Aditya-L1: Reference Attitude design and analysis

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***Abstract* – Aditya-L1 is the first solar observation mission from India which aims to study sun's upper atmosphere and corona, along with in-situ particle and plasma measurement, from halo orbit around Sun-Earth L1 point. The spacecraft reference attitude design holds a key role in the mission's success, as the satellite subscribe to the designed attitude immediately after the launch and throughout its mission life. The reference attitude design should cater to all mission requirements and sensor constraints in all the phases of mission life namely earth-bound phase, cruise phase and the halo orbit phase. The earth-bound phase further comprises of nominal (non-manoeuvre) and manoeuvre subphases. The nominal earth-bound phase has distinct attitude constraints pertaining to onboard thermal and communication constraints. In this phase attitude was constructed such that Earth is constrained always along negative Yaw axis while sun remained in Yaw-Roll plane but only in the +Roll hemisphere. On the other hand, during earth-bound manoeuvre, the attitude design goal was to ignite the spacecraft engine to change the trajectory of the satellite and thus includes five Earth bound manoeuvres (including the Trans L1 Injection). In the cruise phase, the spacecraft was primarily in the power generation attitude along with satisfying the sensor constraints. Furthermore, the spacecraft attitude during Trans Correction Manoeuvres and Halo Orbit insertion is similar to any manoeuvre when the spacecraft thrust axis is aligned in the appropriate direction along with satisfying the sensor and operational constraints. Finally, in the halo orbit phase, the principal objective is to perform precise payload operations. Considering the high attitude accuracy requirements onboard, the reference attitude should be simple yet accurate (better than 1 arc-second) in a way that +Yaw axis points to sun. This paper would highlight Aditya-L1 reference attitude design and analysis for all the above-mentioned phases of the mission.**

I. INTRODUCTION

Sun being the nearest star and the most dominant source of energy on Earth requires special attention in terms of fundamental understanding, exploration and scientific advancement. Solar observation strengthens our understanding in many ways including dynamics and evolution of celestial bodies, predicting space weather that consequently affects climate on earth and comprehending knowledge of astrophysics and stellar observations. [1] Studying sun via satellites offer greater benefits by facilitating long term continuous observation using advanced instruments and thereby enabling multi wavelength investigation. Additionally, the Lagrange point L1 between Sun and Earth is an ideal vantage point for solar observation, as it offers continuous, unobstructed view of sun. Furthermore, it enables early detection and monitoring of significant solar phenomenon like solar flares and coronal mass ejections. In view of the above, several space agencies around the world has launched solar observation satellites around the Sun-Earth L1 point.

India launched its first solar observation satellite, Aditya-L1 on September, 2nd 2023 which was subsequently captured in a halo orbit around the Sun-Earth L1 point on January 6th 2024. The satellite was launched by PSLV in an elliptical orbit around earth [2]. Aditya-L1 primarily aims to study solar upper atmosphere dynamics, chromosphere and coronal heating, along with in-situ particle and plasma dynamics.[3] The spacecraft undergoes multiple manoeuvres in order to reach halo orbit around Sun-Earth L1 point. The journey of the spacecraft from Earth's orbit to Halo orbit can be divided into multiple phases. Each of the mission phase has its own significance and is defined by respective mission constraints. These mission constraints demand innovative design and meticulous planning from each satellite subsystem. From flight dynamics point of view, the satellite's position, velocity and attitude remain the foremost area of concern. In this paper, reference attitude algorithm design of Aditya-L1 is presented in detail. The satellite orientation in each of the phases are predesigned such that it meets mission requirements and facilitate benign environment for the healthy functioning of every subsystem onboard. In view of this, reference attitude is defined and the spacecraft is commanded to follow it as closely as possible. Section 2 presents a brief overview of the satellite and its subsystem. In section 3, spacecraft mission constraints are outlined from spacecraft attitude point of view. These constraints will be explicitly defined for each of the mission phase. Section 4 will illustrate the algorithm design for the reference attitude adopted in each of these phases. In section 5 results are presented as per onboard data, thereby justifying the constraints.

