

## Cosmic Dawn Intensity Mapper: Spacecraft and Mission Design for a Probe-Class Space Telescope

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Cosmic Dawn Intensity Mapper (CDIM) is a Probe-class near-IR space telescope with the scientific goal of conducting large spectro-imaging surveys over a five-year mission in the next decade. A high-level system architecture was designed to identify key features and technologies aboard the CDIM spacecraft in preparation for more detailed studies such as a Team-X session at NASA Jet Propulsion Laboratory.

*Keywords:* spacecraft, telescope, system, cryogenic, infrared, design.

### 1. Introduction

Derive context for the case for CDIM from CDIM 2 page (Cooray *et al.*, 2016).

Probe-class missions occupy a role on a larger scale than Discovery missions, such as Kepler and Dawn, but not as large as Flagship missions such as JWST. Such missions are intended to be PI-led scientific investigations rather than general observatories, and have a firm \$1B cap (Wiseman *et al.*, 2015).

The Cosmic Dawn Intensity Mapper (CDIM) is a concept for a Probe-class 1.5 m aperture telescope, passively cooled to 45 K, with an actively cooled  $6 \times 6$  detector array that utilizes linear variable filters (LVFs) (Cooray *et al.*, 2016) capable of three-dimensional spectro-imaging observations over the wavelength range of 0.75 to 7.5  $\mu\text{m}$  at a spectral resolving power  $R = 500$ . CDIM has a  $10 \text{ deg}^2$  instantaneous field of view (FoV) atop a focal plane of thirty-six  $2048 \times 2048$  detectors. The survey strategy using spacecraft operations following a shift and stare mode will result in 1360 independent narrow-band spectral images of the sky on a given location. Surveys are planned to span from  $25 \text{ deg}^2$  up to  $1000 \text{ deg}^2$  over a five year lifetime in an orbit about Sun-Earth Lagrange point  $L_2$ .

CDIM is optimized to search for the first cosmic sources of dust and evidence of the very earliest stellar populations, bridging the gaps in the JWST and WFIRST cosmic dawn surveys and exceeding them in capability. Critical design requirements are summarized in Table 1.

### 2. Optical Telescope Assembly

Preliminary explorations indicate that a 1.3–1.5 m aperture off-axis primary mirror cooled to 45 K is required to meet CDIM’s spectro-imaging requirements (Cooray *et al.*, 2016). The primary mirror is notionally assumed to be constructed from light-weighted Corning (ultra-low expansion) silica-titania glass with a honeycomb core and a gold-deposition surface coating. For this type of mirror, the estimated mass is in the neighborhood of 200 kg.

**HgCdTe-on-silicon** infrared detectors meet CDIM design goals of operating at cryogenic temperatures, low in cost, and, of course, sensitive in near-IR. Several different off-the-shelf CMOS detectors satisfy CDIM’s

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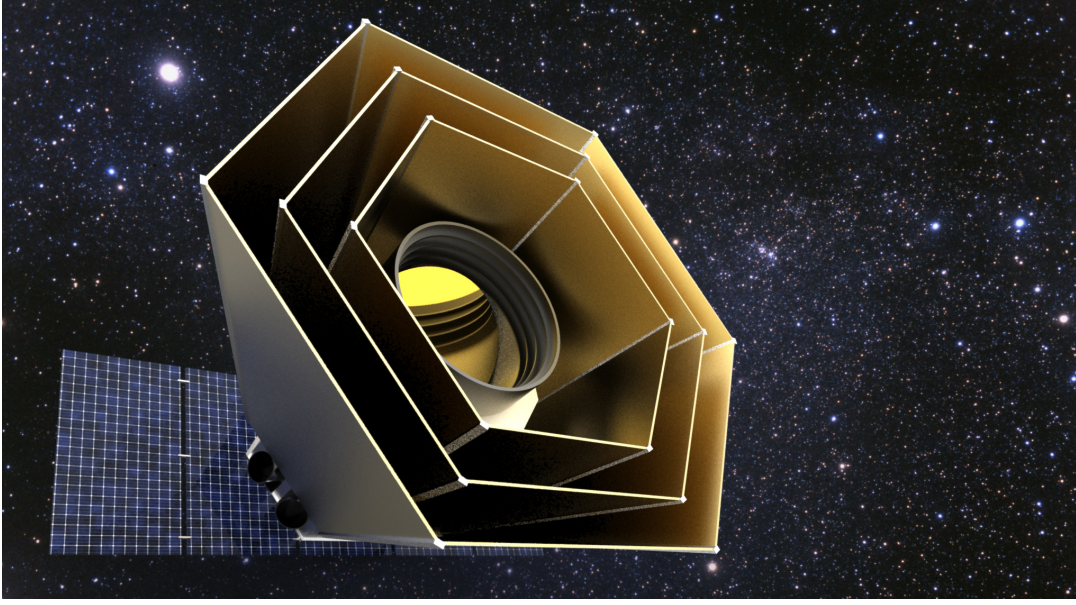


