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The design requirements for Libyan imaging mini-satellite (LibyaSat-1)

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Abstract

In this paper we present the conceptual design of Libyan remote sensing satellite (LibyaSat-1) and its sub-systems requirements. LibyaSat-1 is a 300 kg mini satellite, which will be used to support high resolution multi-spectral earth imaging camera to fulfill the civilian needs. This satellite will operate at LEO of 775 km and will provide a resolution of 2.5 m for the panchromatic band and 10 m for the VIS/NIR bands with 30 km swath. We have presented the mission overview, mission operation concept and mission requirements. Moreover, the System Tool Kit (STK) simulation is used to show the ground trucks of LibyaSat-1 for three days and to find the contact numbers between LibyaSat-1 and both Murezeq and Tripoli stations. We have also presented the design of telemetry and command subsystem, code and data handling subsystem, electrical power subsystem, altitude orbit control subsystem, and structure subsystem.

Keywords

requirements, libyan, design, imaging, 1, mini, satellite, libyasat

Disciplines

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The Design Requirements for Libyan Imaging Mini-Satellite (LibyaSat-1)

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***Abstract*— In this paper we present the conceptual design of Libyan remote sensing satellite (LibyaSat-1) and its sub-systems requirements. LibyaSat-1 is a 300 kg mini satellite, which will be used to support high resolution multi-spectral earth imaging camera to fulfill the civilian needs. This satellite will operate at LEO of 775 km and will provide a resolution of 2.5 m for the panchromatic band and 10 m for the VIS/NIR bands with 30 km swath. We have presented the mission overview, mission operation concept and mission requirements. Moreover, the System Tool Kit (STK) simulation is used to show the ground trucks of LibyaSat-1 for three days and to find the contact numbers between LibyaSat-1 and both Murezeq and Tripoli stations. We have also presented the design of telemetry and command subsystem, code and data handling subsystem, electrical power subsystem, altitude orbit control subsystem, and structure subsystem.**

Keywords—mini-satellite; large satellie; sub-systems; Low Earth Orbit (LEO); payload.

I. INTRODUCTION

Large (conventional) polar orbiting satellites have a wet mass larger than 1000 kg and operate at altitudes of about 1000 km. These satellites are sun-synchronous, which means that any object on the orbit or the movement of the satellite will appear in the same position when viewed from the sun [1, 2]. This is important as it provides a constant amount of sunlight to solar cells on the satellite. Conventional satellites are mainly used for high resolution images. One example is the Quick-Bird satellite, which high spatial resolution, i.e., panchromatic is 61 cm GSD at Nadir and Multispectral is 2.4 meter GSD at Nadir [3]. As set out in Table 1, these large satellites are really costly, heavy and consume large amount of power.

Mini Satellites are small, have low power consumption and cheap as compared to conventional satellites. They have a mass ranging from 100 to 500 kg and consume power of about 53.2 W [4, 5]. They play a very important role in the field of remote sensing and they perform similar tasks to conventional satellites. The applications of remote sensing include land use, agriculture, environment monitoring, disaster assessment, research, and national security. An example of mini satellite is FORMOSAT-2, which is a Taiwanese imagery satellite with a mass of 693 kg (without original propellant) and operate in the Low Earth Orbit (LEO) at altitude of 891 km [3]. Another example is SPOT 5, which is a French remote sensing mini satellite. It is a sun-synchronous imaginary mini satellite that orbits at altitude 832 km. This satellite provides high resolution images for Libyan Centre for Remote Sensing and Space Science (LCRSSS) as part of commercial operations between France and Libya [6].

LCRSSS established in 1989 and is a governmental research organization, which belongs to a government space organization dedicated to the researches in remote sensing, space science, and earthquakes. This organization does not have its own remote sensing satellite and all images of Libya are taken by French SPOT satellites and then send to Libyan ground station for image processing. Fig. 1 shows an image of Tripoli that was captured by SPOT 6. In this paper, we present the design requirements for the most important sub-systems of the first Libyan mini satellite (LibyaSat-1). We will present conceptual design of LibyaSat-1 and the design requirements for the most important sub-systems. We will also list the designed and achieved budget for each sub-system.