II. ADITYA-L1 SPACECRAFT

The Aditya-l1 spacecraft was configured with a mass of about 1475 kg and power generation capability of 1820W. The spacecraft accommodate seven different payloads in order to meet its prime objectives:

1. Visible Emission Line Coronagraph (VELC) observes Sun corona (specifically from 1.05 solar radii to 3 solar radii) in visible and infra-red spectrum [3]
2. Solar Ultraviolet Imaging Telescope (SUIT) studies lower and middle part of Sun's atmosphere in via specific filter operating in 200 – 400 nm ultra violet region. It thus enables to focus on photosphere, chromosphere and the transitional region from chromosphere to corona. [4]
3. Aditya Solar Wind Particle Experiment (ASPEX) analyses in situ solar energetic particles [5],
4. Plasma Analyser Package for Aditya (PAPA) examines composition of solar wind and its energy distribution [6],
5. Solar Low Energy X-ray Spectrometer (SoLEXS) focus on X-ray flares on sun [7],
6. High Energy L1 Orbiting X-ray Spectrometer (HEL1OS) investigates dynamic event on sun in x-ray region [8]
7. Magnetometer studies the Interplanetary Magnetic Field (IMF).

For more details on payload one can refer to [8] [9].

The spacecraft mainframe elements consist of a mix of conventional IRS and interplanetary spacecraft bus that includes subsystem such as structure, mechanism, propulsion, power, on board computer, data handling systems, attitude sensors like star sensor, gyros and sun sensors, actuators like reaction wheels and thrusters. [9] [10]

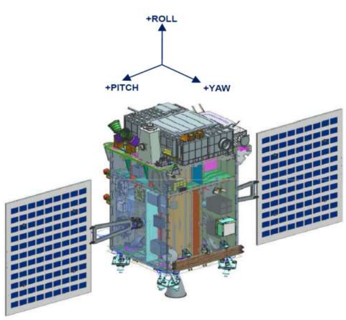

Figure 1 Aditya-L1 Spacecraft schematic

The attitude and orbit control subsystem (AOCS) of Aditya L1 is constituted as 3-axis stabilized platform. It receives input attitude data via star sensors (provides inertial quaternion), sun sensor (provides sun position in body frame) and gyros (provides spacecraft body rate). Star sensor (arguably the most important among all) is a sensitive instrument and thus needs to be protected from bright objects like Sun, Earth Albedo and Moon for appropriate functioning [11].

In order to align the spacecraft to the uplinked reference attitude, the AOCS generates appropriate command for the actuators. The functional requirement of AOCS includes acquisition of 3 axis stable attitude and co aligning spacecraft attitude to reference attitude in all phases of the mission.

Post launch, the spacecraft journey from earth bound orbit to halo orbit can be divided into three major phases (and subsequent sub phases) namely:

1. **Earth Centric phase** - After injection of Aditya-L1 spacecraft underwent 4 orbit (apogee) raising manoeuvres using Liquid Engine Burns (LEB). After these burns, the trans-L1 insertion burn ensured that the satellite escape from the earth sphere of influence (SOI). The Earth centric phase can be further distinguished into
   a. **Nominal (non- manoeuvre) phase** – It refers to all the duration in the Earth centric phase when no manoeuvre is performed.
   b. **Manoeuvre phase** – It denotes only those discrete small durations when the satellite undergoes orbit raising engine burns.
2. **L1 Transfer phase** -This phase starts once the spacecraft crosses earth's sphere of influence and follows a cruise trajectory around the sun until it reaches the defined Halo orbit around L1. The satellite spent ~101 days in this phase before reaching the halo orbit.
3. **L1 centric phase** – Once the spacecraft reaches halo orbit insertion point and is ready for the halo orbit insertion, the L1 centric phase kicks off. This phase remains active until the spacecraft completes its life time.
   a. **L1 Insertion phase** – It refers to the small engine burn duration when the spacecraft enters the halo orbit (from the transfer trajectory)
   b. **L1 centric Normal phase (or the Halo phase)** – This phase designates the payload operation phase in which the payload view axis (+Yaw) is accurately aligned to the sun for observation and data collection.
   c. **Station Keeping Phase** – This refers to the small engine burn phase which happens time to time for correcting the deviations in a halo orbit

III. ADITYA-L1 MISSION CONSTRAINTS

All the above-mentioned phase of Aditya-L1 spacecraft has specific objectives which further demands unique requirements and constraints. From the spacecraft reference attitude perspective following are the constraints which needs to be satisfied in different phases of the mission:

