 2.5 km/s. Soon after ignition of the solid motor, the OM will separate to reduce the deceleration mass. After DV2 is completed, the SP will land on the lunar surface, where it has been designed to survive for at least a few minutes. On the other hand, the OM will collide to the surface with 2.5 km/s, so it will be completely destroyed. Figure 2 shows the mission sequence in outline.

The landing trajectory of OMOTENASHI is not vertical but almost horizontal [7][8]. The flight path angle (FPA), defined as the angle between the orbital velocity (before DV2) and the local horizon at the landing point, is only a few degrees. After DV2, the velocity relative to the surface is expected to be zero at a certain altitude. Therefore, the impact velocity is determined by the DV2 velocity error and the terminal altitude, which results in a free-fall velocity. If the altitude is too high, the longer free fall time causes larger impact velocity. However, if the altitude is too low, the spacecraft might hit a hill or a mountain before DV2 has completed. Hence, altitude accuracy is an important parameter for a safe landing. Because the FPA is so small, the DV2 magnitude error does not affect the impact velocity directly, although it does affect the downrange error. The most significant cause of the residual vertical velocity is DV2 direction error, that is, attitude error of the spacecraft during DV2. The landing error analysis is summarized in Fig.6 and the analysis in detail is presented in section 5. The current target velocity of the impact is 50 m/s vertically and 100 m/s horizontally.

