KiboCUBE Academy

Lecture 13

Introduction to CubeSat Thermal Control System

Hokkaido University

Division of Mechanical and Aerospace Engineering

Associate Professor Dr. Yuji Sakamoto

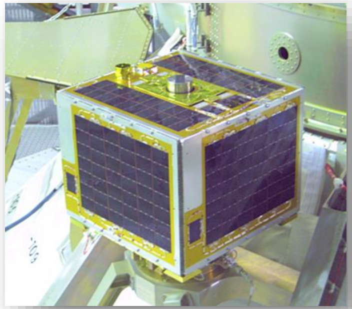
This lecture is NOT specifically about KiboCUBE and covers GENERAL engineering topics of space development and utilization for CubeSats.
The specific information and requirements for applying to KiboCUBE can be found at:
https://www.unoosa.org/oosa/en/ourwork/psa/hsti/kibocube.html

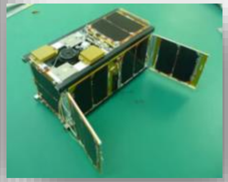
JAXA

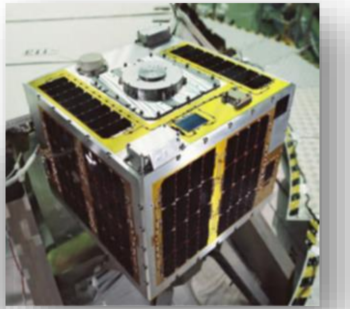
UNISEC JAPAN
University Space Engineering Consortium

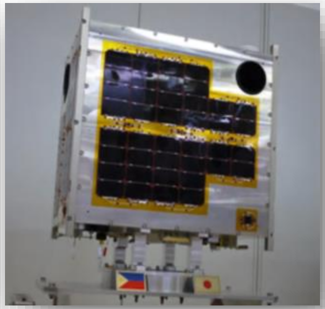
United Nations Office for Outer Space Affairs ★ Japan Aerospace Exploration Agency
Kibo CUBE

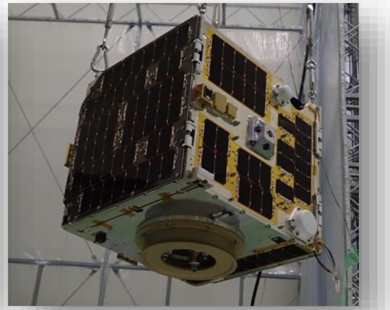
Lecturer Introduction











Yuji Sakamoto, Dr.

Position:

2006 - Assistant Professor (-2015), Associate Professor (2015-)
Department of Aerospace Engineering, Tohoku University

2021 - Associate Professor
Division of Mechanical and Space Engineering, Hokkaido University

Research Topics:

Design, Assembly, and Evaluation of Micro and Nano Satellites

Satellite Operation and Ground Station Management

Contents

1. Introduction
2. Theory
3. Case study I - 50kg microsatellites in SSO
4. Case study II - 2.6kg 2U CubeSat in ISS orbit
5. Conclusion

Tutorial. Example of 1-node analysis for 1U CubeSat

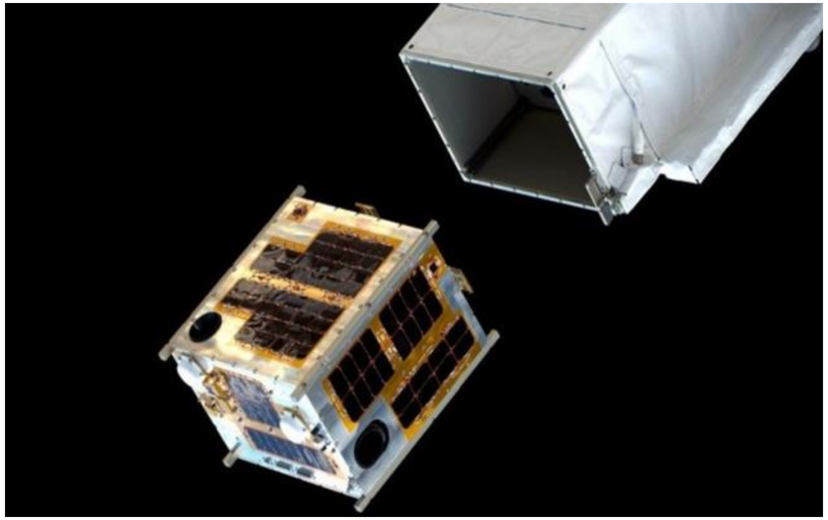
1. Introduction

1. Introduction

1.1 Lecturer's satellite projects

(C)JAXA

DIWATA-1
was released to space
on April 28, 2016
from the ISS

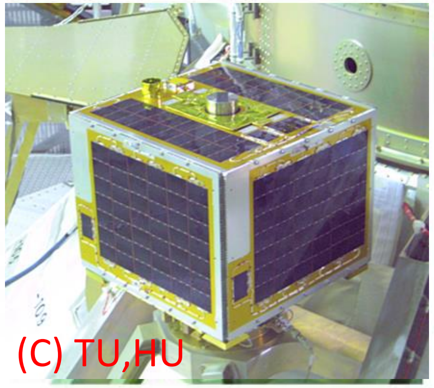


(C)JAXA

1. Introduction

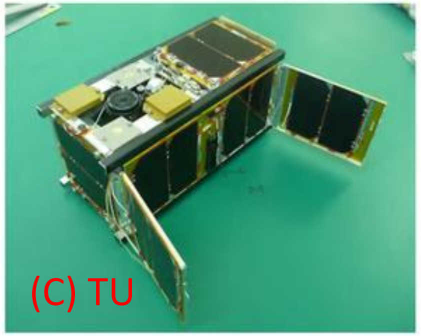
1.1 Lecturer's satellite projects

2009



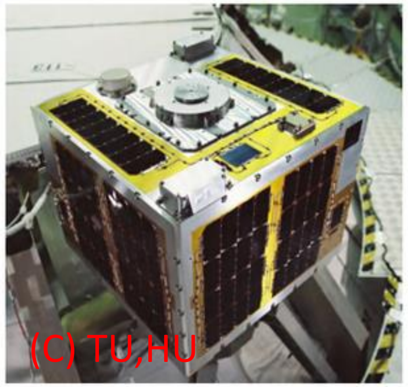
SPRITE-SAT

2012



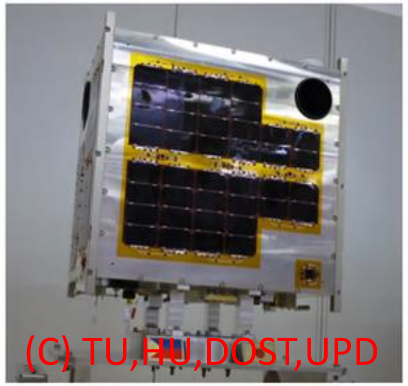
RAIKO

2014



RISING-2

2016



DIWATA-1

2018

DIWATA-2

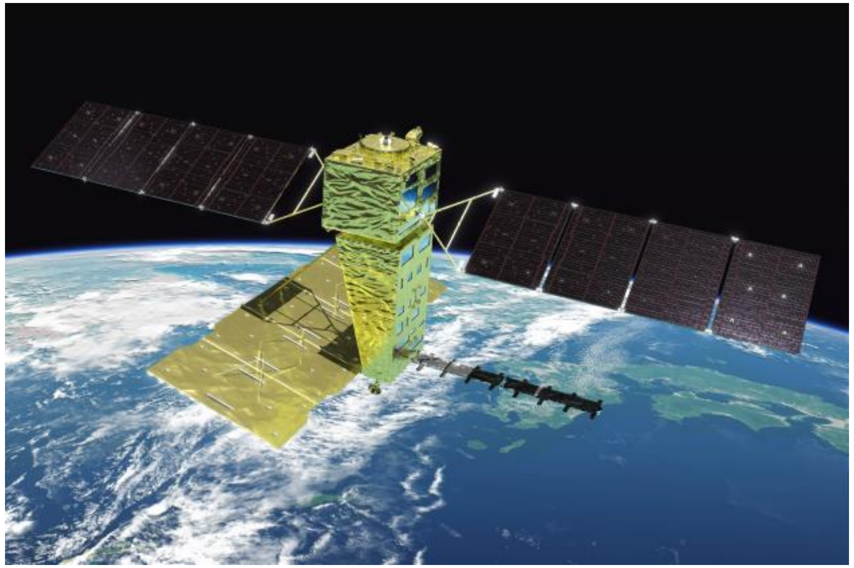
2019

RISESAT

TU = Tohoku University
HU = Hokkaido University
DOST = Department of Science and Technology, Philippines
UPD = University of the Philippines Diliman