1. In the Earth centric-nominal phase (when payload is off), Earth should always align to -ve Yaw direction (i.e. Right-hand circular polarization (RCP) for communication) whereas Sun should always remain in +ve Roll hemisphere (thermal constraints) and in the Yaw-Roll plane (power constraint for solar array pointing).
2. During the manoeuvres, both the Star sensor's (SS)

boresight must be away from both sun and earth albedo by more than 40.0 deg and from moon by more than 20.0 deg
3. In L1 centric phase (when payload is on), Sun should be avoided on +Roll side of the S/c Body due to thermal constraint of the payload.
4. Communication in terms of TM and TC to be ensured during all critical operations.
5. For all attitude orientations RCP should always be ensured towards earth side (as per Deep Space Network requirement). Polarizations change from RCP to LCP and vice versa should be avoided. In case of any change over, during burn or calibration station needs to be informed a priori.
6. The reference attitude in L1 centric phase (or the halo phase) should be accurate better than 1 arc second. This would ensure overall pointing accuracy within the limit of 15 arc sec. Furthermore, the algorithm should be simple and computationally efficient for reliable onboard implementation.
For healthy spacecraft functioning, it becomes critical to satisfy all the above constraints during all mission phases. Furthermore, the reference attitude design should be neat, computationally efficient and reliable for onboard implementation.
Relevant Information:
SS1 Boresight Vector in body frame = [0.0, 0.707, 0.707]
SS2 Boresight Vector in body frame = [-0.707, 0.707, 0.0]
RCP antenna mounted along -Yaw axis
LCP antenna mounted along +Yaw axis
LAM Thrust direction along +Roll axis

IV. REFERENCE ATTITUDE DESIGN

The spacecraft reference attitude design is an important aspect while realizing a satellite mission. The satellite's attitude control system keeps the satellite aligned to the reference attitude immediately after the launch and throughout its mission life. The reference attitude in Aditya-L1 is required for following main reasons:
- To be used in the nominal earth-bound phase. This can be used as a target quaternion in the attitude controller.
- To be used to orient spacecraft appropriately for spacecraft engine firing during manoeuvres.
- Initial alignment estimation between the payload and sensor alignment axis
- To be used to arrive at Sun pointing when completing a station keeping manoeuvre
- To be used to arrive at Sun pointing after loss of lock
- To arrive at Sun pointing after desaturation

The reference attitude design for each of the phases differ from each other

1. **Earth Centric phase** – The Earth centric phase reference attitude for Nominal and Manoeuvre durations are different.

a. Nominal (non- manoeuvre) phase:
In order to meet the constraint (thermal and power specifically), the reference attitude design offers a yaw steering profile which ensures following:
- -ve Yaw axis is aligned towards Earth.
- Sun continually remain in (or near) the Yaw-Roll plane, but constrained towards +ve Roll hemisphere.

The reference attitude (reference quaternion) is computed via Model driven approach (an onboard algorithm). The model driven reference quaternion computation utilizes orbit information (available onboard) and inertial target pointing vector (which is earth centre in our case) as inputs for the algorithm.
Model driven pointing quaternion computation has following steps

- Step 1: Unit Yaw vector ($\vec{\mathrm{Y}}$) in inertial frame is computed using inertial target vector ($\vec{\mathrm{T}}_{\mathrm{g}}$) here it is Earth centre and satellite position vector ($\vec{\mathrm{R}}_{\mathrm{s}}$) $\vec{\mathrm{Y}}$ is then normalized

$$\vec{\mathrm{Y}} = \vec{\mathrm{T}}_{\mathrm{g}} - \vec{\mathrm{R}}_{\mathrm{s}} \qquad (1)$$
$$\vec{\mathrm{Y}} = norm(\vec{\mathrm{Y}}) \qquad (2)$$

- Step 2: Unit Pitch vector ($\vec{P}$) is defined perpendicular to Yaw vector and spacecraft velocity vector ($\vec{\mathrm{V}}_{\mathrm{s}}$). $\vec{\mathrm{P}}$ is then normalized

$$\vec{\mathrm{P}} = \vec{\mathrm{Y}} \, \mathrm{x} \, \vec{\mathrm{V}}_{\mathrm{s}} \qquad (3)$$
$$\vec{\mathrm{P}} = norm(\vec{\mathrm{P}}) \qquad (4)$$