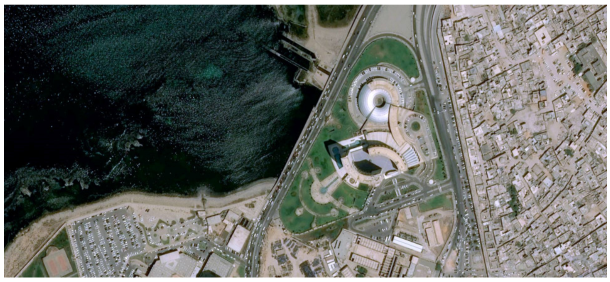

Fig. 1. Image of Tripoli taken by SPOT 6

TABLE 1 COMPARISON BETWEEN SATELLITES

Types	Mass (kg)	Cost (US $)	Power Consumption
Conventional	>1000	0.1-2 B	~ 1000 W
Medium	500-1000	50-100 M	~ 800 W
Mini	100-500	10-50 M	53.2W
Micro	10-100	2-10 M	35 W

II. MISSION OVERVIEW

The main mission of the LibyaSat-1 satellite under the remote sensing program is to acquire and monitor terrestrial, marine environment, desertification and resources throughout Libyan area and its surrounding waters, and possibly over other regions of the world for international cooperation, via satellite imaging data to fulfill the civilian needs.

- The remote sensing satellite program is one of the major satellite programs to be executed by Libyan center for remote sensing and space science for the first one phase space program of Libya.
- The remote sensing instrument is a push-broom electrical-optical type of sensor with spectral bands in the visible (VIS) and near infrared (NIR). The spectral bands include panchromatic and four

Land sat type multi-spectral bands. The ground sampling distances (GSD) are 2.5 m for the panchromatic band and 10 m for the VIS/NIR bands respectively.

- The swath is 30 km for all selected bands.

Table 2 shows the simulation results after testing LibyaSat-1 conceptual design at different altitudes. We can see that LibyaSat-1 is assumed to be sun-synchronous with an altitude of 775 km which can communicate with the ground station using RF links.

TABLE 2 LIBYASAT-1 CHARACTERISTTICS

Repeat (day)	561Km (SSO)	**775Km (SSO)**	888Km (SSO)	1000Km (SSO)
Repeat (day)	1	**3**	1	???
Rev/day	15	**14 1/3**	14	13.661
Circular Velocity (Km/Sec)	7.579	**7.465**	7.406	7.35
Ground Velocity (Km/Sec)		**6.750**		
Period (Min)	95.877	**100.347**	102.734213	105.1186
Eccentricity	0	**0**	0	0
Inclination (deg)	97.635	**98.497**	98.98	99.478
Width (Km) (FOR:+- 45 Deg)	1178	**1662.9**	1928.8	2200.1

III. MISSION REQUIRMENTS AND SATELLITE CHARACTERISTICS

Table 3 shows the key mission requirements and the systems specifications of LibyaSat-1. LibyaSat-1 satellite is expected to operate in sun synchronous LEO at altitude of 775 km with a total mass of no more than 300 kg and data storage volume of no less than 80 Gbits. The minimum expected life time of LibyaSat-1 is 5 years.

1) *Agility*

Roll and Pitch: The spacecraft bus shall be capable of providing a cross-track viewing tasking rate better than 24 degrees within 60 seconds regardless initial point.

Yaw: The spacecraft bus shall be capable of providing a Yaw viewing tasking rate better than 7 degrees within 60 seconds regardless of the initial point.

2) *Satellite Reliability*

0.6 at the end of the 5-year mission.

3) *Remote Sensing Insturment*

Panchromatic (PAN) with 4 Visible/NIR bands.

TABLE 3 MISSION REQUIREMENTS SPECIFICATIONS

Parameter	**Specification**	
RSI Mass (kg)	95	
Swath (km)	30	
RSI Power (watts) (Orbit Average)	75	
Download Rate (Mbits/s)	150	
	PAN (Panchromatic)	MS (Multispectral)
GSD (M) (Ground Sampling Distance)	2.5	10
Pixel (numbers)	12000	3000
A/D (Bits)	12	12
Mass For Satellite lift-off weight	No more than 300kg total weight (MAX)	
Lifetime	5 years	
Agility	Roll and Pitch: 24 deg/60s Yaw: 7 deg/60s	
Launcher	Falcon 1e	
Data storage volume (Gbits)	80	
FOR	+-45 deg	
Downlink Rate (X Band)	150 Mbps	
Downlink Rate (S Band)	2.0 Mbps	
Local Time	10:00 am ~ 10:30 am	
Reliability	0.6	

IV. SYSTEM ENGEERING

A. *Operational Concept*

The libyansat-1 operations are to be conducted from Tripoli Ground System (TGS). The TGS is composed of the Mission Operations Center (MOC), the Telemetry, Tracking, and Command (TT&C) stations, a Flight Dynamics Facility (FDF), a Mission Control Center (MCC), a Science Control Center (SCC), a Ground Communications Network (GCN), an X band Data Receiving Only Station, and an Image Processing Center (MOC) personnel, which will monitor and command the spacecraft using real-time health and status data. Command uplink files will be generated by MOC personnel, forwarded to the TT&C station and transmitted on an uplink data stream. Fig. 2 shows the location of TGS and MGS.