Fig. 1. An artistic rendition of the Cosmic Dawn Intensity Mapper stationed at Sun-Earth  $L_2$ .

Table 1. Critical design requirements for the CDIM spacecraft.

Spacecraft Design Driver	Impact	Target
Cost	Science capability, instrument architecture	less than \$1 B
Mass	Launch vehicle	less than 1000 kg
Temperature (OTA)	Passive radiator	45 K
Temperature (Detector)	Cryocooler	35 K
Pointing Requirements	Attitude control system	less than 0.5 arcsec
Lifetime	Redundancy, RCS propellant	5 years
Orbit	Solar array, thermal management, launch vehicle, telemetry	Sun-Earth $L_2$

spatial resolution, wavelength range, and sensitivity requirements. These detectors range in technology-readiness-level (TRL), but all are sufficiently developed to be considered for the 2020 Decadal and will be demonstrated on missions such as NEOCam, SPHEREx, and JWST. **Citation needed.**

Teledyne H2RG-18 HyViSI detectors offer a  $2048 \times 2048$  pixel array format at a pixel pitch of  $18 \mu\text{m}$  (Bai *et al.*, 2008). CDIM will utilize 36 detectors in a close-packed,  $6 \times 6$  mosaic focal plane assembly (FPA). In the expected normal operating mode, each detector dissipates less than 4 mW, for a total power dissipation of less than 150 mW for the full array.

Linear-variable filters (LVFs) will be placed just above the detectors to provide spectral dispersion for spectrometry. LVFs are simple, space-qualified solutions to permit spectral data cubes between  $0.75\text{--}7.5 \mu\text{m}$  that are commercially available and significantly lower in cost than more complex systems. **Citation needed.**

CDIM optics, instruments, and focal plane will be housed in a light-tight box **to minimize signal contamination from stray light. The light-tight box is constructed of gold-plated aluminum.** The optical telescope assembly (OTA) as a whole is estimated to have a **mass of  $x$  kg.**

### 3. Thermal Design

CDIM will employ both passive and active thermal regulation systems. By using passive radiators in tandem with an active cryocooler, the static OTA heat load can be dissipated by the lightweight radiator and a smaller cryocooler may be used to only cool the detector array rather than the whole OTA mass plus FPA. Passive cooling is used to cool the OTA and FPA from solar radiation, while active cooling regulates heat dissipated by active detectors.

The OTA is cooled to 45 K to reduce background photon load in the near-IR. Passive thermal regulation

is maintained using a multi-stage V-groove radiator, which bounces radiative heat into the 3 K background of space (Bard, 1987). V-groove radiators have been demonstrated in passive cryogenic radiators up to 4 K with Planck, SPIRIT, and Spitzer (warm mission). Like Planck, the active coolers are also pre-cooled to less than 50 K this way. **Citation needed: Planck CalTech Paper.** In order to achieve passive cooling from a baseline temperature of 300 K at Sun-Earth Lagrange point  $L_2$ , a  **$x$ -stage** V-groove radiator with an **area of  $x \text{ m}^2$**  is required. **Since CDIM may be at various angles of incidence to solar radiation as it surveys the sky, the V-groove radiator fins will extend outward, more like SPHEREx than Planck.**

CDIM's thermal requirement to have the detector array actively cooled to 35 K in order to reduce thermal noise can be met by a commercially available mechanical cryocooler. Stirling-cycle mechanical refrigerators are low-vibration, high-reliability, and lightweight active cryocooling systems that have significant heritage in space applications. One candidate system is Raytheon's PSC 1-stage Stirling cryocooler, capable of cooling a 1.2 W parasitic heat load to 35 K. This cryocooler is 18.6 kg and requires 88 W of input power (Donabedian *et al.*, 2003).