1. Introduction

1.2 Bus system of micro/nano satellites



(C) JAXA, P100012872

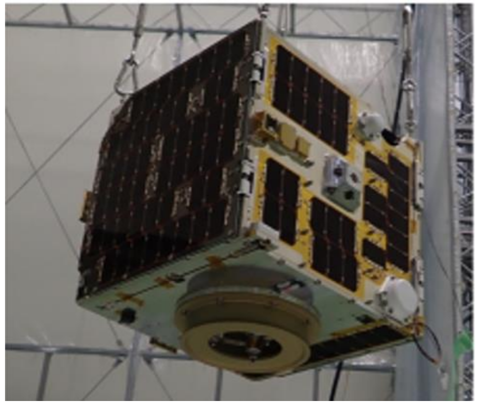
Large Satellites

ALOS-4

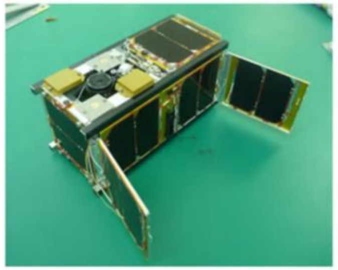
3000 kg

> MicroSatellite x 50

> NanoSatellite x 1000



(C) TU,HU,DOST,UPD



(C) TU

Micro Satellites

DIWATA-2

57 kg

Nano Satellites

RAIKO

2.7 kg

1. Introduction

1.2 Bus system of micro/nano satellites

Launch is *not a GOAL*, it is just a *START*

(C) TU,HU,DOST,UPD

(C)JAXA

(C)ESA

1.5 - 3 years

3 - 5 years

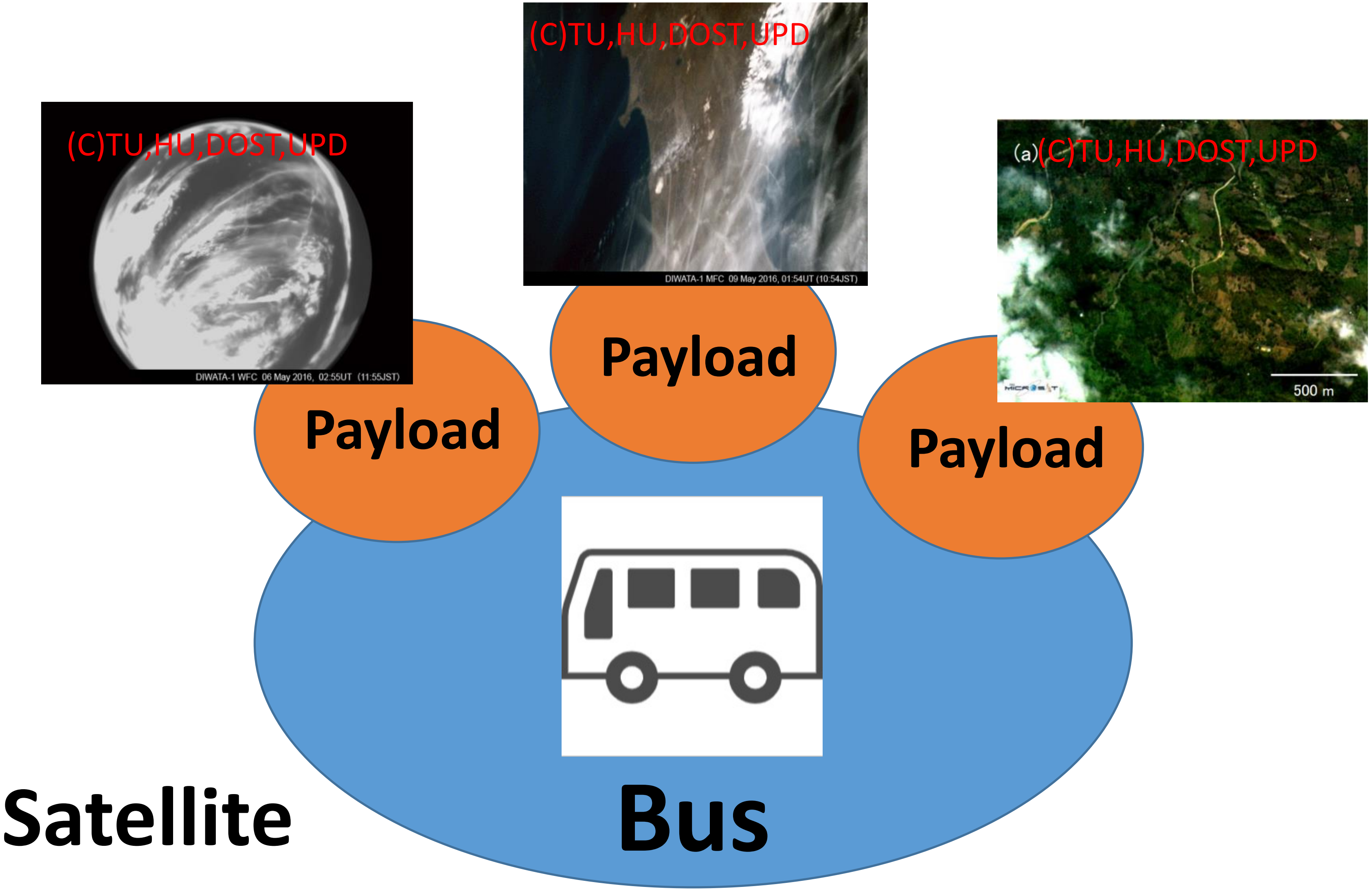
design, assemble

completion, launch

communications, observations

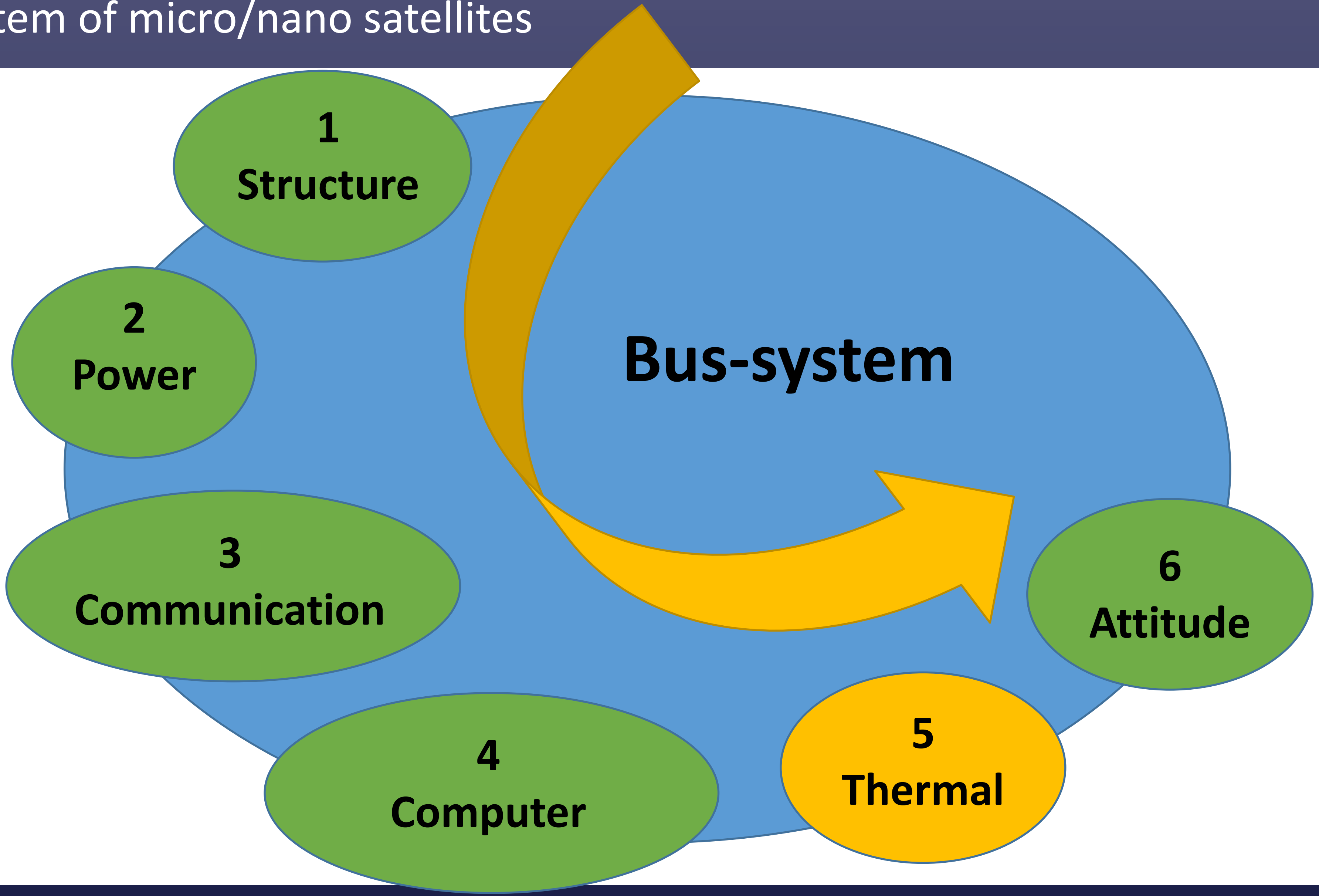
end of *operation*

1.2 Bus system of micro/nano satellites

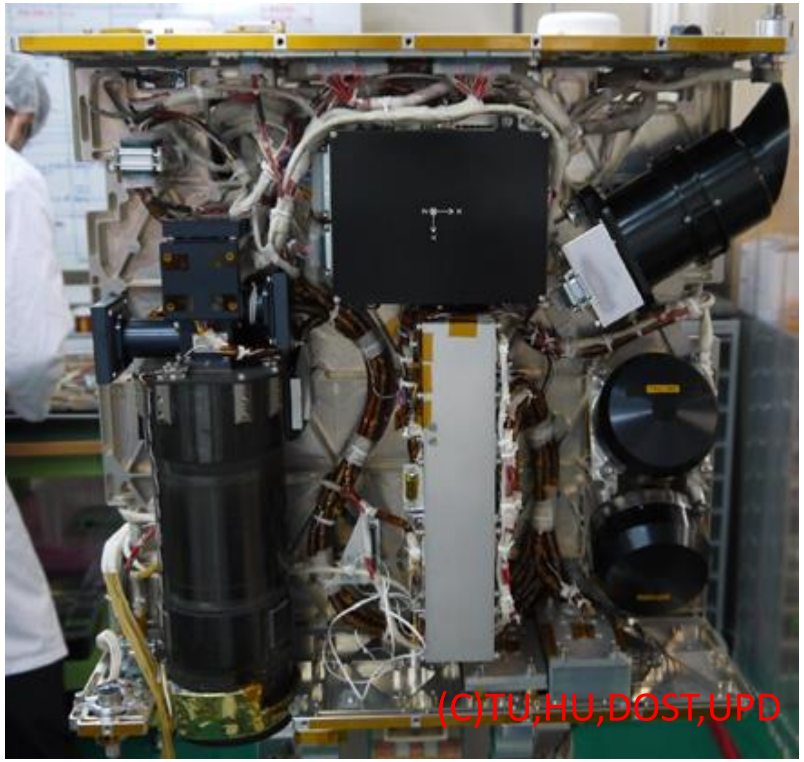


1. Introduction

1.2 Bus system of micro/nano satellites



1.2 Bus system of micro/nano satellites



550 x 350 x 550 mm

52.4 kg

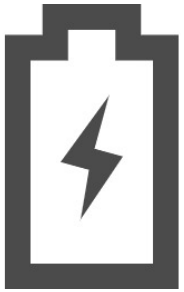
1. Introduction

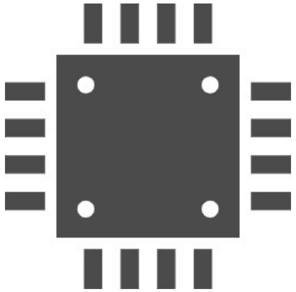
1.3 Importance to estimate and control the satellite temperature

- **If you don't care of the temperature, what will happen ?**
  - **Battery outside** of **temperature range**

    => no charge/discharge of battery current => **life is over, mission failure**
  - **Onboard computers or sensors** temperature out of range => **mission failure**
  - **Mechanical parts** temperature out of range => **mission failure**
- **Important thing = which items have narrow or severe temperature ranges ?**
  - We need to **estimate the temperature** in the preliminary design phase

    => affects the **design of the structure**, **locations of onboard instruments**, **locations of solar-cell area**, and **space for additional heat control items**

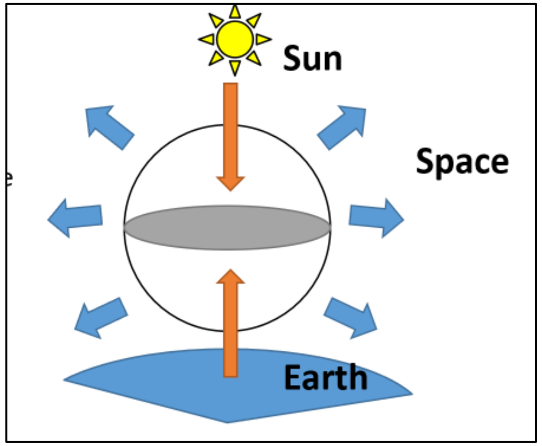




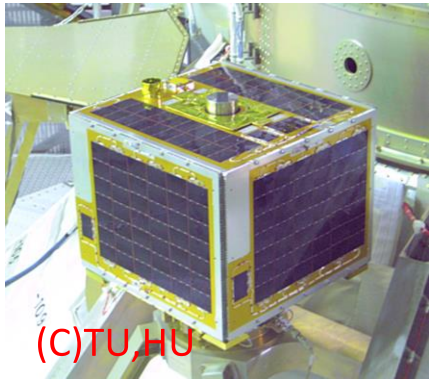
1. Introduction

1.4 Objectives of lecture

- **Fundamentals of thermal analysis and control**
- **Tutorial of 1-node simple thermal analysis for 1U CubeSat**
- **Case studies of analyses, tests, and flight data**
  - 1: microsatellite SPRITE-SAT (2009)
  - 2: nanosatellite RAIKO (2012)
  - 3: microsatellite RISING-2 (2014)

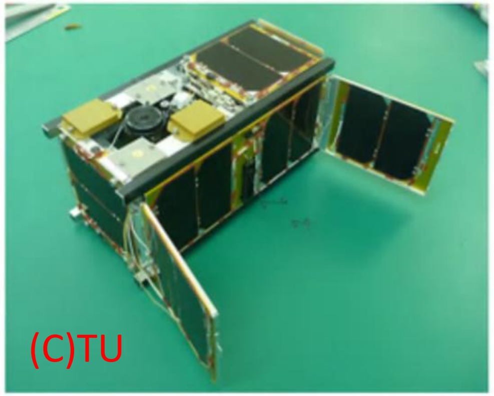


1-node analysis



SPRITE-SAT

RISING-2



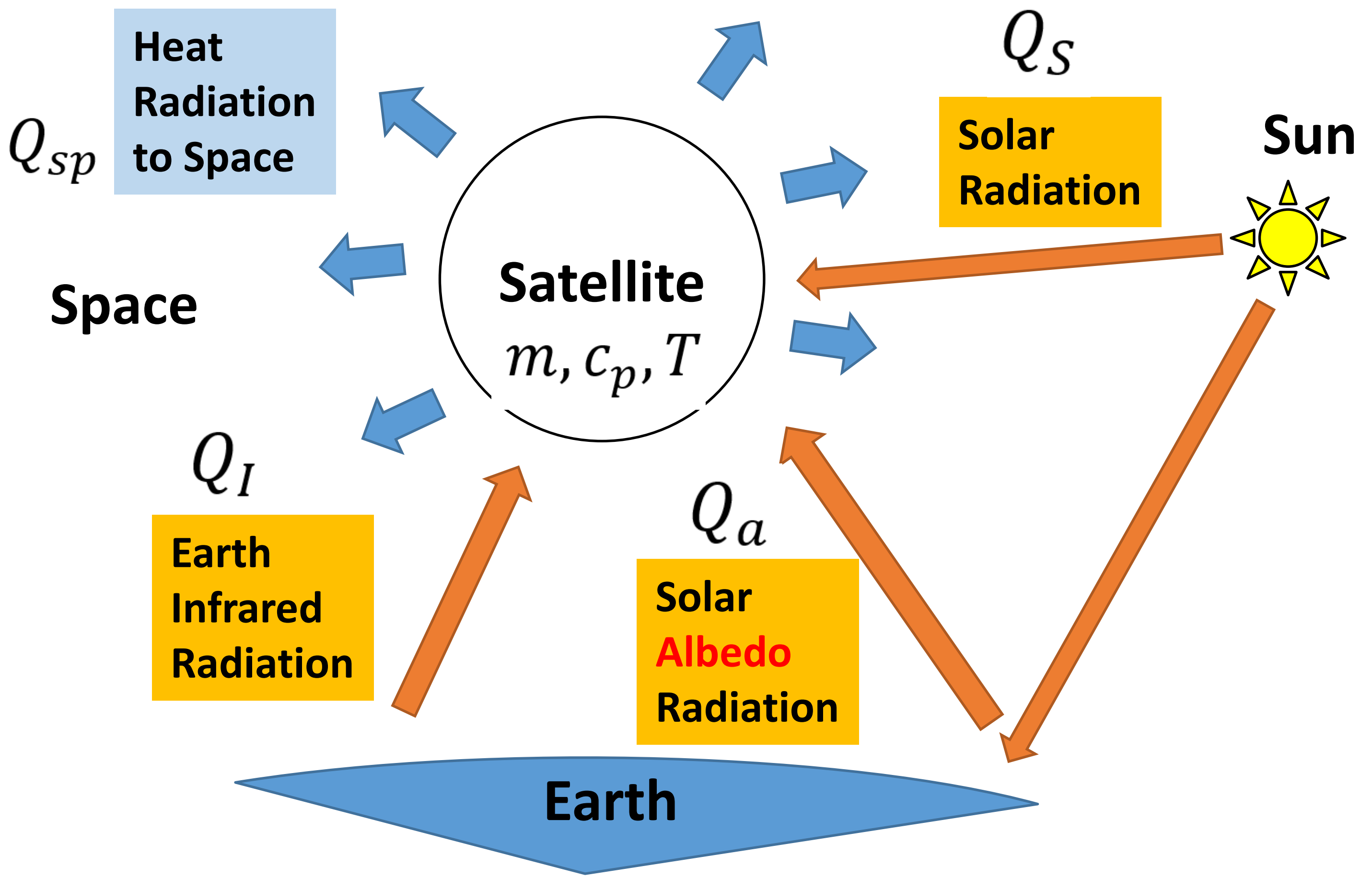
RAIKO

2. Theory

2. Theory

2.1 Elements to decide the temperature of satellite

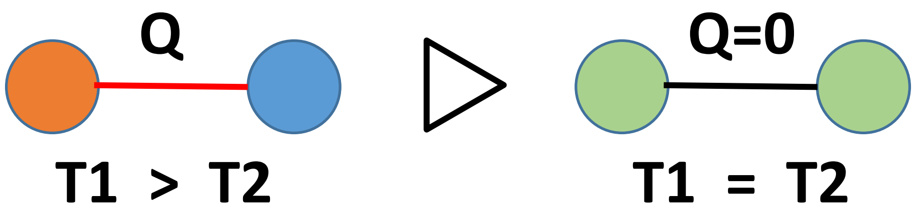
A. Satellite <==> Environment



Simplified ignoring inside of satellite

$$mc_p \frac{dT}{dt} = Q_S + Q_I + Q_a - Q_{sp}$$

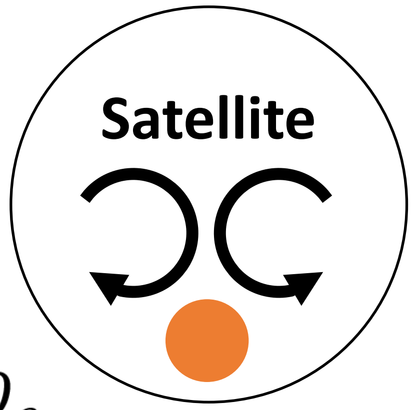
plus => T is **increased**
minus => T is **decreased**
zero => T is **not changed**