- Step 3: Roll vector completes the right handed triad.

$$\vec{\mathrm{R}} = \vec{\mathrm{P}} \, \mathrm{x} \, \vec{\mathrm{Y}} \qquad (5)$$
$$\vec{\mathrm{R}} = norm(\vec{\mathrm{R}}) \qquad (6)$$

- Step 4: Attitude constructed using these $\vec{Y}$, $\vec{R}$ and $\vec{P}$ is termed as Pointing Attitude matrix (A_P) and is is given by

$$\mathrm{A}_P = \begin{pmatrix} Y^x & Y^y & Y^z \\ R^x & R^y & R^z \\ P^x & P^y & P^z \end{pmatrix} \qquad (7)$$

- Step 5: Pointing Attitude matrix (A_P) is converted to Pointing Quaternion (Q_P)
- Step 6: In order to keep Sun in Yaw-Roll plane, Sun vector is then computed in pointing body frame ($\vec{\mathrm{S}}_P$). $\vec{\mathrm{S}}_I$ denotes sun vector in inertial frame. This is needed for computation of body yaw steering quaternion.

$$\vec{\mathrm{S}}_P = \begin{pmatrix} S_p^x \\ S_p^y \\ S_p^z \end{pmatrix} = A_P . \vec{S}_I \qquad (8)$$

- Step 7: Magnitude of the cosine of the angle between sun vector $\vec{S}_p$ and yaw vector ($\vec{Y}$) is computed as

$$\mathrm{AbsCos}\theta_{SY} = \mathrm{fabs}(S_p^x) \qquad (9)$$

- Step 8: If sun vector and yaw vector are nearly parallel to each other sun vector is approximated ($\vec{S}_A$ i.e. a proxy sun vector is fed to the algorithm) otherwise actual sun vector in pointing body frame is taken.

$$\vec{S}_A = \vec{S}_P \qquad (10)$$

If ($\mathrm{AbsCos}\theta_{SY} > Cos(\theta_{limit})$

$$\vec{S}^z_{approx} = K_{amp} \exp(-K_{exp}(1.0 - AbsCos\theta_{SY})) \qquad (11)$$

$$\vec{S}^z_A = \vec{S}^z_P + \alpha \vec{S}^z_{approx} \qquad (12)$$

where
α = -1.0 (if $\vec{S}^z_P < 0.0$)
α = 1.0 (if $\vec{S}^z_P \geq 0.0$)
K_{amp} and $K_{exp}$are tuneable hyper parameters with values 0.18 and 20.0 respectively.

- Step 9: The sun vector $(\overrightarrow{S_A})$ is normalized and the required yaw steering angle (ψ), to keep sun in the Yaw-Roll plane is computed

$$\psi = arctan2(\vec{S}^z_A, \ \ \vec{S}^y_A) \qquad (13)$$

- Step 10: Then Yaw steering quaternion (q_Y) is computed as

$$q_Y = \begin{pmatrix} q_Y^1 \\ q_Y^2 \\ q_Y^3 \\ q_Y^4 \end{pmatrix} = \begin{pmatrix} \sin(\psi/2.0) \\ 0.0 \\ 0.0 \\ \cos(\psi/2.0) \end{pmatrix} \qquad (14)$$

- Step 11: Final inertial to body quaternion is obtained using pointing and yaw steering quaternion as

$$q_{IB} = Q_P \ * q_Y \qquad (15)$$

where * represents quaternion multiplication

- Step 12: Additional Roll bias of 180 deg is given such that the -Yaw axis points to Earth centre (instead of +Yaw) and Sun still remains in the Yaw-Roll plane. (Provision was also made to give additional bias about any of the axis if required.).

$$Q_{final} = q_{IB} \ * q_{bias} \qquad (16)$$

b. Manoeuvre phase:

Since the Liquid Apogee Motor (LAM) is mounted in the –Roll side, the LAM thrust is directed towards +Roll. In order to increase the apogee height, following attitude was constructed on ground:

- Step 1: Roll vector ($\vec{R}$) was aligned to Spacecraft Velocity ($\vec{V}_s$) direction

$$\vec{R} = \mathrm{norm}(\vec{V}_s) \qquad (17)$$

- Step 2: Pitch axis was constructed perpendicular to Roll and spacecraft position vector.