#### 4. Attitude Determination and Control

To conduct a survey, the spacecraft must first understand its orientation and then act to align itself with a given area of the sky. Redundant systems of varying levels of fidelity are included to allow CDIM to operate in different power states. Low-fidelity attitude determination sensors such as sun sensors are cheap, accurate to less than one degree, and lightweight.

High-fidelity attitude determination is conducted by off-the-shelf star tracking cameras. Star trackers identify constellations in their field of view to determine the spacecraft's heading to within 0.25 arcseconds. **Identify candidate star trackers.**

In a heliocentric orbit, the primary disturbance to the spacecraft's heading is solar radiation pressure (SRP). At  $L_2$ , solar radiation pressure presents itself as torque on the spacecraft with a maximum on the order of  $10^{-5} \text{ N m}$ .

Cold-gas or hydrazine thrusters will be used for orbit station-keeping as well as momentum management. A desired heading is maintained by the spacecraft using a 3-axis zero-momentum inertial system, whereby the error in heading due to SRP is cancelled out by spinning up or slowing down reaction wheels. Reaction wheel systems are capable of torques ranging from .01 to 1 N m and store 0.4 to 3000 N m s of practical momentum (Wertz *et al.*, 2015). Power consumption varies with reaction wheel speed, with a maximum estimate of roughly 100 W. After some time the inertial attitude control may become saturated. Desaturation is managed by engaging station-keeping thrusters for short periods of time. Additionally, these thrusters will be used to maintain an orbit at  $L_2$  as it is an inherently unstable orbit.

#### 5. Flight Computer

**CDIM is capable of autonomous operation and system diagnostics. Nominal operation includes maintaining an attitude during imaging, capturing images, and downlinking data to Earth. Images will be processed on-board CDIM using an algorithm demonstrated by SPHEREx.**

#### 6. Telemetry

Typically for high-Earth and deep-space missions, the X-Band Space Science frequency band is used for uplink and downlink between the spacecraft and Ground Stations. Thus, high-gain antennas are best suited for both links (Wertz *et al.*, 2015).

A survey conducted by CDIM will generate **168.39 Gb** of data per day employing on-board data processing akin to SPHEREx (Zemcov *et al.*, 2016). With a compression ratio of 2.5:1, CDIM will downlink a total of 63.7 Gb/day during a survey. Transmission rates are dependent on the total time available for CDIM to send data to a ground station. For example, the spacecraft could transmit continuously at a very low transfer rate, or send larger volumes of data once per day over 1 hour at the expense of a higher transfer rate. **Example calculation of data to downlink 1 day in 1 hr.**

Uplinks will follow standard protocols and do not require transmitting large volumes or particularly fast transfer rates.

Table 2. For redundancy, CDIM is outfitted with multiple communication modes. Downlink transfer rates reflect estimates based on the target of 63.7 Gb/day. Typical data transfer rates are outlined for uplinks (Wertz *et al.*, 2015).

Mode	Uplink	Downlink
Emergency	7.8 bps	5–10bps
Engineering data	15.6–2000kbps	Up to 10 bps
Science data	15.6–2000kbps	0.74 Mbps (continuous) or 17.7 Mbps (1 hour per day)

**Identify candidate telemetry systems.** The CDIM telemetry system, including antenna and power converter, are roughly 2 kg.

## 7. Power

Since CDIM will be located at  $L_2$ , it is exposed to constant and significant solar flux. All power generation will come from an array of photovoltaic cells facing the sun. While the array may be fixed, the required area of the array is minimized if the cells are able to adjust to account for different incident angles to the sun. The array will deploy after the launch and orbital insertion phases of the mission.

Based on a rough power budget and the spacecraft’s position at  $L_2$ , **the array must be min  $x$ –max  $xm^2$**  in area to sustain operation. The dark side of the array acts as a radiator to contribute to the thermal regulation of the spacecraft bus.