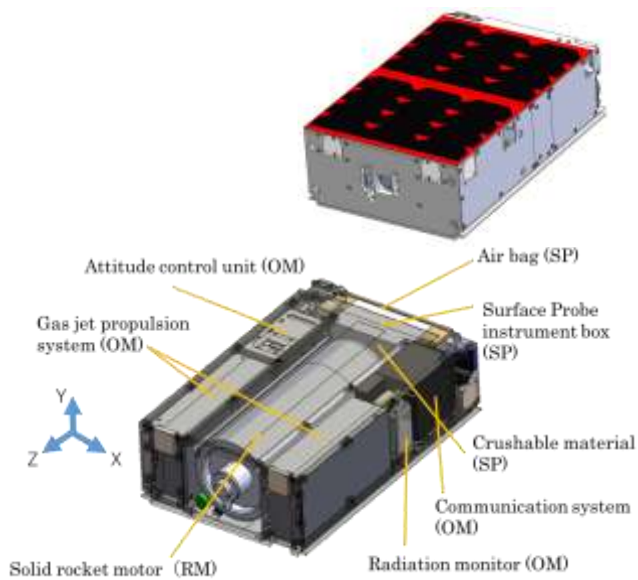

Fig. 1 Spacecraft configuration

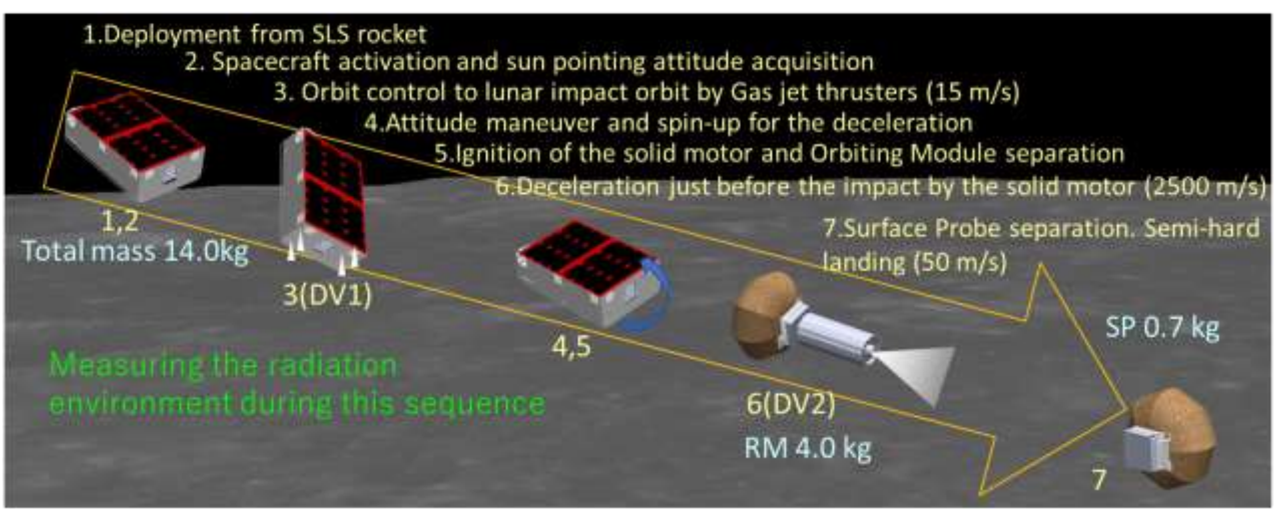

Fig.2 Mission sequence of OMOTENASHI

3. Requirements for AOCS

To realize the mission sequence, AOCS should have following functions,

(1) Sun pointing attitude control
  Initial autonomous sun acquisition
  Safe mode attitude control

(2) Three axis attitude control
  Nominal sun pointing considering communication antenna pattern
  Attitude maneuver between delta-V attitude and sun-pointing attitude

(3) Angular momentum control using gas-jets
  Separation disturbance from SLS rocket
  Attitude disturbance by solar radiation pressure

(4) DV1 and TCM using gas-jets to inject the proper lunar impact orbit
  Perform required velocity increment (15 m/s)
  Off-modulation/Attitude control while DV1

(5) Spin up for DV2 using gas-jets
  Spinning up around Z-axis up to 8 Hz (TBD)

(6) DV2 using a solid rocket motor to decelerate for landing (2500 m/s)
  Ignition of the solid rocket motor with 100 ms accuracy

4. Specification of AOCS

To meet the requirements, XACT attitude control unit of Blue Canyon Technologies (BCT) is used for fundamental three-axis control. It consists of a star tracker, four sun sensors, a three-axis IMU, three reaction wheels, and a control computer. It has sun-pointing, three-axis pointing, and attitude maneuver functions. Since it is developed for satellites operated in low earth orbits, however, it does not have actuators to manage the angular momentum outside Earth's magnetosphere. Therefore, the angular momentum should be controlled by the on board computer (OBC) software using gas-jet propulsion system, that is, two units of MiPS of Vacco Industries. The specifications of XACT and MiPS are listed in Table 1 and 2, respectively. Thrusters of MiPS are shown in Fig. 3.

One minute (TBD) after the separation from SLS rocket, XACT is powered on by OBC and the sun acquisition sequence starts automatically if the angular momentum is less than 0.01 Nms. The sun acquisition by XACT completed within 10 minutes in many numerical simulation cases. When the angular momentum exceeds the limit due to the separation disturbance, it should be reduced by MiPS. The maximum tip-off rate caused by the separation is 10 deg/s per each axis, which corresponds to 0.023, 0.031, 0.012 Nms for X, Y, Z axis, respectively. The angular momentum is monitored by OBC and OBC initiates momentum dumping control using gas-jets firings. From the safety requirements, MiPS should be powered on after one of sun sensors detects sun light to ensure that the spacecraft has separated from the dispenser. The initial attitude control sequence is summarized in Fig. 4.

After the sun acquisition has established, XACT keeps "Sun point" mode using a sun sensor and a IMU. During the first contact pass, "Fine Reference Point" commands are sent from a ground station and XACT starts three-axis-control using a star tracker and a IMU. The spacecraft basically continues three-axis-control till the spin-up for DV2.

For DV1, the spacecraft changes its attitude to the designated one using the XACT attitude maneuver function. DV1 is conducted by axial thrusters of MiPS while XACT is in Fine Reference Point mode. Attitude disturbance will be inevitable due to thruster unbalance. Therefore, off modulation of the axial thrusters and attitude control by the tangential thrusters are used.