2. Theory

2.2 Math model of temperature variation by time

B. Inside of satellite



Q_e

Power Consumption

contact/radiation heat transfer (among of panels and instruments)

Full equation for item i

$$m_i c_{pi} \frac{dT_i}{dt} = Q_i - \sum_{j=1}^{n} C_{ij}\left(T_i - T_j\right) - \sum_{j=1}^{n} \varepsilon_i \varepsilon_j F_{ij} A_i \sigma \left(T_i^4 - T_j^4\right)$$

contact heat transfer

radiation heat transfer

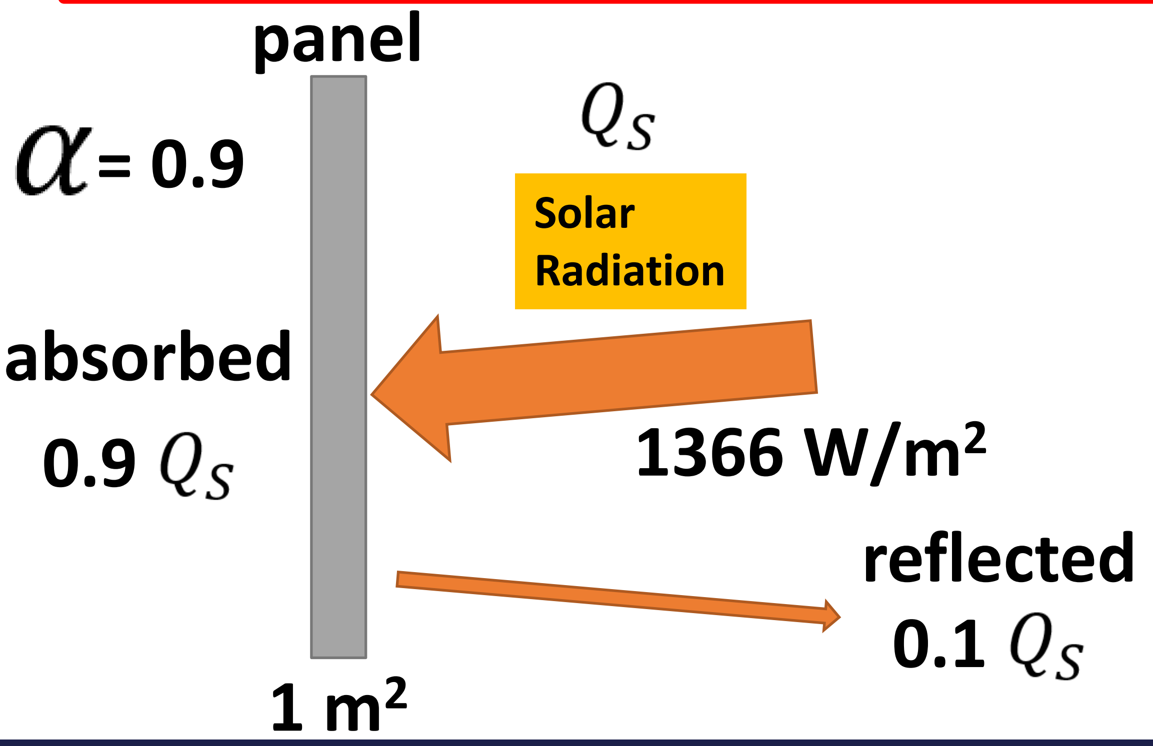
sum of **environmental heats** and **power consumption**

2. Theory

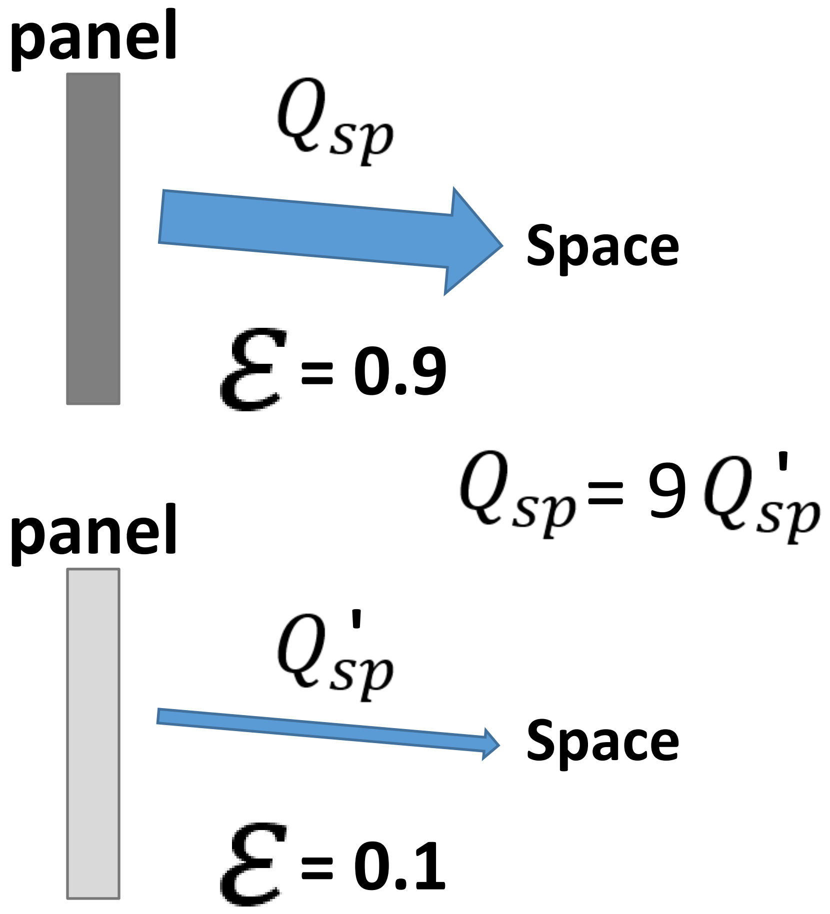
2.2 Math model of temperature variation by time

α solar absorptivity = 0 .. 1

ε infrared emissivity = 0 .. 1



example of radiation to space



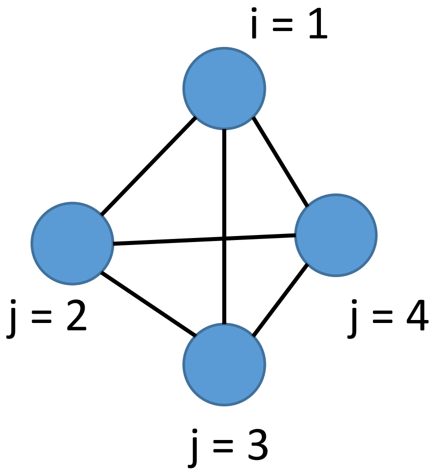
2. Theory

2.2 Math model of temperature variation by time

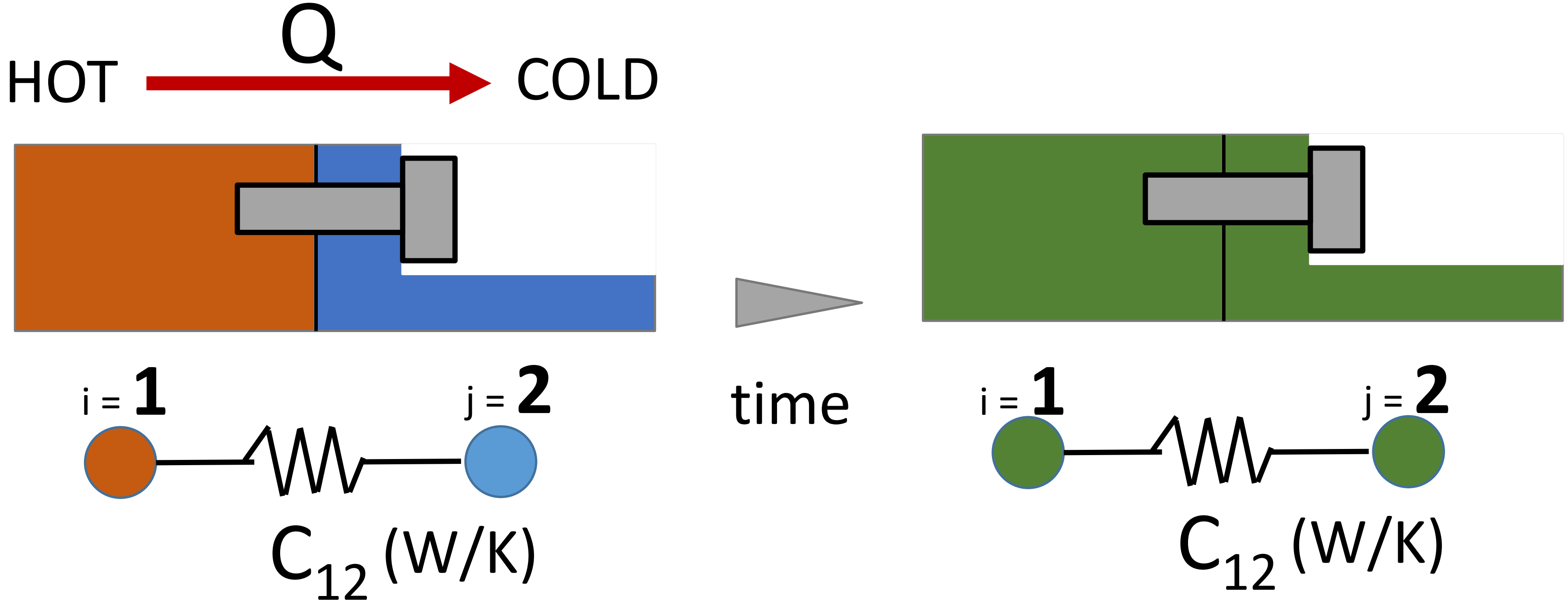
$$-\sum_{j=1}^{n} C_{ij}\left(T_i - T_j\right)$$

contact

heat transfer



simplified to ***2 blocks***



contact thermal resistance

$$Q_{12} = -C_{12}(T_1 - T_2)$$

$$T_1 = T_2$$

$$Q_{12} = 0$$

2. Theory

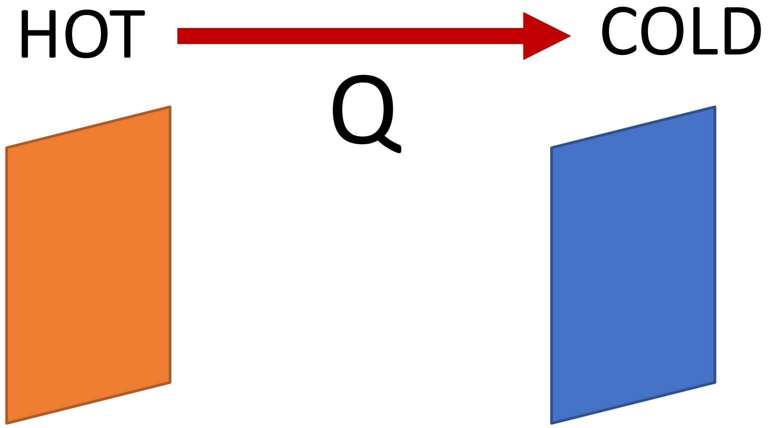
2.2 Math model of temperature variation by time

$$-\sum_{j=1}^{n} \varepsilon_i \varepsilon_j F_{ij} A_i \sigma \left(T_i^4 - T_j^4\right)$$

radiation

heat transfer

simplified to ***2 plates***



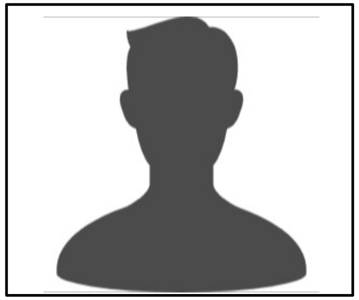
$$Q_{12} = -\varepsilon_1 \varepsilon_2 F_{12} A_1 \sigma (T_1^4 - T_2^4)$$

F view factor = 0 .. 1

near

= large F

= **high** radiation

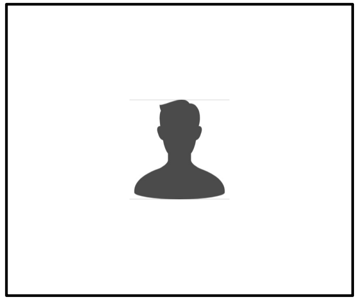


0.6 in view

far

= small F

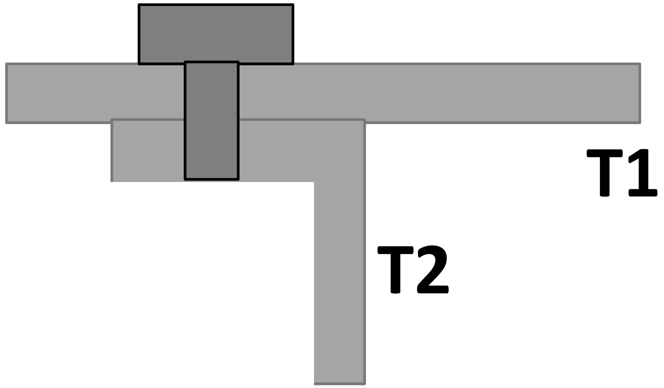
= **low** radiation

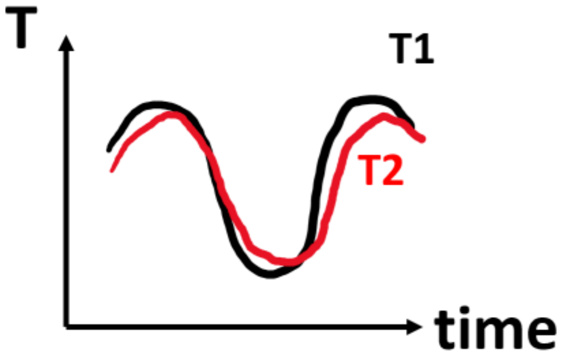


0.1 in view

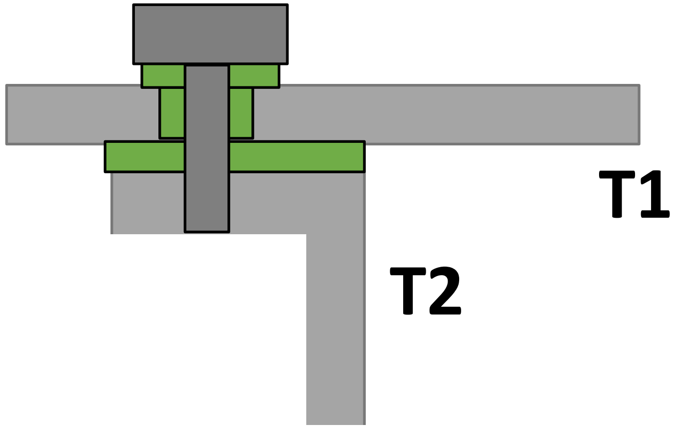
2.3 Effect of panel connections

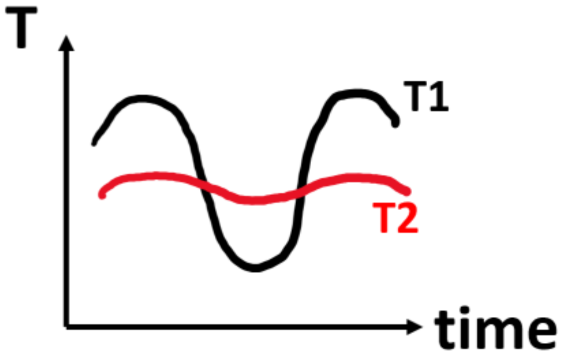
A) direct connection
= **high** contact resistance