**Identify candidate systems.**

## 8. Structure

The spacecraft bus houses all non-instrumentation systems including the cryocooler, ADCS, telemetry, processing, and power modules. The bus will also include hard points for integration with the launch vehicle. **The CDIM spacecraft bus will not feature novel technology.** Rather it will leverage high TRL or off-the-shelf components.

## 9. Launch Vehicle

CDIM will be comfortably within the mass and spatial limits of both currently available and development launch vehicles capable of delivering payloads to Sun-Earth Lagrange Point  $L_2$ . Future launch vehicles will be more than capable of delivering CDIM to  $L_2$ , and industry trends indicate that heavy and super-heavy vehicles will continue to come online by the time CDIM launches.

Due to the rigorous launch environment, CDIM solar panels and passive radiators will not be deployed until CDIM is delivered to a transfer orbit en route to  $L_2$ .

## 10. Cost Estimation

The overall cost of a space telescope may be broken down on the subsystem level. All conclusions based on statistical analysis are only as good as their databases. Fiscal data, such as what is required for proper analysis, is scarce. Estimations are made with engineering judgement based on available data.

To estimate the cost of the CDIM mission, costs are separated into drivers of the *mission cost*, which includes hardware, development, ground support, integration, testing, science, and management. Existing generalized parametric cost estimation approaches identify key cost drivers for mission cost (Stahl *et al.*, 2013; Bely, 2011), but do not take labor or overhead costs into account. Overhead and labor costs are included in a more robust model for *total cost*, where:

Table 3. Available launch vehicle configurations and their capabilities to send NASA payloads to L<sub>2</sub> (Rioux, 2016; Space Launch Report).

Vehicle	Payload to L <sub>2</sub>	Fairing size	Cost <sup>†</sup>
Falcon 9 v1.1	2900 kg	5.2 × 13.1 m	<b>\$97M</b>
Falcon Heavy <sup>*</sup>	14 000 kg	5.2 × 13.1 m	<b>\$120M</b>
Atlas V 551	6100 kg	4.2 × 10.0 m 5.1 × 11.0 m	\$153M
Ariane V	6600 kg	5.4 × 12.7 m 5.4 × 13.8 m 5.4 × 17.0 m	\$165M \$220M
Delta IV Heavy	9800 kg	5.0 × 14.3 m 5.0 × 19.1 m	\$375M

<sup>†</sup> Launch costs may be higher due to NASA mandated oversight and testing.

<sup>\*</sup> Costs and capacities are representative of design specifications for launch vehicles that are currently in development.

$$\text{Total Cost} = (1.5) \times \text{Mission Cost} \quad (1)$$

Stahl *et al.* present an approach to estimating OTA cost based on correlations with data on flown space telescope missions. CDIM's projected costs may be obtained from these findings with engineering judgement, knowing that the data is drawn from a relatively small sample set. Thus, an OTA aperture diameter of 1.5 m yields a median OTA cost of \$58.2M. Since OTA cost is found, estimates for other cost drivers may be obtained from relative cost values.

Driver	% Mission Cost	Est. Cost	
		1.3 m	1.5 m
OTA	13 %	\$48.0M	\$58.2M
Spacecraft	20 %	\$76.8M	\$91.3M
Instruments	15 %	\$57.6M	\$69.8M
Ground Support	5 %	\$18.8M	\$22.8M
Integration & Testing	7 %	\$25.6M	\$31.0M
Systems Engineering	4 %	\$16.0M	\$19.4M
Management	4 %	\$16.0M	\$19.4M
R&D	21 %	\$80.0M	\$97.0M
Science Team	10 %	\$37.6M	\$45.6M
Mission Cost		\$188.2M	\$456.5M
Labor & Overhead		\$376.5M	\$228.2M
Total Cost		\$564.7M	\$684.7M

Table 4. CDIM total cost breakdown by driver. **Make total cost less dependent on OTA cost. Ground support and science costs shouldnt rise so much along with OTA cost. Maybe just show one diameter and hide the other graph.**

The most robust model for the CDIM mission predicts its total cost to be \$684.7M, and even the more conservative model predicts CDIM costing to fall under \$800M. The CDIM mission has significant margin under the \$1B cap for Probe-class missions.