Table 1 Specification of XACT

Pointing accuracy	±0.003 deg (1s) for STT cross axis ±0.007 deg (1s) for STT boresight axis
Data Interface	RS-422 115.2 kbps
Sensors	STT x 1 Sun sensor x 4 3-axis IMU Magnetometer x 5 (not used)
Actuators	Reaction wheel x 3
Mass	0.91 kg
Volume	100 x 100 x 50 mm (0.5U)

Table 2 Specification of MiPS

Required total impulse (for two units)	560 Ns DV1: 20 m/s = 280 Ns Attitude control: 50 Ns Spinning up: 40 Ns Margin: 190 Ns
Propellant	R236fa (Isp=40 sec)
Thrusters	25 mN x4 (see Fig. 3)
Tank	589 ml, Pressure 0.69 MPa
Mass	Dry 830g, propellant 740g
Volume	53 x 94 x 180 mm

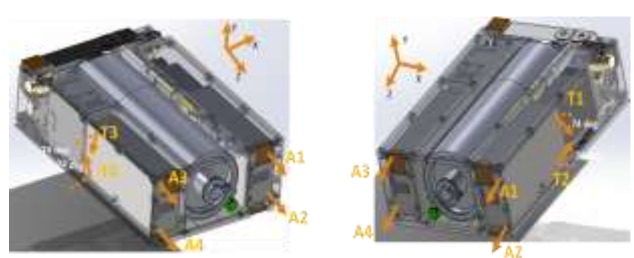

Fig. 3 Thrusters of MiPS

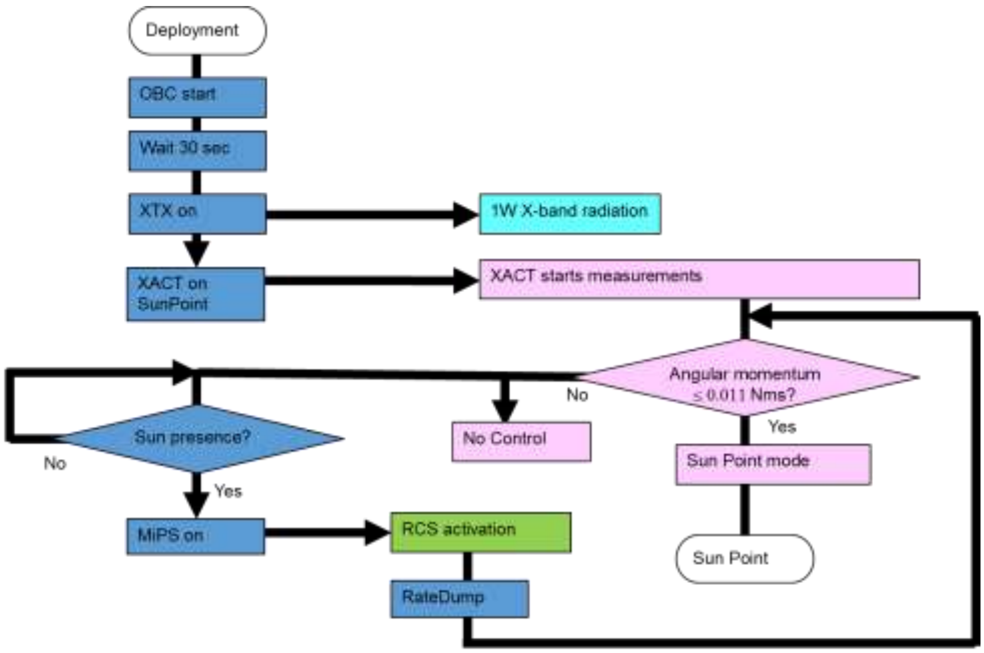

Fig.4 Initial attitude control sequence

Table 3 AOCS control mode

	OBC control	RCS	XACT				
			software	RW	STT	SAS	IMU
Stand by	STNBY		External			chk	chk
Rate dump	RateDump	use	External			chk	use
Sun acquisition	NCNT	use	Sun Point	use		use	use
Three axis	NCNT		Fine Reference Point	use	use	chk	use
RW unloading	UnLoad	use	Fine Reference Point	use	use	chk	use
Orbit Maneuver	OMV	use	Fine Reference Point	use	use	chk	use
Spin up/down	SPIN	use	External				
Ground test	TEST	use	External	use	use	use	use