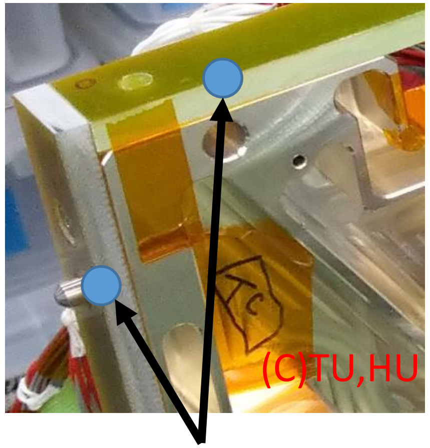


B) insert insulation plate and washer
= **low** contact resistance





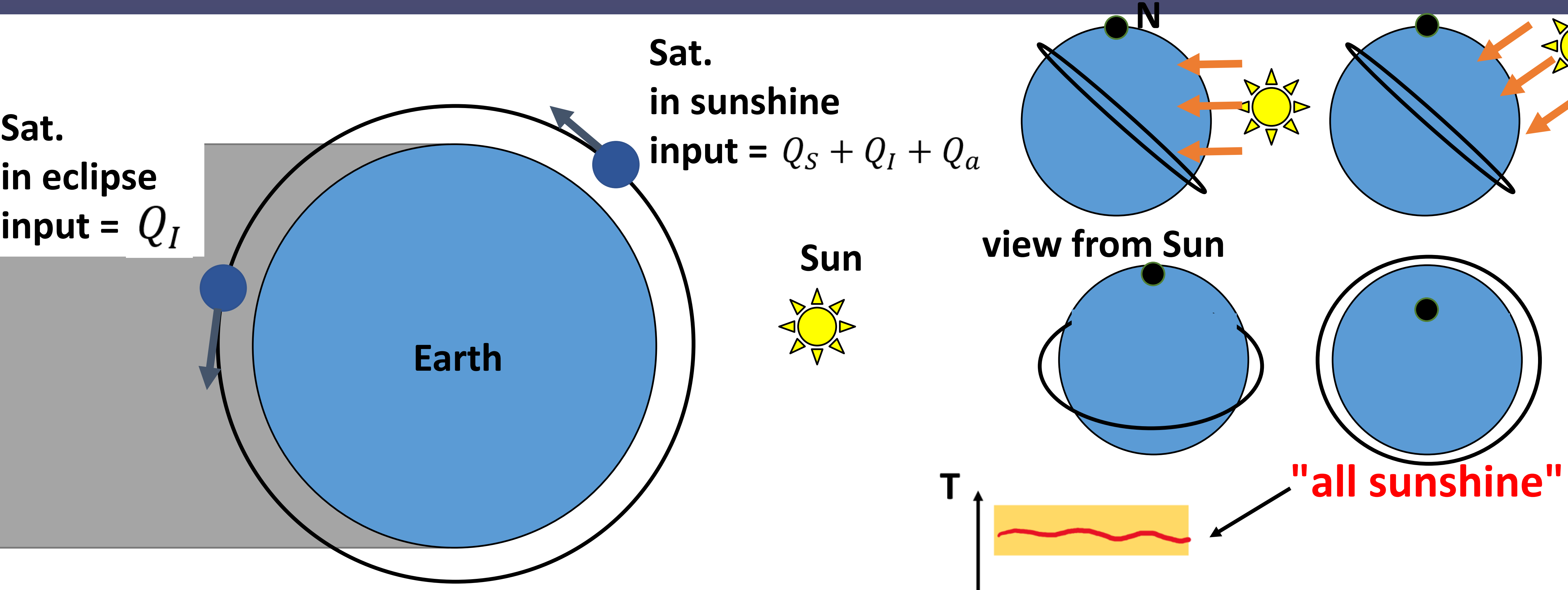
glass epoxy washer



glass epoxy plates

2. Theory

2.5 Effect of orbital motion

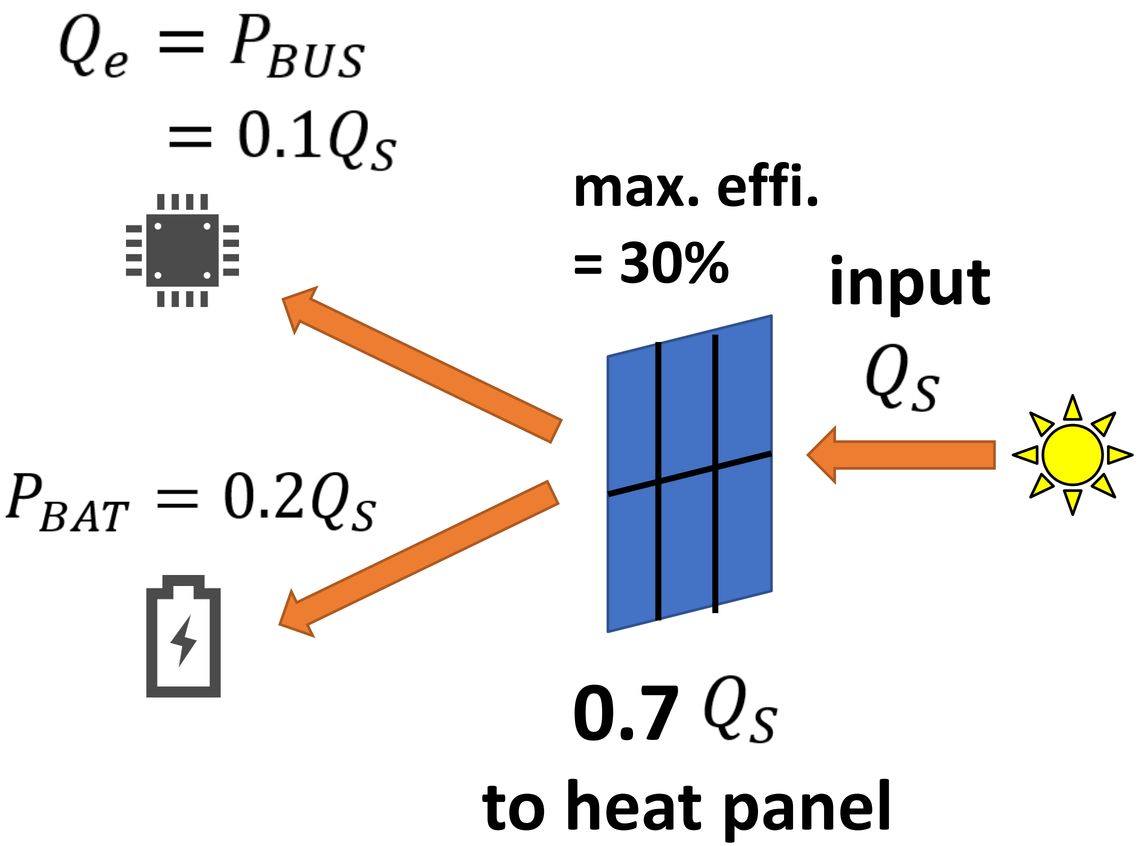


90-100 minutes/round

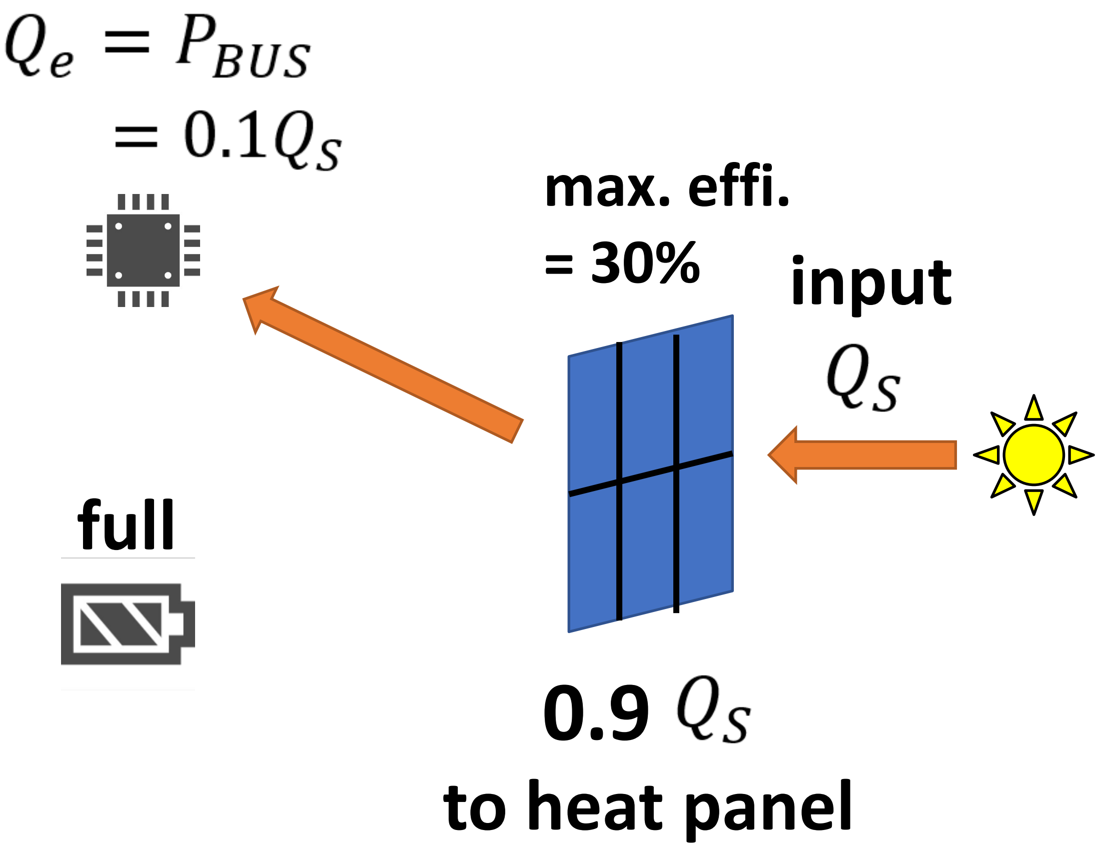
14 .. 15 rounds/day

2.7 Effect of power generation and consumption

A) charging battery (sunshine)

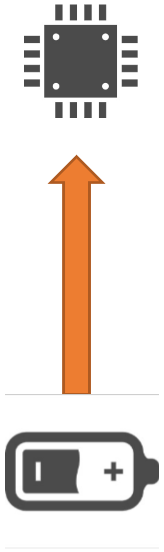


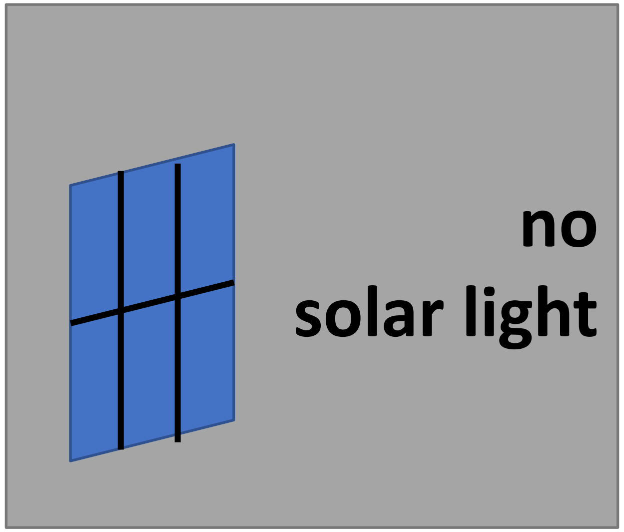
B) battery full (sunshine)