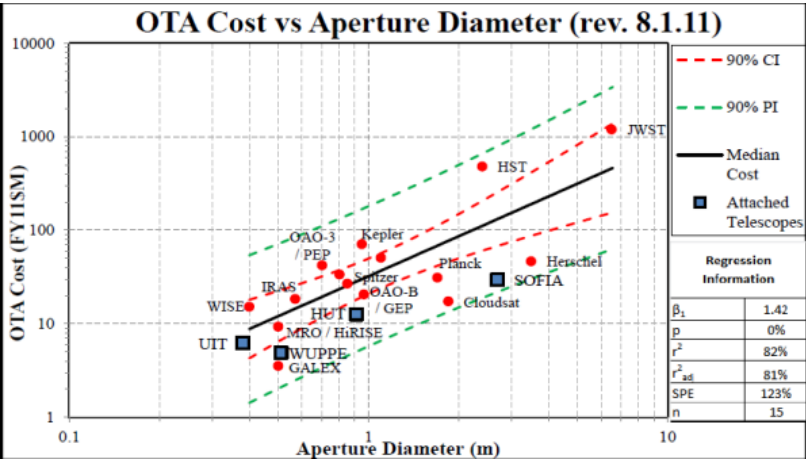


Fig. 2. Optical Telescope Assembly vs. cost correlation (Stahl *et al.*, 2013). Given a target OTA aperture diameter of 1.5 m for CDIM, a reasonable estimate of OTA cost is obtained from the median cost trendline.

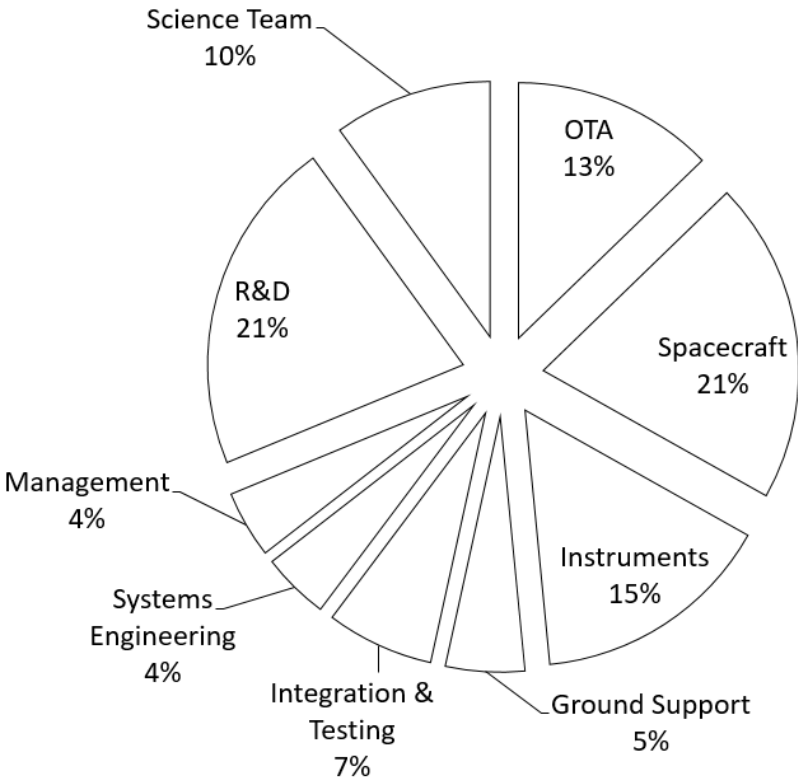


Fig. 3. CDIM estimated cost breakdown in percent of mission cost.