$$\vec{P} = \mathrm{norm}(\vec{R} \times \vec{R}_s) \qquad (18)$$

- Step 3: Yaw vector completes the right handed triad.

$$\vec{Y} = \vec{R} \times \vec{P} \qquad (19)$$

- Step 4: Attitude Matrix (A_{man}) is constructed as in (7)
- Step 5: Attitude matrix is converted to Quaternion (Q_{man})
- Step 6: Additional bias about the Roll axis was provided such that the star sensor avoids Sun, Earth albedo by more than 40 deg and Moon by more than 20 deg. Bias in case of Aditya-L1 also ensured that no changeover of polarization for ground antennas. Rotation about Yaw and Pitch was strictly avoided as that will disturb the thrust direction.

$$Q = Q_{man} * Q_{bias} \qquad (20)$$

The four components of quaternion computed here are fitted using Chebyshev polynomial fit on the ground. The fitted coefficients are uplinked in order to reconstruct the attitude onboard.

2. **L1 Transfer phase** – In this phase, the +Yaw axis was aligned to Sun, +Roll point near the North Ecliptic Pole whereas +Pitch completes the triad. The quaternion was constructed using the given definition on ground and was subsequently fitted via Chebyshev polynomial fit. The coefficient of the Chebyshev polynomial is uplinked to reconstruct the attitude. One single Chebyshev fit was sufficient to fit for the entire 101 days of cruise attitude in terms of pointing accuracy requirement. In the cruise phase the primary goal was to maintain an attitude that generates sufficient power and thus the pointing accuracy requirement was much benign (better than ~1 deg).

3. **L1 centric phase** – The reference attitude construction for this phase can be classified into two categories as payload pointing and the manoeuvre. The L1 insertion phase and the station keeping phase comes under the manoeuvre attitude category while L1 Centric normal phase belongs to payload pointing category.

a. L1 insertion phase and the station keeping phase:

The engine burn duration for L1 insertion phase and the station keeping phase is extremely small (tens of seconds) and thus demands a single quaternion throughout the burn (unlike the Earth centric manoeuvre phase where a Chebyshev quaternion profile was uplinked).

- Step 1: Roll was aligned to the required Thrust direction ($\vec{T}_s$)

$$\vec{R} = \mathrm{norm}(\vec{T}_s) \qquad (21)$$

- Step 2: Pitch axis was constructed

perpendicular to Earth ($\vec{E}$) and Roll vector ($\vec{R}$).

$$\vec{P} = \mathrm{norm}(\vec{E} \times \vec{R}) \qquad (22)$$

- Step 3: Yaw vector completes the right handed triad.

$$\vec{Y} = \vec{R} \times \vec{P} \qquad (23)$$

- Step 4: Attitude matrix was constructed
- Step 5: Attitude matrix was converted to Quaternion.
- Step 6: Additional bias about the Roll axis was provided to meet star sensor and polarization constraints.

The above computation was done on ground and the Quaternion was uplinked onboard.

b. <u>L1 Centric normal phase:</u>

L1 Centric normal phase requires accurate reference attitude construction for precise pointing of payload axis towards Sun (reference attitude accuracy better than ~1 arcsecond and spacecraft pointing accuracy better than 30 arcsecond). The reference attitude design requires Satellite position, Sun position and North Ecliptic pole vector (NEP) as inputs. Satellite position is derived through the orbit determination process and is propagated for future using full force models (in the sun centred frame – thus Sun position remains at origin). NEP vector is an inertial vector which will remain constant throughout the mission. The attitude in this phase was constructed as follows:

- Step 1: Yaw vector, which is pointing from Spacecraft towards Sun, is constructed at each time instance for 10 days (at sampling of about 100 sec). This computation is done on ground.

$$\vec{R} = \mathrm{norm}(\overrightarrow{\mathrm{Sun}} - \vec{R}_s) \qquad (24)$$

- Step 2: Yaw (derived in step 1 and available at every 100 sec for 10 days) components in x, y, z directions are fitted using 14th degree polynomial each. The coefficients are uplinked on-board to reconstruct Yaw axis
- Step 3: Pitch axis was constructed perpendicular to Yaw and North Ecliptic Pole vector (NEP). This step is done on board.