C) discharging battery (eclipse)

$$Q_e = P_{BUS}$$





- input heats Q_S to panel

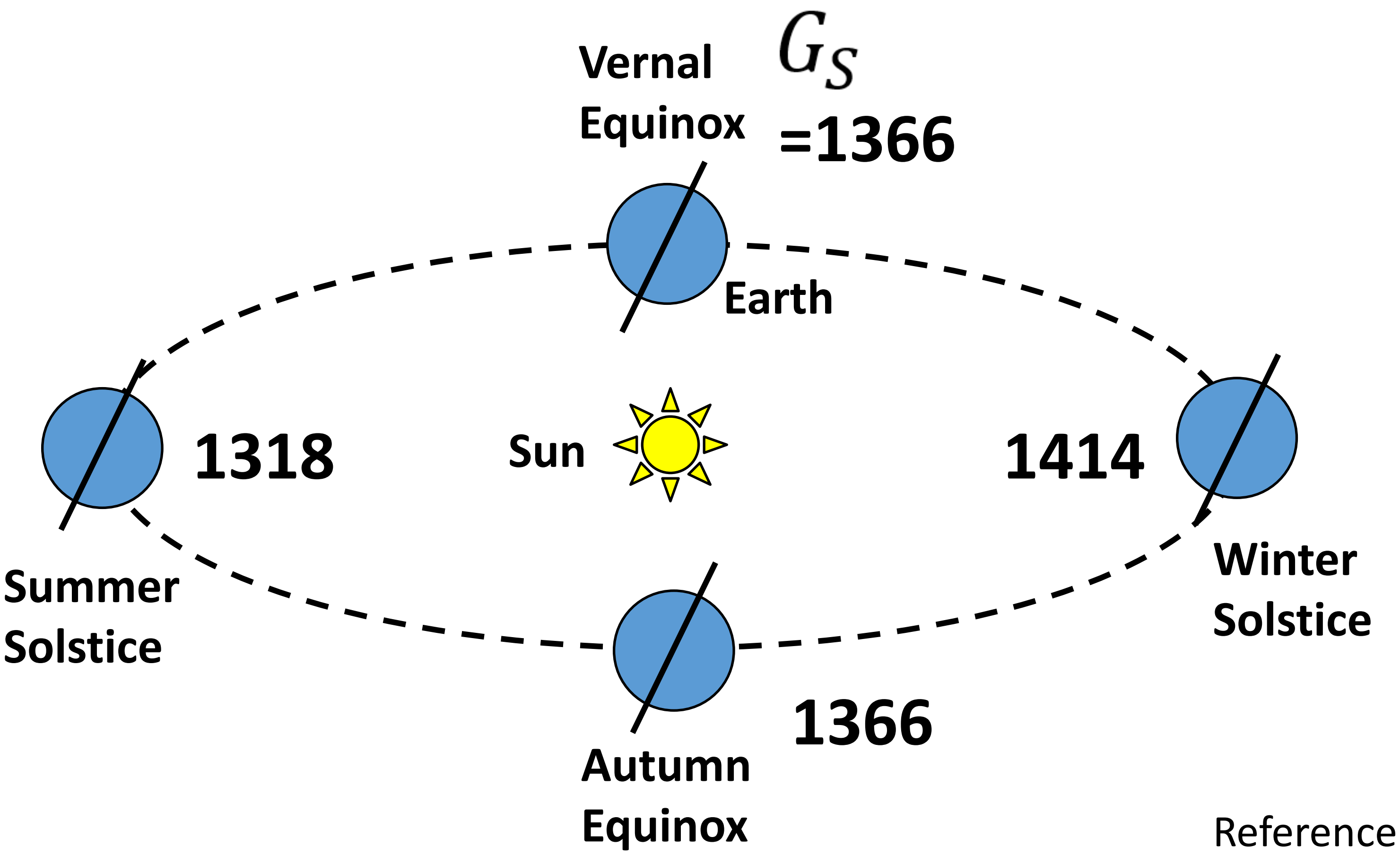
=> **partly**, consumed by bus instruments (sunshine)

=> **partly**, used to charge battery (sunshine) **[!] no heat**

=> consumed by bus instruments (eclipse)

2. Theory

2.8 Effect of 4 seasons



	G_S W/m²	q_I W/m²	a
MAX	1414	261	0.4
AVG	1366	237	0.3
MIN	1318	189	0.2

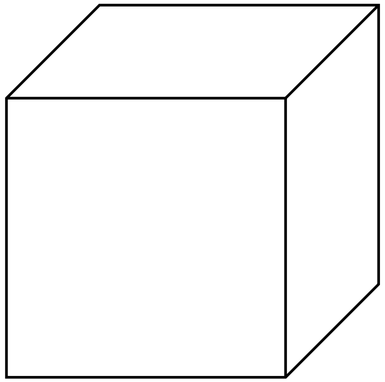
Reference:
JAXA Design Standard : Spacecraft Thermal Control System,
https://sma.jaxa.jp/en/TechDoc/index.html

Tutorial

Example of 1-node analysis for 1U CubeSat

Tutorial

Example of 1-node analysis for 1U CubeSat

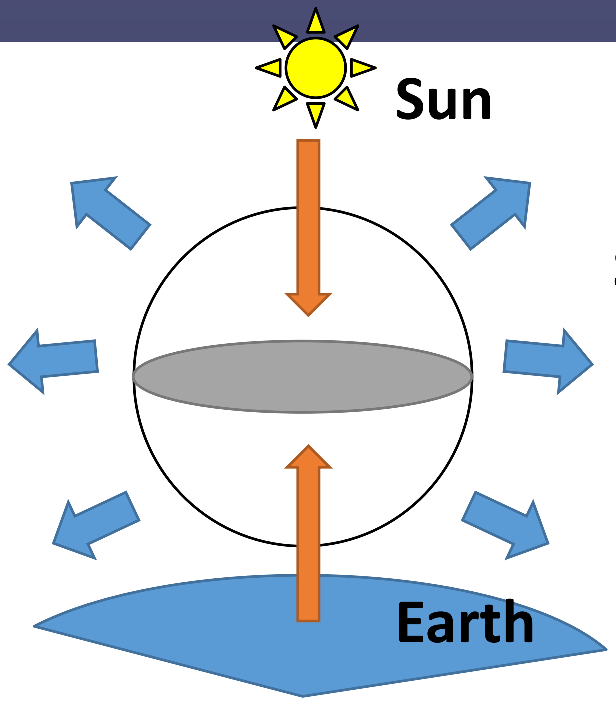


10cm (each side)
1U CubeSat

surface area =
0.010^2 x 6 = **0.060** m^2

complex
cross section area

simplified
by **sphere**
with same-surface
area



surface area = **0.060** m^2
cross section area
= surface area / 4 = **0.015** m^2

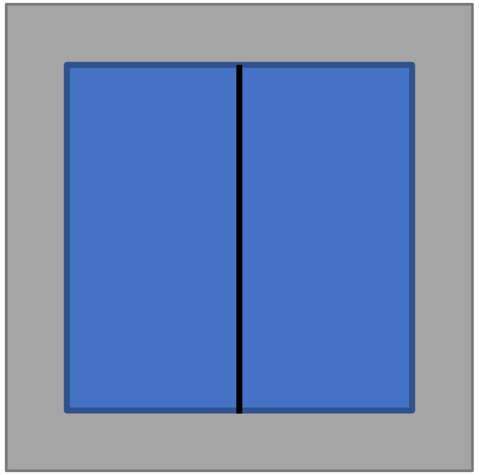
constant
cross section area

Tutorial

Example of 1-node analysis for 1U CubeSat

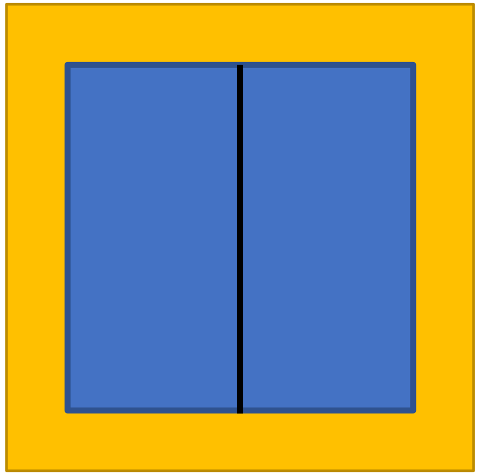
Step 1: Define the alpha and epsilon of the surface

Case A)
solar cells (8x7cm) + aluminum



Case A	occupancy	alpha	epsilon
solar cells	56%	0.920	0.800
aluminum	44%	0.255	0.025
polyimide	0%	0.515	0.760
	avg.	**0.627**	**0.459**

Case B)
solar cells (8x7cm) + polyimide



Case A	occupancy	alpha	epsilon
solar cells	56%	0.920	0.800
aluminum	0%	0.255	0.025
polyimide	44%	0.515	0.760
	avg.	**0.742**	**0.782**

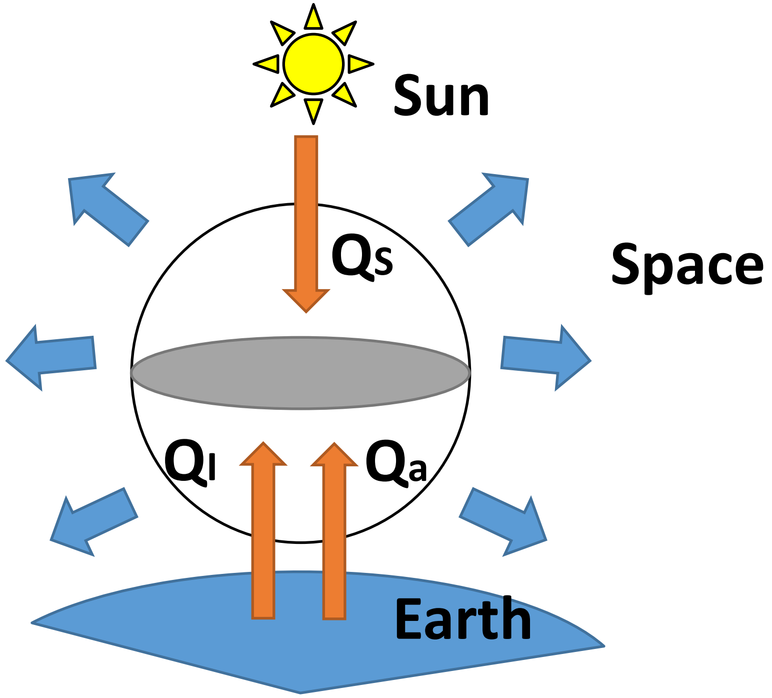
Example of 1-node analysis for 1U CubeSat

Step 2: Define the input heats from Sun and Earth

$$Q_S = AG_s\alpha\cos\theta_s,\quad Q_I = Aq_I\varepsilon F_e\cos\theta_e,$$

$$Q_A = AG_s a K_a \alpha F_e\cos\theta_e$$

* $A\cos\theta_s = A\cos\theta_e = A_{cross}$ in this sphere model



G_s direct solar flux	W/m^2	1366
q_I earth infrared emission	W/m^2	237
a albedo rate		0.3
Fe@400km alt. view factor of earth		0.885
Ka@400km alt. for Qa calc.		0.998

		Case A	Case B
alpha_avg		**0.627**	**0.742**
epsilon_avg		**0.459**	**0.782**
Q_s	W	12.86	15.20
Q_I	W	**1.44**	**2.46**
Q_a	W	3.41	4.03
$Q_s+Q_I+Q_a$	W	**17.71**	**21.69**

Example of 1-node analysis for 1U CubeSat

Step 3: Calculate temperature time rate of change (dT/dt) as example

$$mc_p \frac{dT}{dt} = Q_S + Q_I + Q_a + Q_e - Q_{sp}$$

$Q_S + Q_I + Q_a$ = **Q**

Q_e = **0** (in this example)

heat radiation to space

[!] A = $A_{surface}$

$$Q_{sp} = \varepsilon A \sigma \left(T^4 - T_{sp}^4\right)$$

[!] **T** is satellite temperature

other constant values

σ Stephan-Boltzmann constant	$W/m^2.K^4$	5.671E-08
m satellite mass	kg	1.33
c_p specific heat of aluminum	J/kg.K	879
T_{sp} temperature of deep space	K	4

when T = 20 degC

Case A

Sun/Ecl	T(degC)	T(K)	Q(W)	Qsp(W)	dT/dt(K/s)
Sun	20	293	17.71	11.53	0.00528
Ecl	20	293	1.44	11.53	-0.00863

[!] Try to calculate by spread sheet

Example of 1-node analysis for 1U CubeSat

Sunshine ... Eclipse (**60** minutes)

* integrated by every **10** seconds

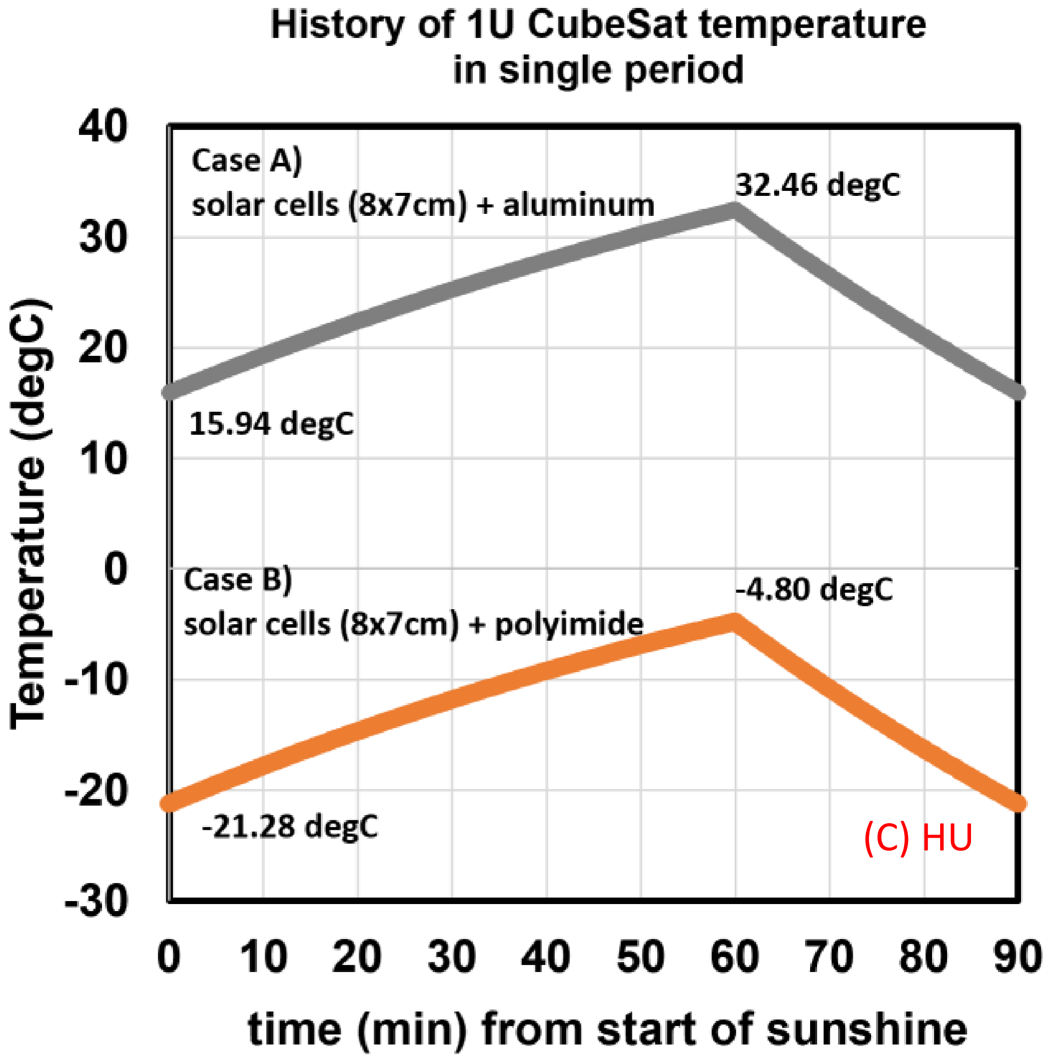
time(s)	time (min)	Sun/Ecl	T(degC)	T(K)	Q(W)	Qsp(W)	dT/dt(K/s)
0	0.0	Sun	**15.94**	289.09	17.71	10.91	0.00581
10	0.2	Sun	16.00	289.15	17.71	10.92	0.00581
20	0.3	Sun	16.06	289.21	17.71	10.93	0.00580
30	0.5	Sun	16.11	289.26	17.71	10.93	0.00579
40	0.7	Sun	16.17	289.32	17.71	10.94	0.00578
50	0.8	Sun	16.23	289.38	17.71	10.95	0.00578
3540	59.0	Sun	32.25	305.40	17.71	13.59	0.00352
3550	59.2	Sun	32.29	305.44	17.71	13.59	0.00352
3560	59.3	Sun	32.32	305.47	17.71	13.60	0.00351
3570	59.5	Sun	32.36	305.51	17.71	13.61	0.00351
3580	59.7	Sun	32.39	305.54	17.71	13.61	0.00350
3590	59.8	Sun	32.43	305.58	17.71	13.62	0.00350
3600	60.0	Ecl	32.46	305.61	1.44	13.62	-0.01042