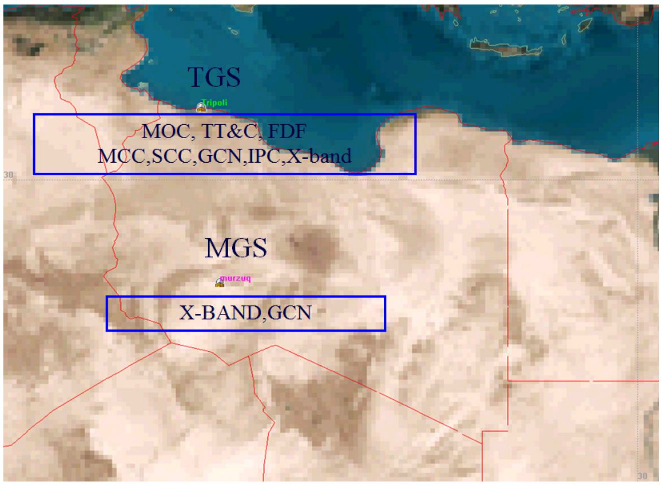

Fig. 2. The geographical locations of TGS and MGS.

There will be 2~4 contacts of the spacecraft with the TT&C stations on Libya per day. The remote sensing instrument will operate mainly over the region of Libya. Moreover, the equatorial descending time is around 10:00 am to 10:30 am local time. The baseline operation mode is simultaneous panchromatic and multi-spectral imaging, with real-time downlink data at a rate up to 150 Mbps via a dedicated X band channel. The STK simulation [7] is used to test LibyaSat-1 in the orbit.

Fig. 3, 4, 5 and 6 show the ground tracks for 3 days. LibyaSat-1 traces out a path on the earth surface, called its ground track, as it moves across the sky. As the earth below is rotating, the satellite traces out a different path on the ground in each subsequent cycle. We see that after three days the satellite traces out many different paths and covers most of the area.