$$\vec{P} = \mathrm{norm}(\vec{Y} \times \overrightarrow{\mathrm{NEP}}) \qquad (25)$$

- Step 4: Roll vector is constructed as

$$\vec{R} = \vec{P} \times \vec{Y} \qquad (26)$$

- Step 5: Attitude matrix was constructed
- Step 6: Attitude matrix was converted to Quaternion. Provision was also made to give additional bias to nullify misalignments

V. RESULTS AND CONCLUSIONS

1. Earth Centric Nominal phase – As per design, the earth is exactly aligned to the -ve Yaw axis while the sun always remains in the Yaw-Roll plane, constrained in the +ve Roll hemisphere. The spacecraft in this phase is continuously rotating about Yaw axis. A sample variation of Sun angle from +Yaw towards +Roll (Sun RA in spacecraft body frame) is shown for post injection elliptical orbit. Similar sun profile is obtained for all times in non-manoeuvre phase. Note that since Sun always remain in Yaw-Roll plane, Sun vector's Pitch component is always zero and its RA varies from 0 to 180 deg only (indicating Sun available only in +Roll hemisphere). The sharp dip in Fig. 2 denotes near perigee region in the orbit where the Sun vector and Earth vector with respect to Satellite are almost aligned to each other. On the other hand, Sun RA of 180 deg in Fig. 1 refers to near apogee region where Earth and Sun are diametrically opposite to each other w.r.t. satellite.

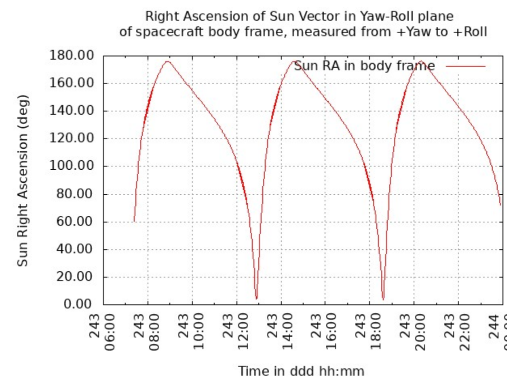

Figure 2 Right Ascension of Sun Vector in Yaw-Roll plane \n of spacecraft body frame, measured from +Yaw to +Roll (in degree). Similar profiles were observed post EBN-2,3,4,5 as the geometry was almost same (only the size of the orbit increased after every burn). Sun was always kept in the Yaw-Roll plane so the Pitch component was zero for Sun vector in body frame

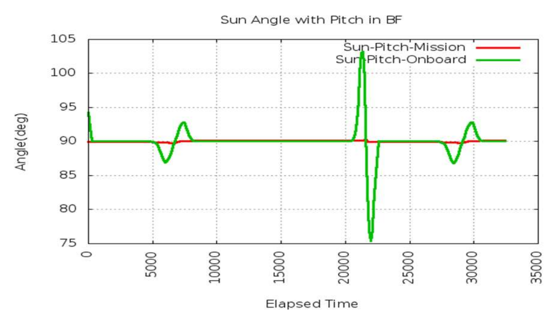

Figure 3 Sun in body frame is contained in Yaw-Roll palne for most of the time (thus Pitch to Sun angle is 90 deg for most of the time). The Pitch to Sun angle deviates when Sun comes near to +ve Yaw or -ve Yaw axis due to approximation. Red shows Pitch-Sun angle when Sun is always in the Yaw-Roll plane. Green shows onboard observation of Sun-Pitch angle.

2. Earth Centric Manoeuvre phase – The orbit raising manoeuvre were performed at perigee which also happens to be the eclipse region of the orbit. The reference attitude was constructed as per previous section. An additional bias of 50 deg was provided so that Star sensor clears Earth albedo by 40 deg. Since the sun is positioned towards apogee for all the burns, SS

also clears sun by more than 40 deg. Meanwhile the Sun earth spacecraft geometry was similar for all the 5 burns, same bias was applicable and similar angle profiles were observed for all burns. A sample sun RA and Declination profile (in body frame), Star sensor1&2 angles with respect to Sun, Earth albedo and Moon is shown in the plots below. The plot refers to the first Earth bound manoeuvre. The angle profile for all other burns were very similar.