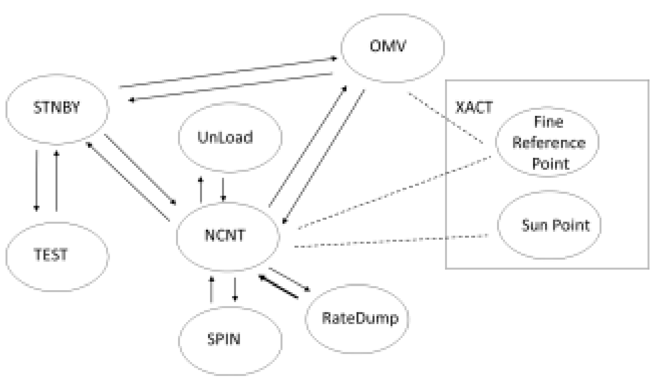

Fig. 5 State transition diagram of AOCS

Since the spacecraft has almost symmetrical shape and nominal attitude is sun pointing, solar radiation torque is negligibly small. Therefore, angular momentum management will be necessary only while the initial attitude acquisition and the orbit maneuvers.

DV2 requires spin stabilization. To withstand disturbance torque of the solid motor, 8 Hz (TBD) spin is needed as shown in the section 5. Since XACT does not work at such high speed spin, spin-up is conducted by MiPS with an open loop manner.

AOCS control modes are summarized with sensors and actuators usage in Table 3. Figure 5 shows state transition diagram of the control modes of OBC and XACT.

5. Landing error analysis

The success rate of OMOTENASHI semi-hard landing depends on the impact velocity to the lunar surface. There are many factors which determine the impact velocity as shown in Fig. 6. Though the precise orbit determination and control are indispensable, however, they can be discussed separately due to the small FPA (α). That is, the velocity errors of the orbit control are small enough compared with other error factors. Since the landing area is not restricted, along-track and cross-track position accuracy are not required. Only the absolute time synchronization of roughly 100 ms is needed to determine the DV2 execution timing. The altitude error of the trajectory affects the terminal altitude, namely free-fall-start altitude described in section 2. However, when we use current advanced technologies such as Delta Differential One-way Range (DDOR), the residual impact velocity of 50 m/s can be achieved. Therefore, the DV2 direction error will be the main cause of the residual vertical velocity. The sources of the direction error are,

(A) The separation disturbance from the OM
(B) The lateral thrust of the RM

(C) The centre of mass error
(D) The spin dynamics error

The spin rate of the spacecraft (n) affects those errors. Higher spin rate improves angular momentum stability and the error factors (A), (B), and (C) become small. The error (D) might become bigger if the nutation motion diverges. We currently consider that the error of (B) and (C) are dominant, though further analyses are needed.

We conducted some firing tests of solid rocket motors and measured their lateral thrusts. 5 N lateral force and 1 Nm disturbance torque were observed at most. If 1 mm difference between the centre of mass and the spin axis of the spacecraft exists, it corresponds to 0.5 Nm disturbance torque, when main thrust of the solid rocket motor is 500 N. Therefore, we assume the disturbance torque caused by factor (B) and (C) is 0.5 Nm constant. When an axisymmetric spacecraft is spinning at n, its attitude error A_p caused by disturbance torque N_d is described as follows. [9]

$$\lambda = \frac{(\mathrm{I}_x - \mathrm{I}_z)}{\mathrm{I}_x} n \tag{1}$$

$$A_\mathrm{p} = \frac{N_\mathrm{d}}{\lambda n \mathrm{I}_\mathrm{z}} \tag{2}$$

where I_x, I_y, and I_z are the moment of inertias around X, Y, and Z axis, respectively. When $\mathrm{I}_x = \mathrm{I}_y = 0.0547$ kgm^2, $\mathrm{I}_z = 0.0086$ kgm^2, $N_d = 0.5$ Nm, $n = 18.8$ rad/s (3 Hz), A_p becomes 11.2 deg. When $n = 50.3$ rad/s (8 Hz), A_p becomes 1.57 deg. When $n = 62.8$ rad/s (10 Hz), A_p becomes 1.00 deg.