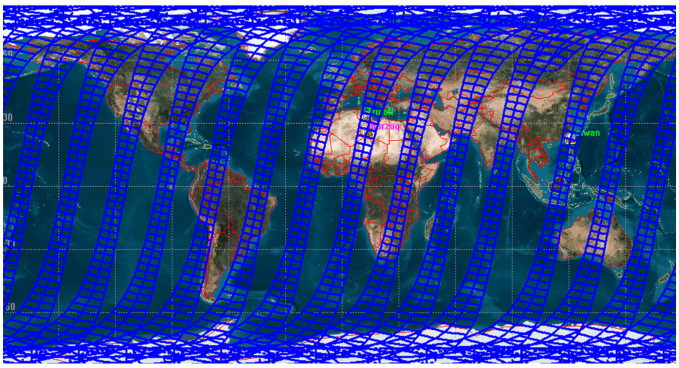

Fig. 3. Ground tracks of the first day

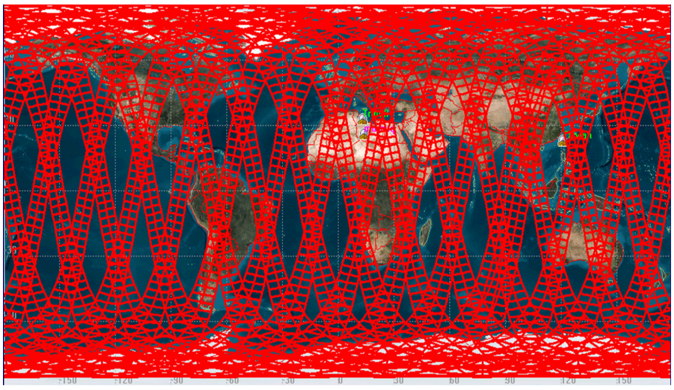

Fig. 4. Ground tracks of the second day

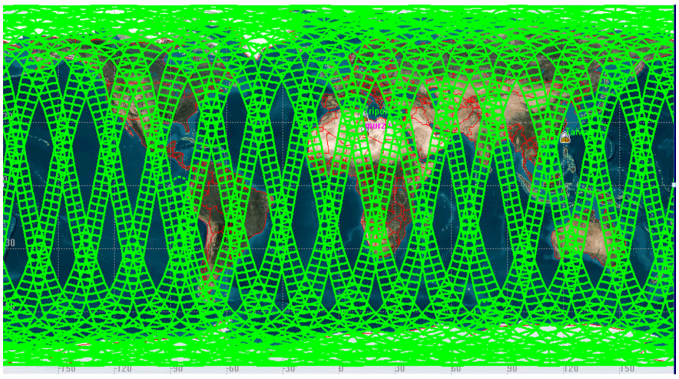

Fig. 5. Ground tracks of the third day

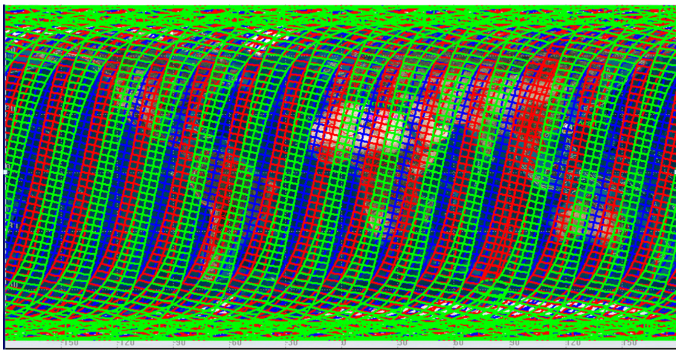

Fig. 6. Ground tracks (repeat period,3 days)

Fig. 7 and 8 show the contact number and time of LibyaSat-1 with Murezeq and Tripoli stations. We see that there are two contacts in the first and the third days between LibyaSat-1 and Tripoli and Murezeq stations at two different times. In the second day there is four contacts with both stations, which is a most double as compared to the first and the third days

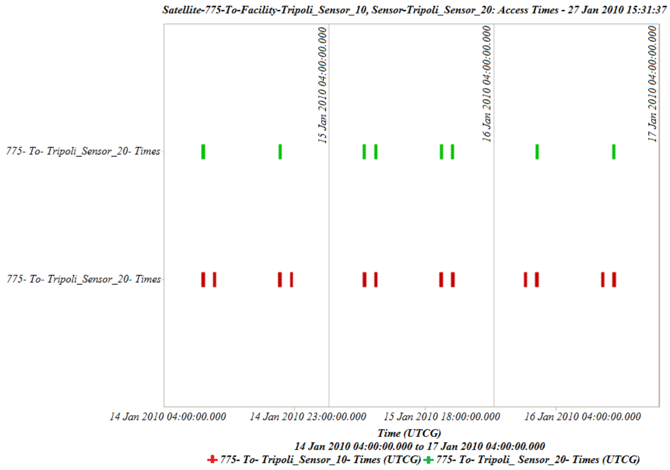

Fig. 7. The contact numbers between LibyaSat-1 at 775 km orbit and both Murezeq and Tripoli stations

775-To-Tripoli_Sensor_10

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	14 Jan 2010 09:37:17.751	14 Jan 2010 09:47:35.004	617.253
	2	14 Jan 2010 11:20:04.752	14 Jan 2010 11:21:51.770	107.018
	3	14 Jan 2010 20:48:58.915	14 Jan 2010 20:59:12.024	613.109
	4	14 Jan 2010 22:33:03.964	14 Jan 2010 22:34:20.195	76.232
	5	15 Jan 2010 09:04:39.166	15 Jan 2010 09:13:57.074	557.908
	6	15 Jan 2010 10:44:20.699	15 Jan 2010 10:52:14.526	473.827
	7	15 Jan 2010 20:16:40.001	15 Jan 2010 20:25:40.952	540.951
	8	15 Jan 2010 21:56:15.126	15 Jan 2010 22:04:28.490	493.364
	9	16 Jan 2010 08:33:01.662	16 Jan 2010 08:39:01.652	359.991
	10	16 Jan 2010 10:10:31.201	16 Jan 2010 10:20:22.505	591.304
	11	16 Jan 2010 19:45:32.739	16 Jan 2010 19:51:17.419	344.680
	12	16 Jan 2010 21:22:08.836	16 Jan 2010 21:32:10.814	601.978
Global Statistics				
Min Duration	4	14 Jan 2010 22:33:03.964	14 Jan 2010 22:34:20.195	76.232
Max Duration	1	14 Jan 2010 09:37:17.751	14 Jan 2010 09:47:35.004	617.253
Mean Duration				448.135
Total Duration				5377.615