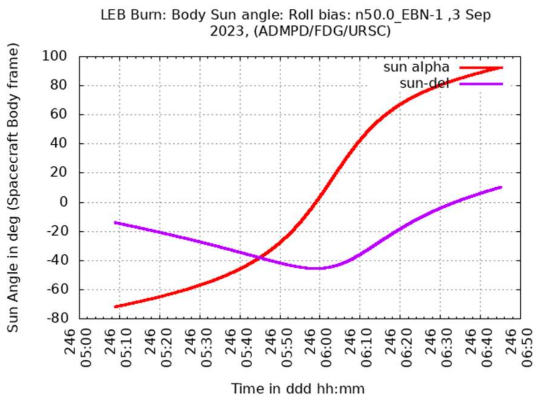

Figure 4 Sun position with respect to spacecraft body frame during the manoeuvre EBN-1. Sun alpha (in red) is the right ascension angle of Sun in the Yaw -Roll plane (measured from +Yaw towards +Roll). Sun del (in purple) is the sun declination angle, measured from Yaw Roll plane towards +Pitch Similar profiles were observed for EBN-2,3,4,5 as the geometry was almost same for all burns.

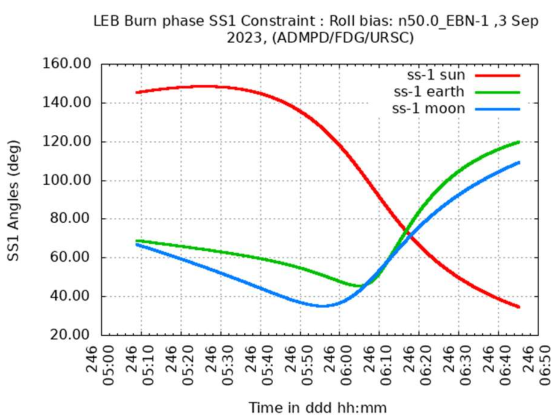

Figure 5 Star Sensor 1 angles with Sun (in Red), Earth albedo(in green) and Moon (in blue) during the manoeuvre EBN-1. Similar profiles were observed for EBN-2,3,4,5 as the geometry was almost same for all burns.

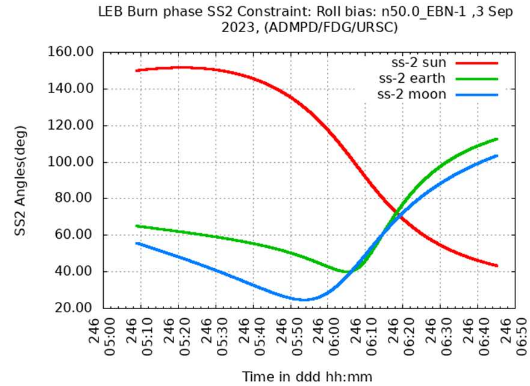

Figure 6 7 Star Sensor 2 angles with Sun (in Red), Earth albedo(in green) and Moon (in blue) during the manoeuvre EBN-1. Similar profiles were observed for EBN-2,3,4,5 as the geometry was almost same for all burns.

3. L1 Transfer phase – In this phase, the Star sensor clears Sun by design as the Yaw axis is pointed towards Sun and both the Star sensor boresight is directed towards -Yaw hemisphere. For telemetry and telecommand the purposes, Earth is ensured in the -Yaw hemisphere thus enabling RCP for the ground antennas.

4. L1 Centric phase:
   a. L1 Insertion phase:

Table 1 Aditya-L1 spacecraft attitude and its properties during Halo orbit insertion.

Epoch	2024 1 6 10 31 40 811
HOI Quaternion	0.1790 -0.3182 0.3447 0.8647
Inertial Thrust Vec	-0.7102 0.6981 0.0901
Inertial Thrust Vec RA, Declination	135.492 deg, 5.173 deg
Sun to SS1	122.7 deg
Earth to SS1	54.7 deg
Moon to SS1	57.2 deg
Sun to SS2	102.3 deg
Earth to SS2	80.1 deg
Moon to SS2	77.6 deg
Sun in Body	Yaw: 122.5 deg, Roll: 147.1 deg, Pitch: 85.7 deg
Roll To Sun:	147.09 deg
Earth To Roll	35.14 deg
Earth Albedo	0.29 deg

   b. L1 Centric Phase - By design the Yaw axis is much accurate than 1 arc second. The star sensor mounting takes care of the constraint's angles.