Assuming that the attitude offset A_p is constant and the deceleration during the rocket motor burning is constant, the residual vertical velocity $V_{vertical}$ and the terminal altitude $L_{freefall}$ are expressed as

$$V_{vertical} = 2500 \cos\alpha \sin A_p \tag{3}$$

$$L_{freefall} = 2500 \cos\alpha \sin A_p \frac{T_{burn}}{2} \tag{4}$$

where α (= 5 deg) is FPA, T_{burn} (= 20 sec) is the burning duration of the solid rocket motor. The attitude offset of 1 deg causes 43.4 m/s residual vertical velocity and 434 m altitude error. The altitude error causes longer free-fall length which corresponds to 37.5 m/s residual vertical velocity, according to equation (5).

$$V_{vertical} = \sqrt{2 \mathrm{g_M} L_{freefall}} \tag{5}$$

where $\mathrm{g_M}$ is the gravity of the moon. The orbit determination error and the ignition timing error of the solid rocket motor also affect the terminal altitude. The altitude error of 100 m corresponds to 100 m orbit determination accuracy and the timing error of 0.46 s.

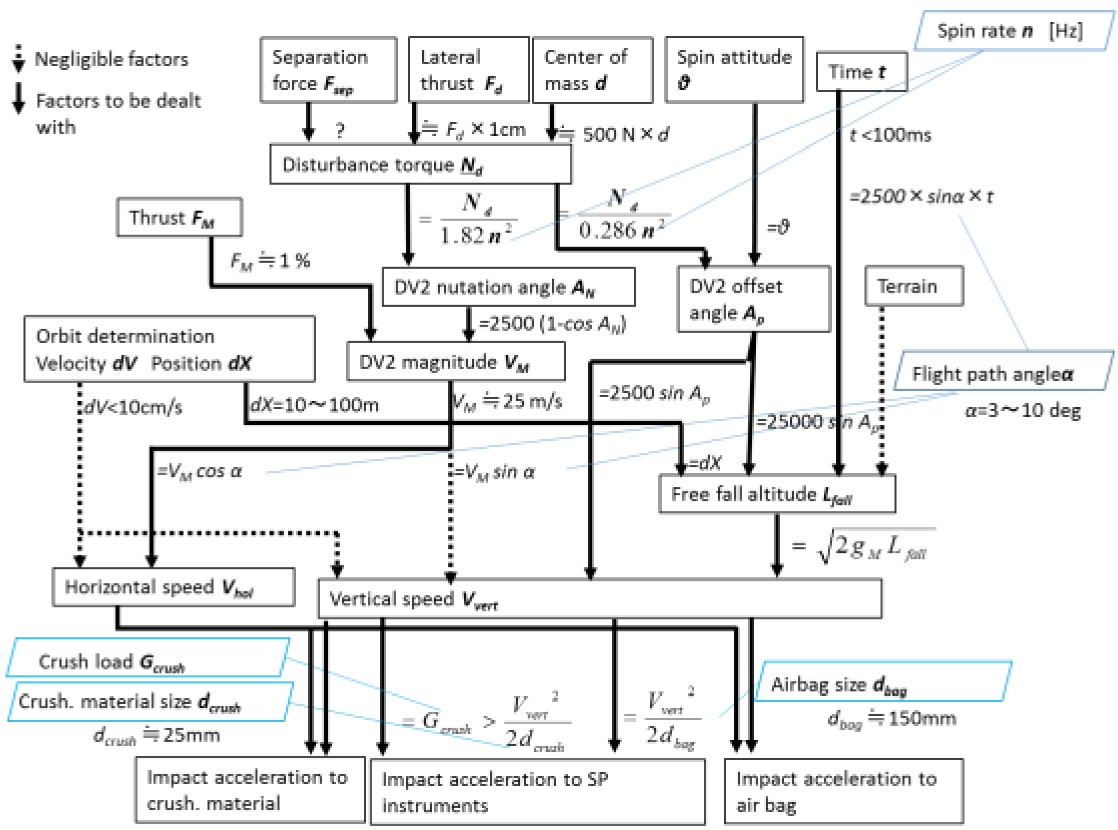

Fig. 6 The error analysis for the impact velocity and acceleration

5. Conclusion

The mission overview and AOCS design of OMOTENASHI spacecraft has been described. Currently, we are testing the flight model hardware and finalizing the OBC software. In parallel, we are conducting error sensitivity analysis for the residual impact velocity [10]. We are preparing to measure the separation disturbance form the OM.

Acknowledgments

This project is conducted by authors in collaboration with many researchers and engineers in JAXA, contractor companies, and universities. The authors appreciate their contribution.

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