775-To-Tripoli_Sensor_20

	Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)
	1	14 Jan 2010 09:38:50.293	14 Jan 2010 09:46:03.943	433.650
	2	14 Jan 2010 20:50:30.470	14 Jan 2010 20:57:39.118	428.647
	3	15 Jan 2010 09:06:29.176	15 Jan 2010 09:12:08.841	339.665
	4	15 Jan 2010 10:46:48.744	15 Jan 2010 10:49:47.483	178.739
	5	15 Jan 2010 20:18:31.374	15 Jan 2010 20:23:48.470	317.096
	6	15 Jan 2010 21:58:34.222	15 Jan 2010 22:02:07.476	213.254
	7	16 Jan 2010 10:12:08.895	16 Jan 2010 10:18:46.037	397.142
	8	16 Jan 2010 21:23:43.869	16 Jan 2010 21:30:34.167	410.298
Global Statistics				
Min Duration	4	15 Jan 2010 10:46:48.744	15 Jan 2010 10:49:47.483	178.739
Max Duration	1	14 Jan 2010 09:38:50.293	14 Jan 2010 09:46:03.943	433.650
Mean Duration				339.811
Total Duration				2718.492

Fig. 8. Contact time of LibyaSat-1 at 775 km with Murezeq and Tripoli stations

B. Telecom, Telemetry and Command Subsystem

The TT&C subsystem comprises of onboard s-band (2.039 GHz) receiver and s-band transmitter (2.215 GHz), transmitter for uplink and down link respectively. A s-band receiver is used to receive command signal from Murezeq ground station while s-band transmitter antenna is used to send a set of signals that show the status of the LibyaSat-1 resources and health, i.e., temperature, voltage and currents. All these signals are sent to the control center via ground station using RF system; see Fig. 9.

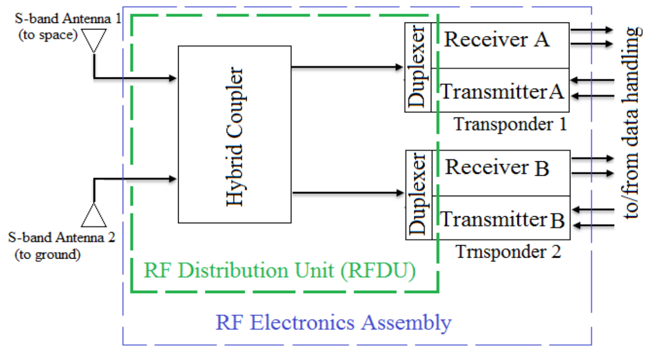

Fig. 9. TT&C subsystem block diagramof LibyaSat-1

The x-band subsystem is used for image data downlink. The 5.4 m x-band dish antenna and its pointing mechanism are used at Murezeq ground station. The link budgets for s-band up and down links for LibyaSat-1 are calculated and presented in [6]. Table 4 shows the up and down link budgets for s-band antennas.

TABLE 4 S-BAND UP AND DOWN LINK DUDGETS

Feature	**Uplink**	**Downlink**
Frequency	2.039 GHz	2.215 GHz
Satellite Altitude	775 km	775 km
Elevation Angel (deg)	10^0	10^0
Transmitter Power(Pt)	20 dBW	7 dBW
Ground Sation Antenna Gain (Gt)	32.3 dB	6.4 dB
EIRP Power	51.3 dBW	13.4 dBW
Space Loss	-165.93 dB	-166.63 dB
Atmospheric Losses	-0.5 dB	-0.5 dB
Rain Loss	-0.5 dB	-0.5 dB
Polarization Loss	-0.3 dB	-0.3 dB
Pointing Loss	-0.2 dB	-0.2 dB
On-Board Losses	-10 dB	-4.5 dB
System Noise Temperature(Ts)	27.9 dB/k	24.6 dB/k
Satellite Antenna Gain	5 dB	5 dB
Figure of Merit (G/T)	-22.88 dB/k	-18.2 dB/k
Bite rates	32 Kbps	2000 Kbps
C/No Received	79.57 dB-Hz	77.07 dB-Hz
Eb /No Received	34.52 dB	14 dB
Bit Error Rate (BER)	10^{-5}	10^{-5}
Required Eb /No	11.8 dB	9.6 dB
Implementation Loss	-2.5 dB	-1 dB
Margin	20.72 dB	3.4 dB

C. Code and Data Handling Subsystem

The Command and Data Subsystem is the "brain" of mini satellite. It is used to process and store information and data from all satellite sub-systems. It also sends commands to all sub-systems and equipment. The function of this sub-system is as following:

- ➢ Receives, validates, decodes, and distributes commands to other spacecraft subsystems.
- ➢ Gather, processes and formats spacecraft housekeeping and mission data for downlink or to be used by an on-board computer.
- ➢ The ideal C&DH system is one which has previously been proven on another spacecraft and which requires no or little modification for the mission under development (i.e. Heritage).