The attitude construction by the given method have several advantages:

a. Yaw axis of reference attitude is accurate up to 1 arcsec (assuming accurate sun position availability)
b. Inaccuracy in NEP vector would not affect the accuracy of Yaw axis (payload view axis) as NEP just acts as an intermediate vector in the attitude design.
c. The duration of fit (here taken as 10 days) order of polynomial and the sampling (100 sec) can be tuned based upon the convenience.
d. The design also offers operational flexibility. For instance: If there is an operational feasibility to uplink polynomial coefficients more frequently (say 4 days), a lower degree polynomial fit (9^{th} order) is enough to provide yaw accuracy of better than 1 arcsecond. Thus, requiring fewer coefficients to uplink every 5 days

The attitude construction is simple, accurate, computationally efficient and reliable for on-board implementation. It just requires up linking of 45 polynomial coefficients every 10 days. As per Table 2, Yaw accuracy of 14-degree polynomial fit for 10 days is two order better than 1 arcsecond, thus one can even fit for greater number of days (more than 10) if satellite propagated position information accuracy is acceptable.

Table 2 Yaw axis (Payload boresight axis) accuracy variations on factors of duration of fit(in days) and order of fit in the L1 centric nominal phase.

Duration (days)	**Yaw axis accuracy (arc-second)**		
	Number of coefficients		
	6 (5 deg poly. fit)	10 (9 deg poly. fit)	15 (14 deg poly. fit)
4	0.147	0.00532	0.00532
7	3.635	0.00532	0.00532
10	16.5368	0.0292	0.00532

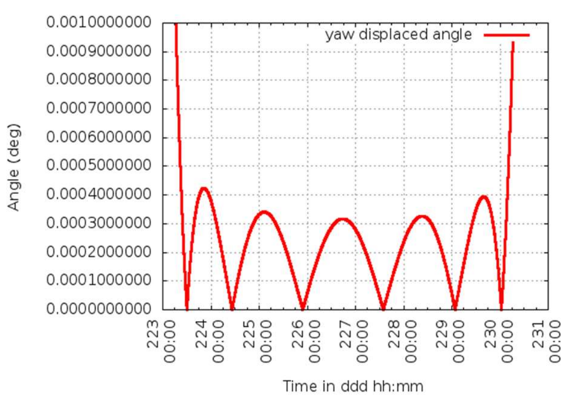

Figure 8 Yaw axis inaccuracy as a function of time, considering 4 days fit and 6-degree polynomial order

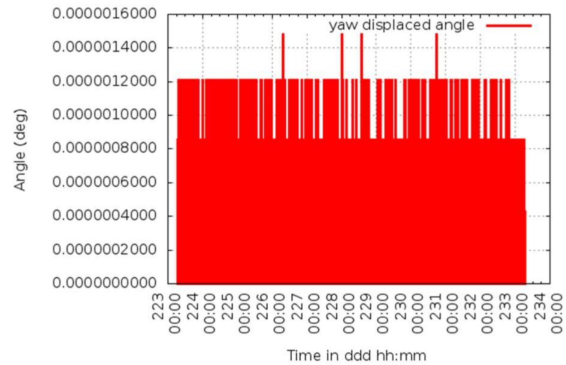

Figure 9 Yaw axis inaccuracy as a function of time, considering 10 days fit and 14-degree polynomial order.

c. L1 Station Keeping:

Table 3 Aditya-L1 spacecraft attitude and its properties for the 1st station keeping manoeuvre in Halo orbit.

Epoch	2024 2 22 7 0 0 0
HOI Quaternion	0.13145 0.18890 -0.83200 0.50477
Inertial Thrust Vector	0.8896 -0.4190 -0.1816
Inertial Thrust Vec RA declination	334.778 deg, -10.465 deg
Sun to SS1	45.00 deg
Earth to SS1	129.84 deg
Moon to SS1	135.00 deg
Sun to SS2	45.00 deg
Earth to SS2	160.0 deg
Moon to SS2	134.92 deg
Sun in Body	Yaw: 89.72 deg, Roll: 0.27 deg, Pitch: 89.9 deg
Roll To Sun:	0.27 deg
Earth To Roll	154.97 deg
Earth Albedo	0.23

VI. ACKNOWLEDGEMENT

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VII. References

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