Eclipse .. Sunshine (**30** minutes)

3600	60.0	Ecl	32.46	305.61	1.44	13.62	-0.01042
3610	60.2	Ecl	32.36	305.51	1.44	13.61	-0.01040
3620	60.3	Ecl	32.26	305.41	1.44	13.59	-0.01039
3630	60.5	Ecl	32.15	305.30	1.44	13.57	-0.01037
3640	60.7	Ecl	32.05	305.20	1.44	13.55	-0.01036
3650	60.8	Ecl	31.94	305.09	1.44	13.53	-0.01034
5340	89.0	Ecl	16.42	289.57	1.44	10.98	-0.00816
5350	89.2	Ecl	16.34	289.49	1.44	10.97	-0.00815
5360	89.3	Ecl	16.26	289.41	1.44	10.96	-0.00814
5370	89.5	Ecl	16.18	289.33	1.44	10.94	-0.00813
5380	89.7	Ecl	16.10	289.25	1.44	10.93	-0.00812
5390	89.8	Ecl	16.02	289.17	1.44	10.92	-0.00811
5400	90.0	Sun	**15.94**	289.09	17.71	10.91	0.00582
time(s)	time (min)	Sun/Ecl	T(degC)	T(K)	Q(W)	Qsp(W)	dT/dt(K/s)

[!] **find** the satellite **temperature**, which can be **same** at **beginning** and **end** of a single orbital period

Example of 1-node analysis for 1U CubeSat



- The only difference is the external surface materials, but the temperature can be shifted by **37** degC (= 16 - (-21)).
- without an insulation concept, satellite temperature can be changed by **16** degC (= 32 - 16) in a single 90-minute period

3. Case study I -
50kg microsatellites for SSO

3. Case study I - 50kg microsatellites for SSO

3.1 Summary of satellite specifications

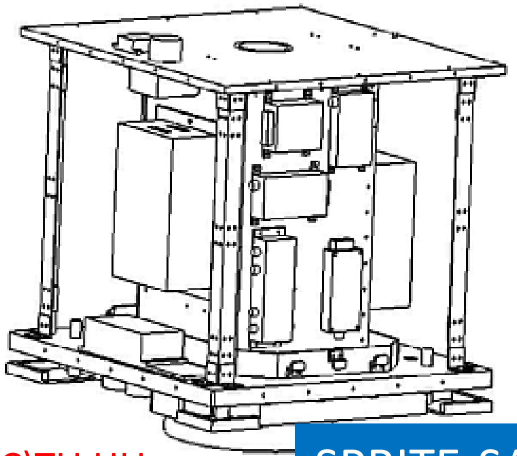
SPRITE-SAT

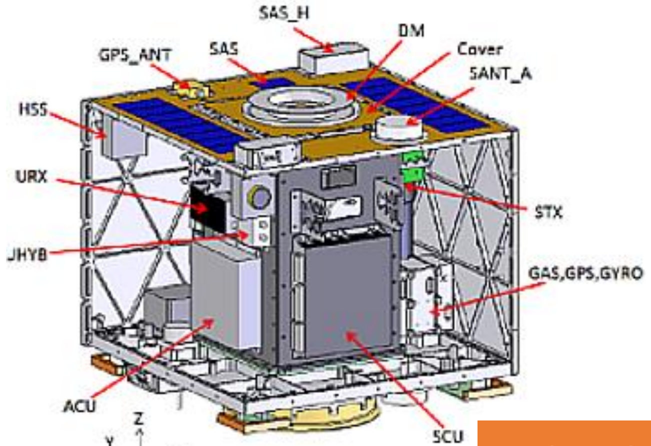
RISING-2

	SPRITE-SAT	**RISING-2**
Missions	Remote Sensing	Remote Sensing
Orbit	SSO, **666 km** height	SSO, **628 km** height
Mass	**45.3 kg**	**43.2 kg**
Size	50 x 50 x 50 cm (total) * 50 x 50 x 42 cm (panels)	50 x 50 x 50 cm (total) * 50 x 50 x 42 cm (panels)

3. Case study I - 50kg microsatellites for SSO

3.2 Outside/Inside appearance


(C)TU,HU
SPRITE-SAT



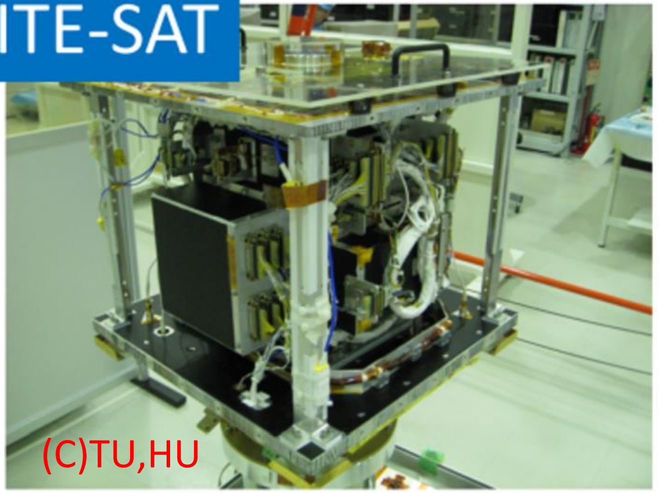
(C)TU,HU
RISING-2

	SPRITE-SAT	RISING-2
Common	**6 external** panels + **center pillar** (4 panels) * **most of instruments** are attached on the **center pillar** * center pillar is **thermally insulated** from external panels	
External panel material	aluminum skin/core **honeycomb** panels	**aluminum grid** panels * **fabrication speed** and **cost** merits * heavier than SPRITE-SAT

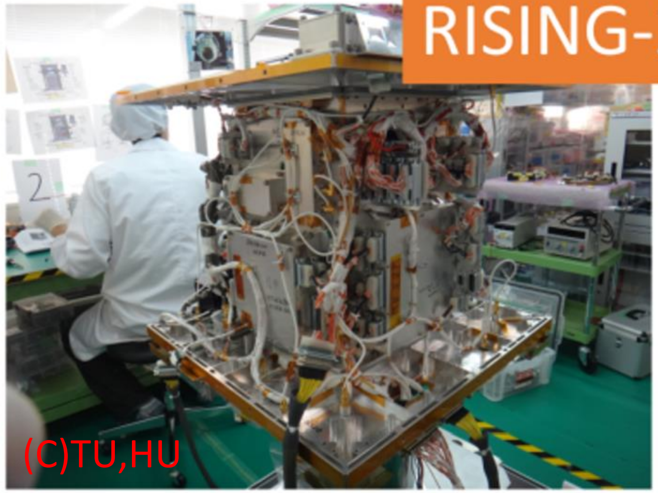
3. Case study I - 50kg microsatellites for SSO

3.3 Concept of inside heat connection model

SPRITE-SAT



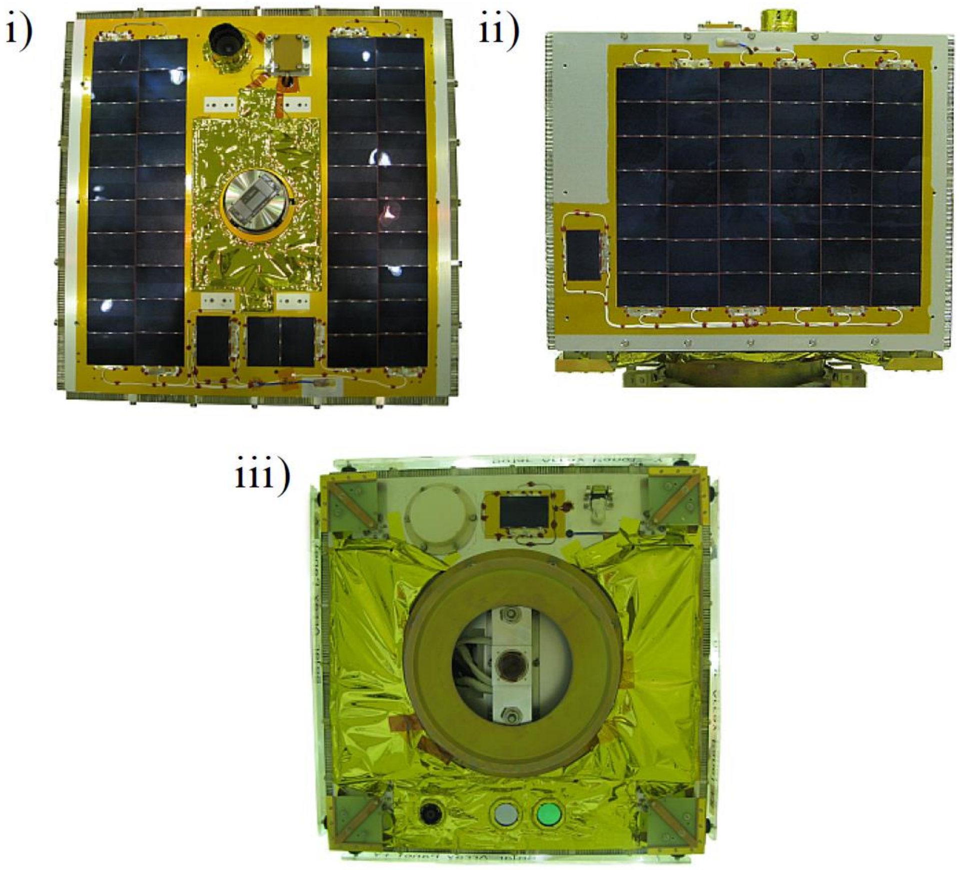
RISING-2



	SPRITE-SAT	RISING-2
Surface of internal panels and instruments	**Black**	Aluminum (**no paints**)
Heat radiation transfer with ext. panels	**Large** * complex thermal analysis * negative for center insulation concept	**Small** (or **Negligible**)

3.4 Concept of outside heat connection model

Case of SPRITE-SAT



(C)TU,HU

Surface of panels

(i: top panel, ii: side panels, iii: bottom panel)

Table Heat specifications of elements and panels

element	α	ε
solar cell*	0.68	0.80
aluminum **	0.26	0.03
MLI	0.05	0.05
Polyimide	0.52	0.76

panel	$A(\mathrm{m}^2)$	α	ε
top	0.2397	0.463	0.619
side	0.1862	0.517	0.652
bottom	0.2397	0.118	0.078

*power generation efficienty = 0.17

*** original value was 0.85 w/o power generation**

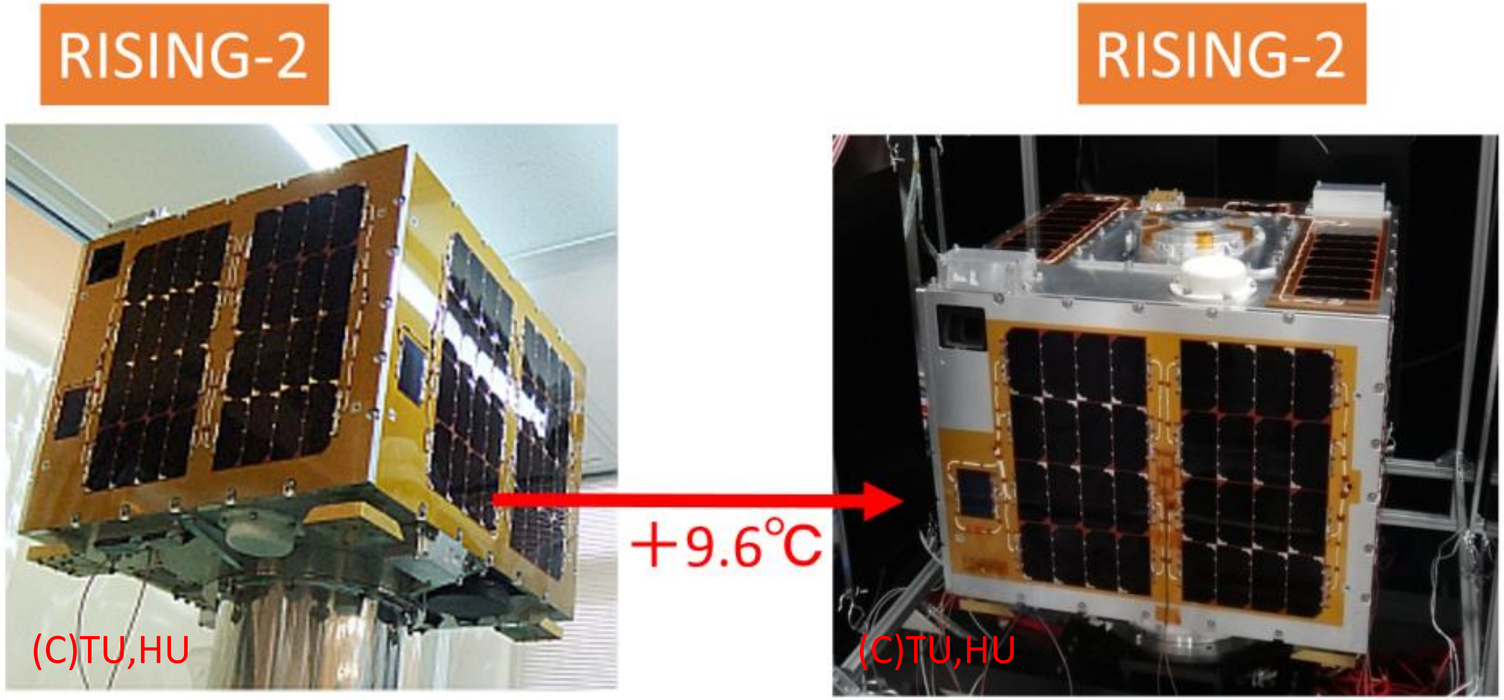
** A7075 with Alodine 1000

[!] Area, avg. alpha and epsilon values are summarized in each panel

Y. Sakamoto, et al., "Pre-Flight Analysis, Test Evaluation and Flight Verification of the Thermal System of Tohoku University SPRITE-SAT," TRANSACTIONS OF THE JAPAN SOCIETY FOR AERONAUTICAL AND SPACE SCIENCES, AEROSPACE TECHNOLOGY JAPAN, 2010 Volume 8 Issue ists27 Pages Tf_1-Tf_6