1) C&DH Requirements

a) Computer system

- *Processor:* up to 20 MIPS, with fault tolerance and supports real-time operating systems, i.e., eCOS, VxWorks.
- *SRAM / SDRAM:* 128 MBytes, protected with EDAC (single error correction, double error detection).
- *EEPROM:* 4 MBytes, storage of application code.
- *PROM / EEPROM:* 64 KBytes bootloader, SSR 50Gbits for SHO (Status of health), and file management raw image data.
- *Real Time Clock:* 1-Hz with synchronization to GPS clock.
- *FDIR:* Supervisory circuit / Watchdog timer.
- *Redundancy:* Full or partial implementation (TBD).

b) I/O Interfaces

- *Attitude Control Subsystem:* 2 x Sun sensor, 2 x Magnetometer, 4 x Gyroscope, 2 x Star camera, 2 x GPS receiver, 3 x Magnet torque, and 4 x Reaction wheel.
- *Thermal Control: up to 80 thermistors.*

c) Telemetry and Control

- *S- Band* :
  - *1-* CCSDS (Consultative Committee for Space Data Systems) communication protocol with uplink of 32 Kbps and downlink of 2 Mbps.
  - *2-* Telecommand decoder, with local memory for command storage, authentication and configuration.
  - *3-* Compliant with packet telemetry and channel coding standards CCSD102.0-B-5 and CCSDS101.0-B-5.
- *X-Band (Payload):* 150 Mbps (Downlink) – shown here for completeness.

d) Power Specification

- *28V unregulated voltage input*
- *Over-current and under-voltage protection*
- *DCDC switch-mode converters and linear regulators:* 5V, 3.3V and possibly 1.5V (for on-board electronics) and +/-12V for Analog.
- *Power consumption:* 45 Watts, in continuous mode (exact figure is to be determined) and about 1.8 A maximum input current (continuous mode).

e) Miscellaneous:

- *Mass:* 20kg (approx.).
- *Operating Condition:* military (i.e. -55°C to 125°C).
- *Component Selection:* space or military-grade

- *Radiation-Hardened:* up to 50KRad, depending on mission requirements (e.g. low-earth orbit, mission life).

D. Electrical Power Subsystem

Electronic Power Sub-system (EPS) is designed to provide a sufficient power to all subsystems including payload to ensure a reliable operations of all equipment and components during the mission. EPS consists of batteries, solar panel, Battery Charge Regulator (BCR), Power Conditioning Module (PCM), Power distribution Unit (PDU), and Battery Monitoring (BM); see Fig. 10. The multi-junction GalnP/GaAa solar cells are chosen to be used as a power generator for LibyaSat-1 and the battery will be Lithium-ion. Table 5 and 6 show different types of solar cells and batteries.

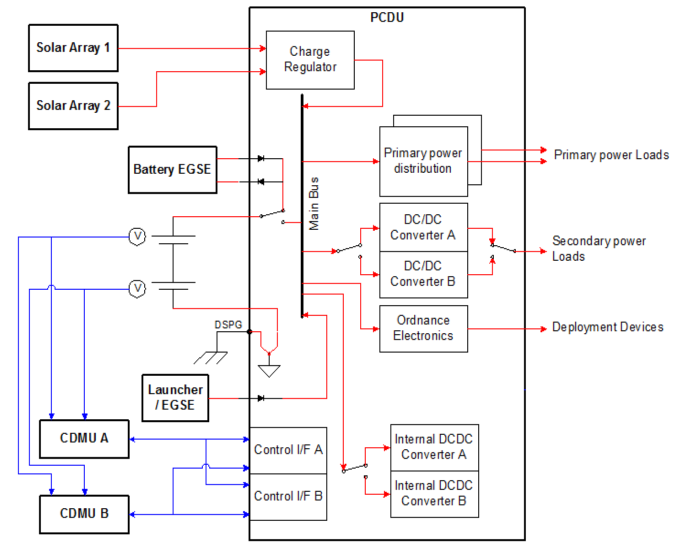

Fig. 10. EPS Architecture

TABLE 5 DIFFERENT TYPES OF SOLAR CELLS

Cell Type	**Silicon**	**Gallium Arsenide**	**Multi-junction Galnp**	**Indium phosphide**
Efficiency	14%	18%	27%	19%
Radiation degradation	High	Medium	Medium	Low
Cost	Low	High	High	High