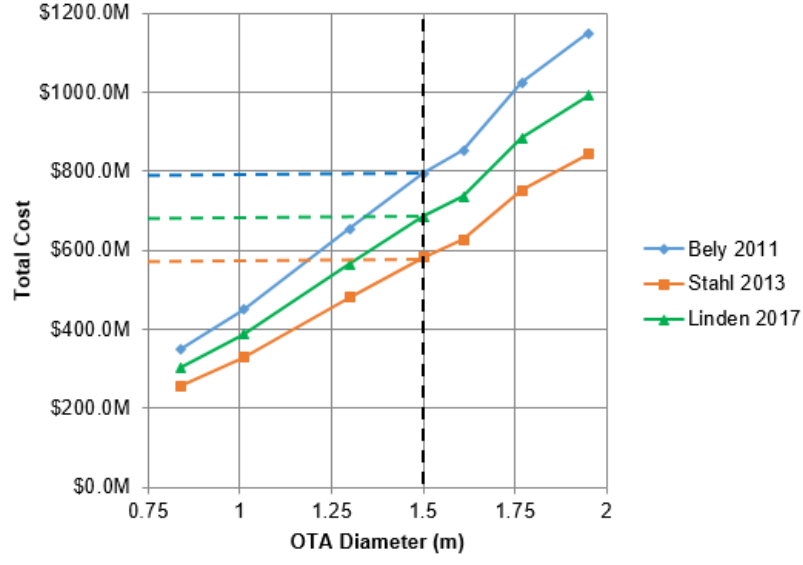


Fig. 4. CDIM total cost relation to OTA aperture diameter under various cost models. The model described here is robust and presents a conservative estimate compared to similar models by Bely and Stahl *et al.*, after a total cost approximation is applied following Equation 1.

## 11. Conclusion

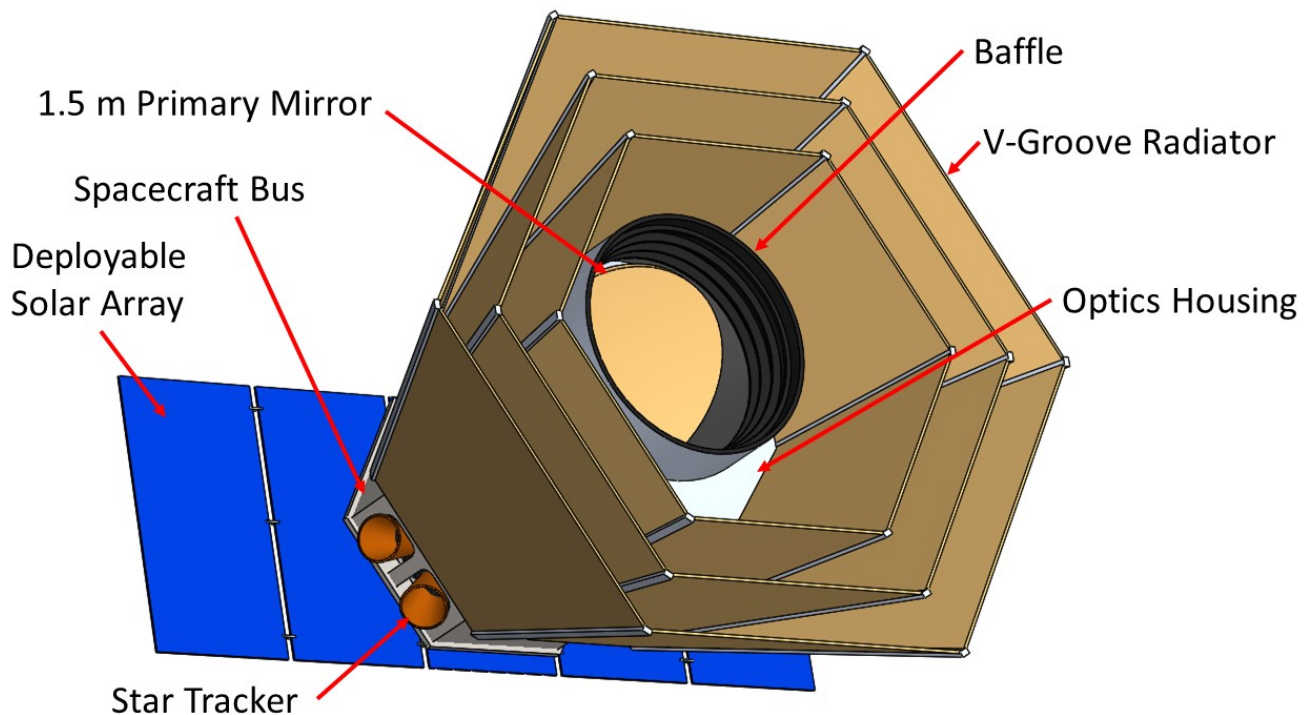


Fig. 5. CDIM consists of a passively cooled 1.5 m aperture OTA, actively cooled focal plane, and off-the-shelf spacecraft components where applicable. *Add cutaway view.*

## Acknowledgments

Thanks.

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