3. Case study I - 50kg microsatellites for SSO

3.4 Concept of outside heat connection model



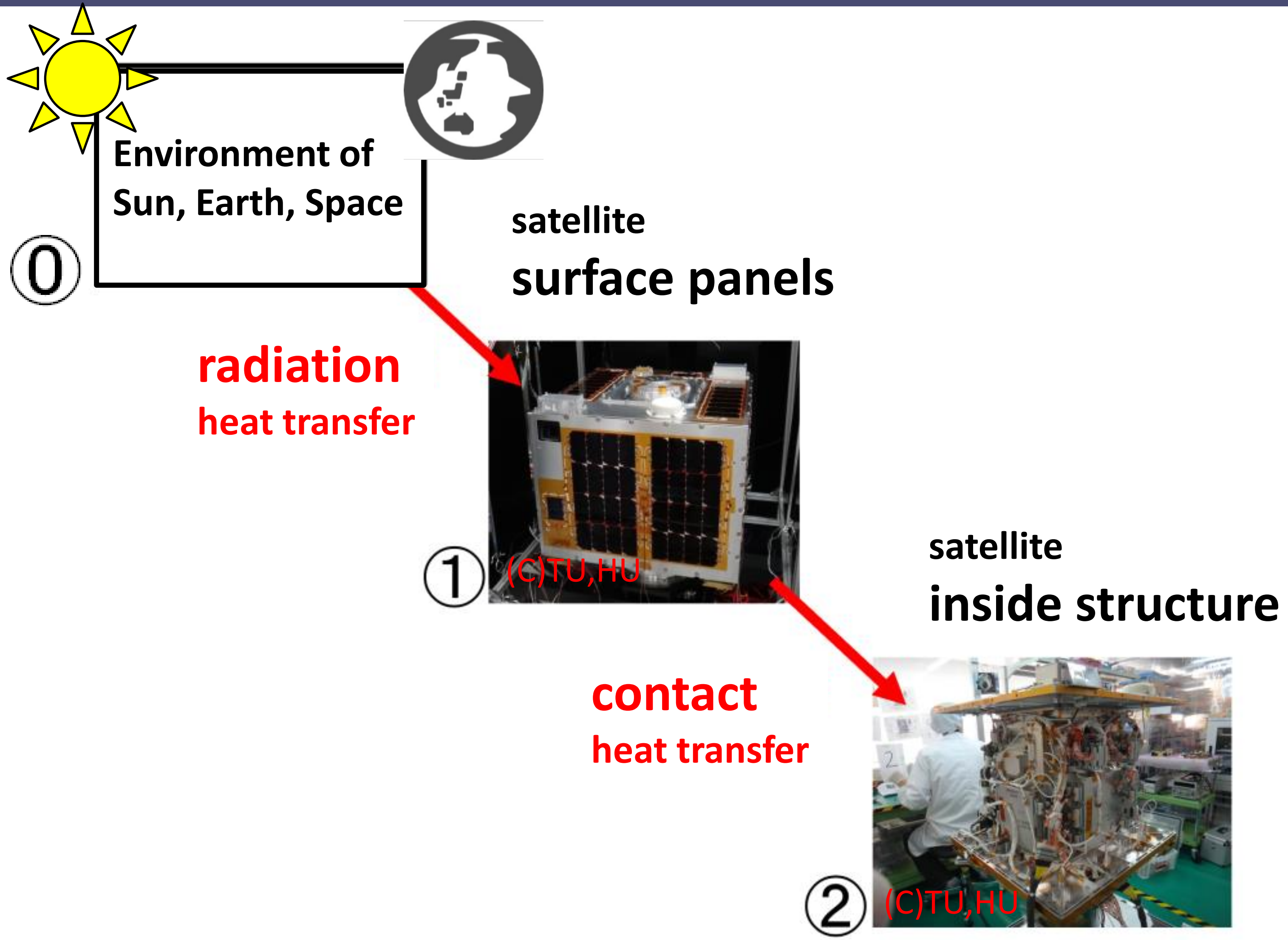
Adjustment of Polyimide area is an **easy method** to control **heat balance**

- average temperature can be **decreased** by a **larger** Polyimide area
- Polyimide area (0.76 epsilon) can emit the larger heat from aluminum surface (0.03 epsilon)

[!] Exact alpha, epsilon values are **different depending on the coating method or tape thickness**, they must be measured by special instruments

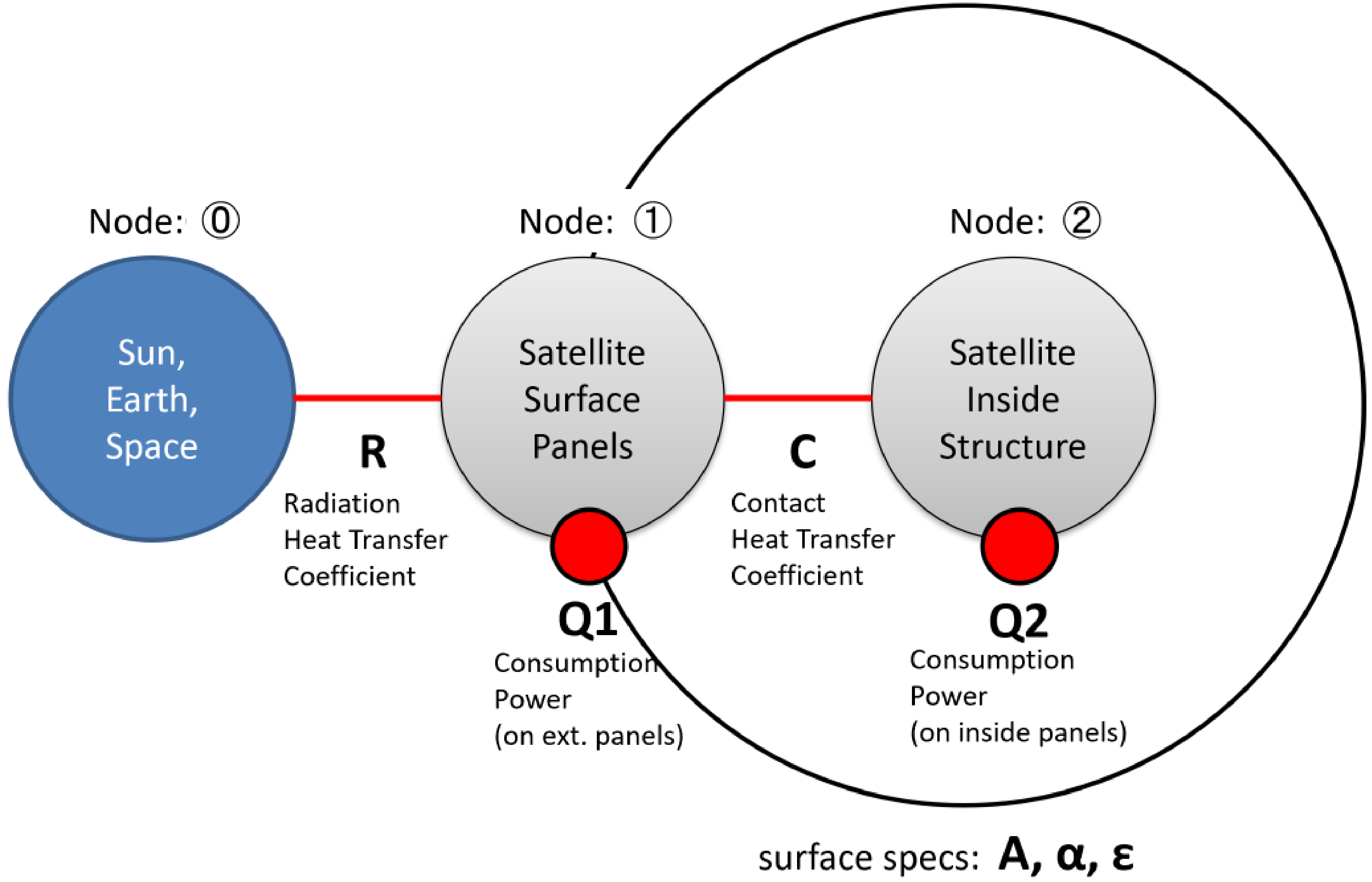
3. Case study I - 50kg microsatellites for SSO

3.5 Method of 2-node analysis

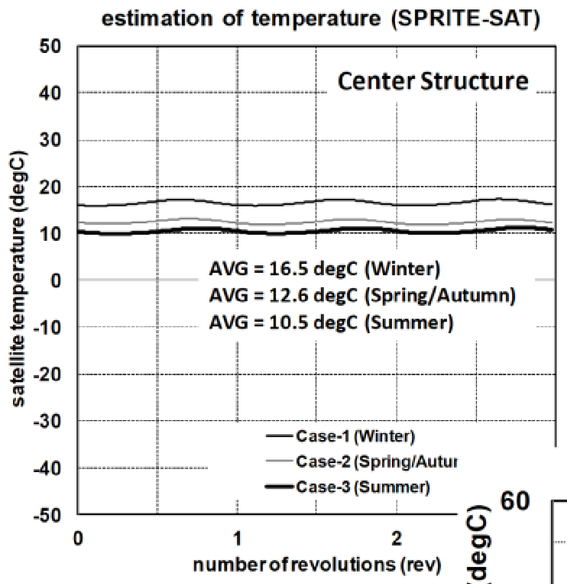


3. Case study I - 50kg microsatellites for SSO

3.5 Method of 2-node analysis



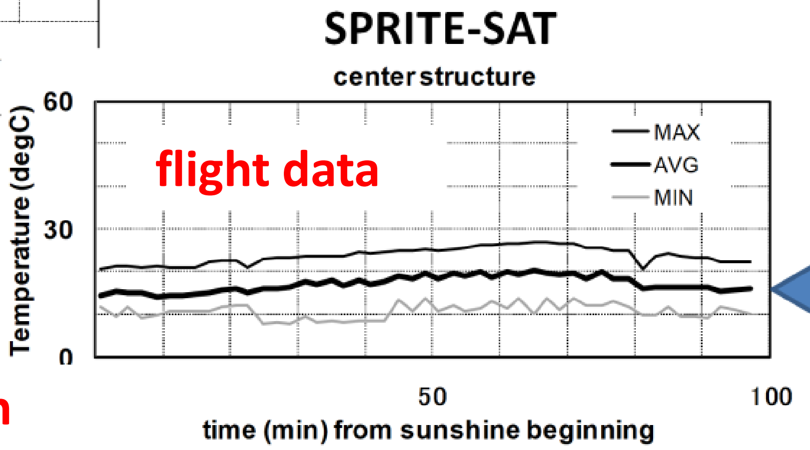
3.6 Analysis result and flight data for SPRITE-SAT



temperature of central pillar

SPRITE-SAT

estimate = **16.5** degC (in winter)



estimation was included in flight data range (15-20 degC)

estimation = 16.5 degC (Winter)

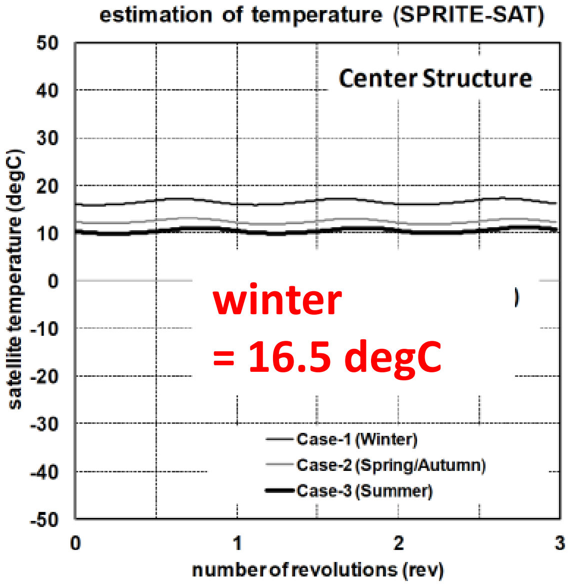
(C) TU,HU

3. Case study I - 50kg microsatellites for SSO

3.7 Analysis result and flight data for RISING-2

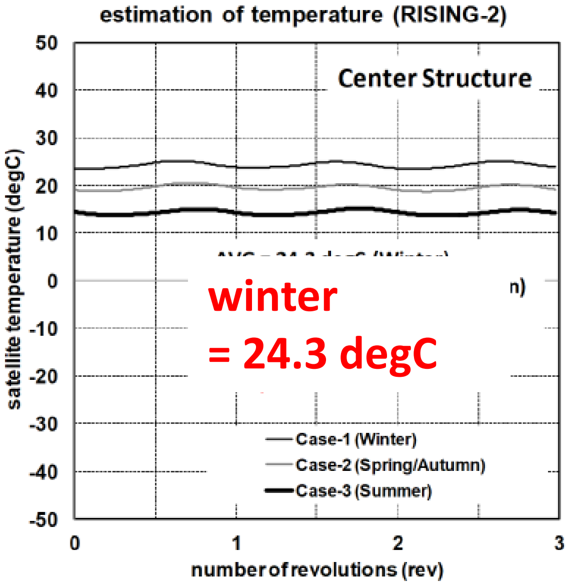
SPRITE-SAT

estimation



RISING-2

estimation



about +8 degC
is estimated
(better for battery)