TABLE 6 DIFFERENT BATTERY TYPES

Secondary Battery Couple	**Specific energy density (W. hr/kg)**	**Status**
Nickel-cadmium	15-30	Space-qualified, extensive database
Nickel-hydrogen (individual pressure vessel)	35-43	Space-qualified, good database
Nickel-hydrogen (common pressure vessel)	40-56	Space-qualified for GEO and planetary
Nickel-hydrogen (common pressure vessel)	43-57	Space-qualified
Lithium-ion (LiS02, LiSF, LiS0CI2)	70-110	Space-qualified
Sodium-sulfur	140-210	Under development

1) EPS requirements

- 3% margin from the whole power budget for every orbit.
- Primary power bus voltage: unregulated power of 28v (+5.6 V/-4 V); see Fig. 11.
- Secondary power bus voltage: using converters to reduce 28 V to:
  a) +4.8v ~ +5.6 V
  b) +14.25 ~ +15.75 V
  c) -15.75v ~ -14.25 V

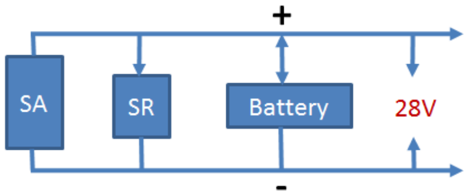

Fig. 11. DET System

- Battery low voltage protection: to prevent S/C from recovery until solar arrays get sufficient power and charge batteries.
- Battery high voltage protection: battery voltage > 34 V, terminate all charge regulators.
- Battery charge management: the operating system for batteries charging must be done automatically by special flight software without a ground uplink command.
- Power distribution short circuit protection: all equipment and payload instruments must be over- current protected using Solid-state switch (MOSFET).
- Deployment control using NEA.
- Redundancy: PCDU shall be cold redundant except for power distribution and charge regulator.

For LEO we expect the battery's DOD to be 40-60% for NiH2 technology compared to 10-20 % for NiCd. We base these expectations on the average DOD over 24 hours and assume the batteries are fully recharged at least once during this period. The number of batteries, N, may be equal to one for this calculation if you simply require a battery capacity .Two to five batteries are typical.

We must have at least two (unless the battery uses redundant cells) because the spacecraft needs redundant operation if one unit failed. But more than five batteries require complex components for recharging.

2) Power Profile

Table 7 shows the mass and power consumption for EPS. These numbers were calculated taking in to account different assumptions such as when we do not operate the payload and x-band transmitter.

Ps/c basic power consumption when we do not operate the payload and the transmitters.

P_{SRI} remote sensing instrument for power consumption = 50 watt.

Transmitter:

- Power consumption for X-band transmitter 55 watt & APM 5 watt.
- Power consumption for S-band transmitter 35 watt.

- S-band receiver power consumption 8 watt is included in Ps/c as this receiver will be always on.

Fig 12. below gives a good idea about the operation of the power profile.

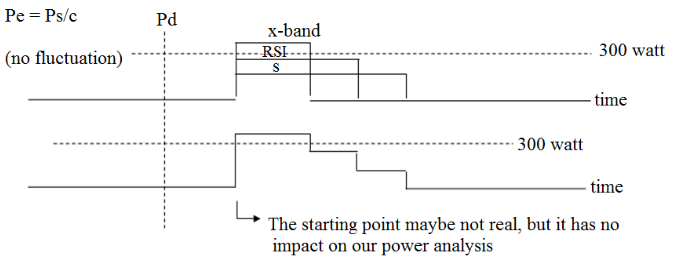

Fig. 12. The operation of power profile for s/c

TABLE 7 MASS AND POWER CONSUMPTION FOR EPS COMPONENTS

Component	**Mass**	**Consumption**
Battery	7 kg	0 watt (passive device)
S/A	16 kg	0 watt (generator device)
PCDU	15 kg	17 watt
Harness	22 kg	3 watt(dissipation power)

E. Altitude Orbit Control Subsystem

The aim of Attitude and Orbit Control Sub-systems (AOCS) is to control the attitude of the micro satellite depends on the operation mood in the mission orbit by using different sensors and actuators. As set out in Table 8, AOCS provides an altitude pointing accuracy and control stability of better than 0.20 (3δ) for allaxis and pointing knowledge less than 0.036^0.

TABLE 8 AOCS DESIGN PARAMETERS

Pointing accuracy	< 0.2 deg -- 3δ ,3D
Pointing knowledge	< 0.036 deg
Positioning knowledge	< 100 m
Jitter	< 0.33 arc sec/ 373.7 µsec
Agility	Cross track maneuver rate should be better than 30 degrees within 60 seconds
	Along track maneuver rate should be better than 30 degrees within 60 seconds
	Yaw maneuver rate should be 10 degrees within 60 seconds

F. Structure Subsystem

This Subsystem provides mechanical support and alignment for all equipment and components on the mini satellite. It is also used to provide a thermal conductivity and its body is used as a ground for electrical devices. Fig. 13 shows the configuration of different components on LibyaSat-1. Moreover, Tables 9 and 10 show the mass budget and selected material respectively.

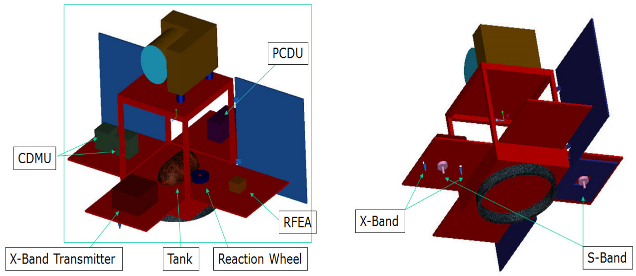

Fig. 13. The proposed components layout of LibyaSat-1

TABLE 9 MASS BUDGET

System	**Current mass (kg)**	LV Adapter	10
		Bus Total	234
AOCS	34		
C&DH	20	RAL CAM1	50
TTC	13	Total	284
EPS	60	Balance	-
RCS	30	Satellite Total	284
SMS	55	Launch Max.	415
TCS	12	Margins(%)	31.56%

TABLE 10 MATERIAL SELECTION

Item	**description**
Top panel	Honeycomb Panel
Top frame	Milled Al Alloy Ring
Bottom cone panel	Honeycomb Panel
Tank support panel	Milled Al Alloy Plate
Wall panel	Honeycomb Panel
longeron	Milled Al Alloy Plate

G. Satellite Chararcterstics

Table 11 and 12 show the assigned and obtained budgets for mass and power respectively. We can see that the total mass and power of all designed subsystems is within the budget but the margin is less than the expected. Compared to all sub-systems, the EPS has the higher mass because of the batteries and solar cells. In terms of power consumption, the TTC sub-system has the highest power consumption as compared to all subsystems on LibyaSat-1. This is because of the number of antennas used and their constant state of activeness to keep tracking, download images and communicate with ground station.

TABLE 11 MASS BUDGET FOR ALL SUBSYSTEMS

Sub-system	**Mass (kg) System flow-down**	**Design Mass (kg)**
C&DH	42.5(17%)	20
TT&C	22.5(9%)	13
EPS	45(18%)	60
Thermal	10(4%)	12
Structure	37.5(15%)	55
Propulsion	17.5(7%)	30
AOCS	25(10%)	34
FSW	0	0
RSI	50(20%)	50
TOTAL Mass	250	**274**
Max Mass	300	300
Budget	50	**26**

TABLE 12 POWER BUDGET FOR ALL SUBSYSTEMS

Sub-system	Power (W) System flow-down	Design power (W)
C&DH	77.5(31%)	45
TT&C	27.5(11%)	104.3
EPS	17.5(7%)	20
Thermal	27.5(11%)	10
Structure	0	0
Propulsion	12.5(5%)	13.8
AOCS	37.5(15%)	47
FSW	0	0
RSI	50(20%)	50
TOTAL power	250	290.1
Av power	300	300
Budget	50	10.1

V. CONCLUSION

We have presented a conceptual design of LibyaSat-1 and the design requirements of its important subsystem. We have simulated the model of LibyaSat-1 at different LEO altitudes including the 775 km. We found out that the most suitable altitude is 775 km. We have also measured the contact time and number of LibyaSat-1 with both Tripoli and Murezeq stations. The total designed mass and power for all sub-systems were within the assigned budget. The total mass of LibyaSat-1 is 74 kg and total required average power is 190.1 W.

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