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- SPRITE-SAT and RISING-2 have **similar external dimensions**, but **RISING-2** average temperature is **higher** than SPRITE-SAT
  - **power generation efficiency** of solar cell is higher -> **typical power consumption** of inside is **increased**
  - **heat radiation among inside parts** are **insulated w/o black paint**

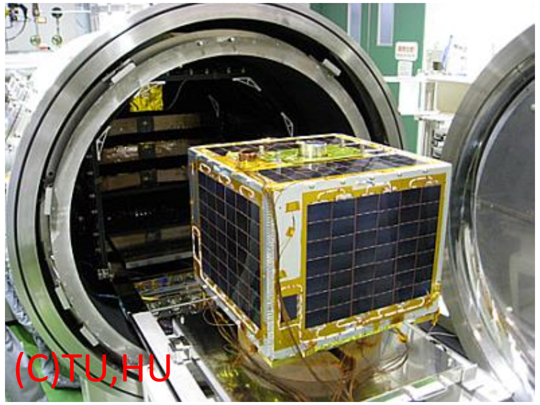
3. Case study I - 50kg microsatellites for SSO

3.8 Purpose of thermal vacuum chamber test

- 1. **Thermal Test** for onboard instruments and harness
  - all the instruments **must work normally** in the **lowest** and **highest** temperature
    - example) troubles by unstable oscillator clocks
  - mandatory for **deployable mechanisms** to ensure the safe temperature range

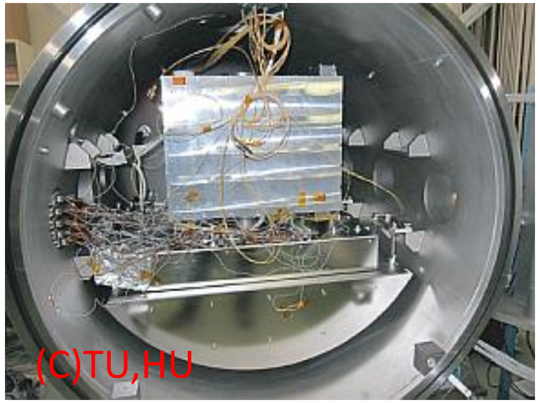
- 2. To decide the **coefficients** of heat transfer
  - **dummy mass** aluminum blocks can be also applied
  - number of **unknown variables** should be **decreased**
  - **epsilon** of **radiation** heat transfer can be measured by other instruments

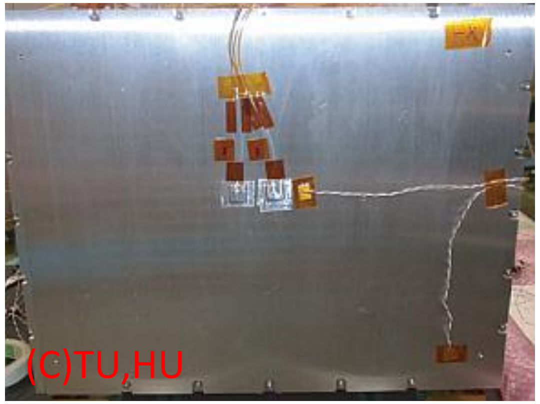
SPRITE-SAT



complex of radiation model

RISING-2



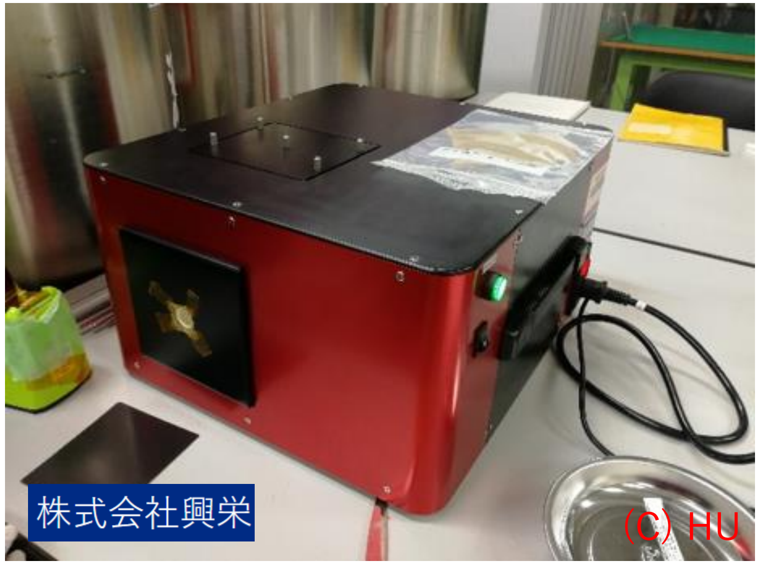


no radiation = good to decide **contact** coefficients

3. Case study I - 50kg microsatellites for SSO

3.8 Purpose of thermal vacuum chamber test

To measure
solar absorption α



Portable Spectral Solar Absorptance Measurement System PM-A2

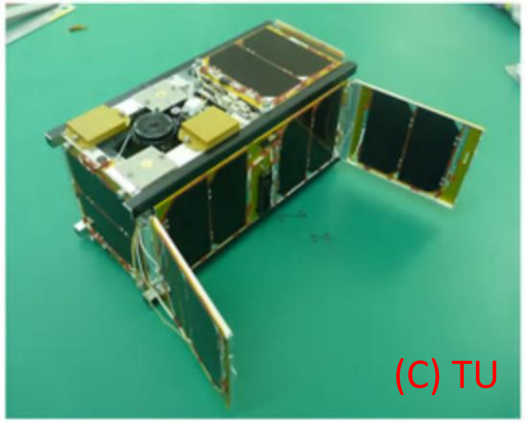
To measure
emissivity ε

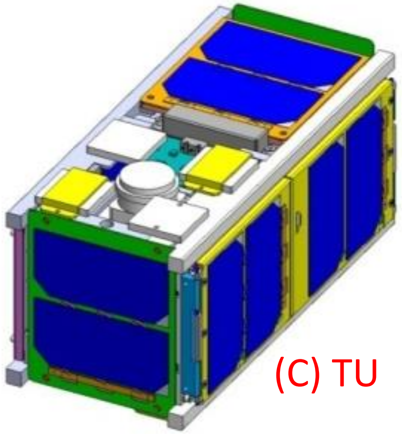
Thermo Fisher Scientific Nicolet iS50 FT-IR

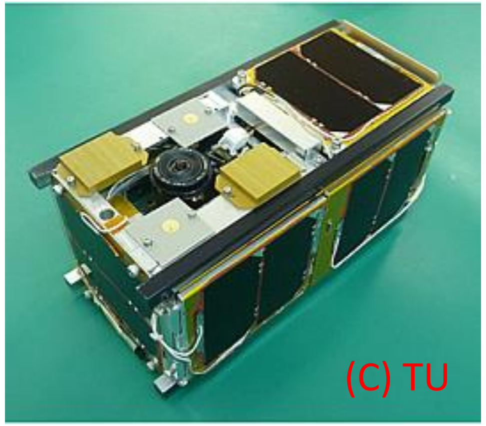
Photos: Hokkaido University
https://f3.eng.hokudai.ac.jp/microsat.html

4. Case study II -
2.6kg 2U CubeSat for ISS orbit

4.1 Summary of satellite specifications





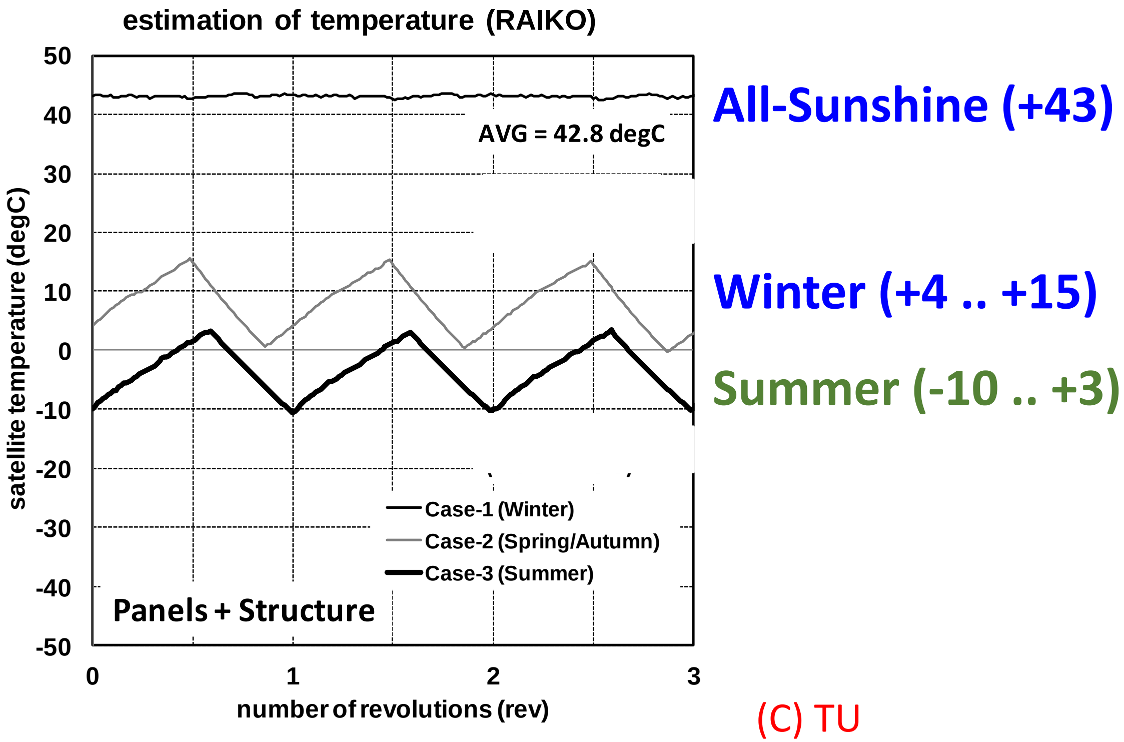


	RAIKO
Missions	Demonstration
Orbit	ISS, **400 km** height
Mass	**2.66 kg**
Size	10 x 10 x 22.7 cm (total)

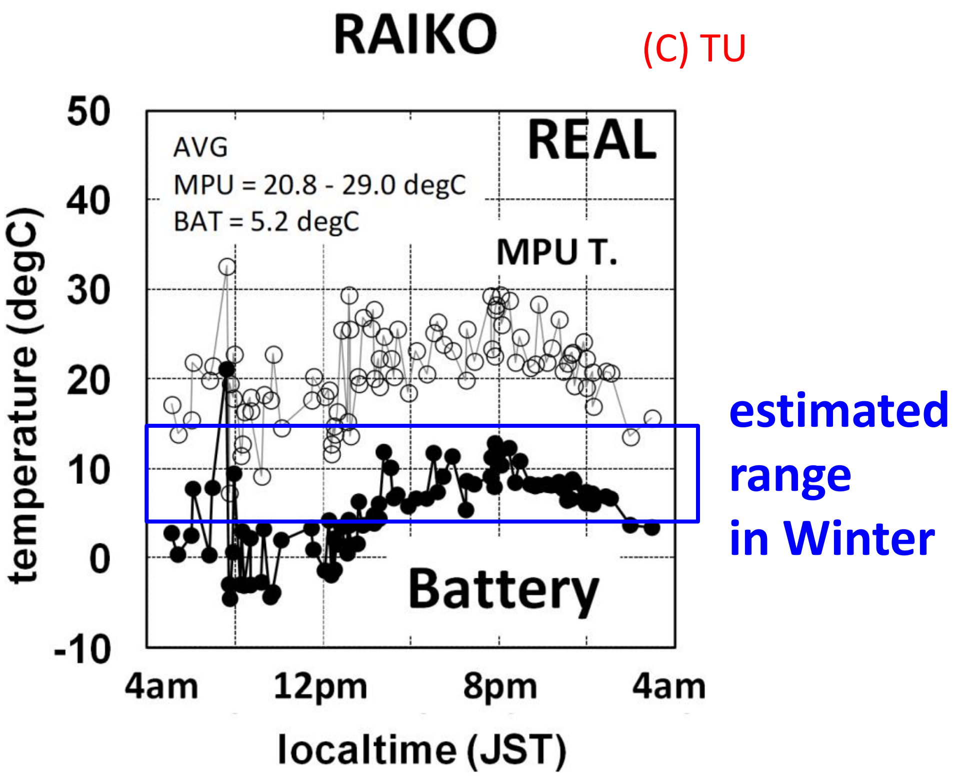
- **No special treatment** for thermal analysis
- **No insulation concept**, the entire structure and instruments are simply connected by stainless bolts
- **No paint** for internal instruments

4. Case study II - 2.6kg 2U CubeSat for ISS orbit

4.4 Analysis result and flight data for RAIKO



(C) TU



(C) TU

- Single node analysis (no insulation concept)

=> **43 degC** in all-sunshine phase

=> **+4 .. +15 degC** in winter phase

=> **-10 .. +3 degC** in summer phase

- Battery temp. was **from -5degC to +13degC**

[!] Lessons Learned: at least **battery module must have been insulated**. minus degC is risk for battery charge/discharge and lifetime

5. Conclusion

5. Conclusion

- **Section 1.3)** importance of estimating the satellite temperature and adjusting it.
  - battery temperature should be in normal temp. range
  - computers, sensors, mechanical parts also important
- result of the thermal analysis have an effect on the design of structure, location of onboard instruments, necessity of heaters
  - in early the phase of design, we need to start the analysis

- **Section 2)** theories are needed in thermal analysis
  - balance of heat input from the Sun and the Earth, and heat radiation to space
  - consumption power, concept of insulation
  - 2.3: insulation method, 2:4: radiation control by surface material, 2.7: power generation

5. Conclusion

- **Section 3)** case studies of 50 kg microsatellites
  - first satellite (SPRITE-SAT) had an ambiguous concept (insulation, black paint)
  - insulation concept is one of many thermal concepts, suitable method is different in each satellite

- **Section 4)** case study of CubeSat
  - battery temperature was often in lower than 0 degC
  - battery wasn't discharged normally in very cold situation => not a successful example
  - Lessons Learned: insulation to battery was necessary in this case => high temperature in all-sunshine phase is a risk

Thank you very much.

[Disclaimer]
The views and opinions expressed in this presentation are those of the authors and do not necessarily reflect those of the United Nations.